High-Fidelity Simulations of a Flexible Flapping Wing in Forward Flight

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Flapping flight is commonly used in nature at low Reynolds numbers. This work investigates the physical aspects of forward flight for flapping unmanned aerial systems. The kinematics discussed in this work is based on an anticlockwise eight-shaped flapping cycle. First the complex aerodynamic structures and their evolution (leading edge and trailing edge vortices) are presented. Then, the typical mechanisms for generating the lift and thrust are discussed with emphasis on the unsteadiness of the flow features during both the downstroke and upstroke phases.

The inhomogeneity of the wing in terms of mass distribution is studied for the rigid case and it is shown that a higher density at the wing root is beneficial in respect to the mechanical power required to achieve the forward flight.

The assumption of rigid wing is then removed and the elastic deformation and its effects are investigated. The relative importance of inertial/aerodynamic effects is assessed with the inclusion of structural geometric nonlinearities.

I. Introduction

INSECTS present excellent flight performance\textsuperscript{1,2} and are the ideal candidates for bio-inspired flapping unmanned aerial systems. The hovering and forward flight of insects requires a high flapping frequency which depends on the insect’s weight. The insect must be able\textsuperscript{3} to accelerate the wing in one direction (downstroke), decelerate it and reverse the motion (supination), perform the upstroke, decelerate the wing again and reverse the motion for the next downstroke (pronation). These accelerations and decelerations imply an increase/decrease of the kinetic energy of the wing. To achieve a high efficiency of flight and decrease the amount of energy required by the flapping motion, the wing must not be rigid;\textsuperscript{4–6} the structural deformation brings changes of the effective angle of attack, camber, the vortex intensity, and the net force. Moreover, the insects do not present muscles on the wings: they are passively deformed under the inertial and aerodynamic forces.

An effective design of flapping unmanned aerial systems will try to reproduce the essential aspects of the insects’ biological features required for efficient flight with focus on the maximization of the payload and minimization of the power required to flap the wings in forward and hovering flight.

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In nature, small flyers flap the wings to produce lift and thrust\textsuperscript{2,7}. It has been shown that complex unsteady aerodynamics take place\textsuperscript{2,8,9} including strong vortical structures at the wing leading edge\textsuperscript{10}. This was shown to be responsible for the high lifting forces\textsuperscript{11} generated during the downstroke. On the structural side, small natural flyers possess a high degree of variability of the structural flexural properties due to both membranes and veins\textsuperscript{12}.

Flexibility and nonlinear geometric effects also impact the deformation of the wing for man-made flapping aerial systems. Moreover, some types of anisotropy\textsuperscript{13,14} have been shown to modify the structural bending-torsion coupling and the consequent aeroelastic deformation. The complex unsteady motion, a result of the interaction between inertia, elastic and aerodynamic forces (which depend on the deformation of the wing), can be opportune modified with consequent increase of the thrust and reduction of power expenditure.

The relative importance\textsuperscript{13} of inertial and aerodynamic contributions on the required power, is affected by the inhomogeneity\textsuperscript{15} of the material properties which are observed in the nature. This work explores this aspect by the means of high-fidelity CFD and CSD solvers.

II. Contribution of the Present Study

This work investigates how anisotropy and inhomogeneity can modify the performance of Flapping Unmanned Aerial Systems. The analyzed kinematics will be a forward flapping motion described by an anticlockwise 8 cycle similar to the one previously investigated in the literature\textsuperscript{16}. The investigations will involve high-fidelity CFD and CSD capabilities with inclusion of viscous effects and geometric nonlinearities on the aerodynamic and structural sides respectively.

Energy expenditure is one of the major challenges in the design of efficient flapping systems. Thus, the power required to maintain a constant forward speed will be numerically evaluated. Answers will be provided to fundamental questions regarding the best choice of inhomogeneous material properties when geometric nonlinearities are present and the wing passively deforms under aerodynamic and inertial loadings. This study will then provide guidelines for the design of flapping unmanned aerial systems with particular focus on forward flight.

The following sections describe the theoretical highlights and present results on both structural and aerodynamic effects, with emphasis on the leading edge, trailing edge, and tip vortex generation and evolution and the mechanisms that create lift and thrust during the stroke. Quantitative indications will also be provided with time-domain high-fidelity CSD and CFD solvers.

III. Rigid Wing and Prescribed Kinematics

Vertical Stroke Plane and Kinematics Relative to the Anticlockwise 8 Flapping Cycle

Forward flight is assumed to take place with a constant velocity. In other words, the freestream velocity $V_\infty$ (see Fig. 1) is kept constant in magnitude and direction. The goal is to simulate an anticlockwise 8 flapping cycle with vertical stroke plane $\chi$ (see Fig. 1). The parameters that need to be prescribed, for a given rectangular wing (see Figs. 2 and 3), are the flapping angle $\varphi$, the deviation angle $\gamma$, and the pitch angle $\psi$. The deviation angle $\gamma$ is assumed to be exactly zero at the pronation, supination and in the middle of each half stroke. Moreover, The flapping angle $\varphi$ corresponding to the supination (indicated with $\varphi_S$) is equal in magnitude but opposite of the flapping angle corresponding to the pronation (indicated with $\varphi_P$). If $\Phi$ indicates the stroke amplitude, it is

$$\Phi = \varphi_P - \varphi_S = \varphi_P - (-\varphi_P) = 2\varphi_P$$  

(1)

The time $t = 0$ corresponds to pronation, whereas $t = T$ indicates the time when the flapping cycle is completed (i.e., another pronation phase is reached). It is assumed that the time required to complete the downstroke phase is equal to the time required to complete the upstroke phase. The “O” portions (identified by the second half of the upstroke and the first half of the downstroke or identified by the second half of the downstroke and the first half of the upstroke) of the “8” cycle are perfectly identical (see Fig. 1). The axes $x_S$ and $y_S$ are on the stroke plane and $z_S$ is perpendicular to it. The axis $y_S$ is also the intersection between the stroke and horizontal planes. The wing (see Fig. 2) is assumed to be rectangular and perfectly planar (i.e., no built-in twist or camber are present). The thickness $s_w$ is equal to 0.2mm. It is possible to demonstrate that the aerodynamic angle of attack is influenced by all Euler’s angles $\varphi$, $\gamma$, and $\psi$ (see...
The flapping angle is assigned with the following law:

$$\varphi (t) = \Phi \cdot \cos \left( 2\pi \frac{t}{T} \right)$$  \hspace{1cm} (2)$$

The stroke amplitude $\Phi$ is assumed to be equal to 60°. From Figs. 1 and 3 it is possible to observe that the deviation angle, $\gamma$, is negative at the first half of the downstroke. After the middle half stroke point the deviation angle is positive. At the beginning of the upstroke it is again negative, and at the end of the upstroke it is positive. The law chosen to describe this behavior is the following:

$$\gamma (t) = -\gamma \cdot \sin \left( 4\pi \frac{t}{T} \right)$$  \hspace{1cm} (3)$$

where $\gamma$ is assumed to be 10°. The following kinematic law is adopted for the pitch angle $\psi$:

$$\psi (t) = \psi_p - \ddot{\psi} \cdot \sin \left( 2\pi \frac{t}{T} \right)$$  \hspace{1cm} (4)$$
Figure 3. Definition of Euler's angles: flapping angle $\varphi$, deviation angle $\gamma$, and pitch angle $\psi$.

$$\Phi = 60^\circ$$

$$\varphi(t) = \frac{\Phi}{2} \cos(2\pi \frac{t}{T})$$

Figure 4. Anticlockwise 8 flapping cycle: kinematic law of the flapping angle.

where $\overline{\psi}$ is assumed to be $32^\circ$. $\psi^P$ is the rotation angle at the pronation phase. Its value is assumed to be $10^\circ$. The kinematics laws are graphically shown in Figs. 4-6.

Alternative Closed Paths Performed by the Point $T$

As it is possible to see from Fig. 7 the point $T$ draws a closed path on a sphere. For simplicity this path is projected on a $y_S = \text{constant}$ plane. It is possible to show that the parametric expression of the curve on the plane $x_S - z_S$ is:

$$x_S^T (t) = - \cos(\gamma(t)) \sin(\varphi(t))$$
$$z_S^T (t) = \sin(\gamma(t))$$

Note that this law does not depend on the pitch angle temporal law $\psi(t)$. Fig. 8 shows alternative 2D closed paths for given temporal laws of the flapping and deviation angles.

A. Aerodynamic Angle of Attack

It is useful to calculate the aerodynamic angle of attack. This is defined in a plane perpendicular to the stroke plane. Its definition and analytical formula for the anti-clockwise eight-shaped cycle are reported in
Fig. 5. Anticlockwise 8 flapping cycle: kinematic law of the deviation angle.

\[ \gamma(t) = -\gamma \sin(4\pi \frac{t}{T}) \]

Fig. 6. Anticlockwise 8 flapping cycle: kinematic law of the rotational motion \( \psi \).

\[ \psi_P = 10^\circ \quad \psi = 32^\circ \]

\[ \psi(t) = \psi_P - \psi \sin\left(2\pi \frac{t}{T}\right) \]

**IV. Planar Rigid Wing: Power Calculation**

The kinematics laws are assumed prescribed according to Figs. 4-6. Thus, Euler’s angles \( \varphi(t) \), \( \gamma(t) \), and \( \psi(t) \) are defined as a function of time and are known. These angles are used to calculate the angular velocity \( \omega \) which is an intermediate result necessary to determine the power expenditure. Using Fig. 3 and applying the transformation of coordinate systems, it is possible to demonstrate that the angular velocity \( \omega \) is:

\[
\omega = \omega_{x,w} \hat{i}_w + \omega_{y,w} \hat{j}_w + \omega_{z,w} \hat{k}_w = \left( \dot{\gamma} \cos \psi - \dot{\varphi} \cos \gamma \sin \psi \right) \hat{i}_w + \left( \dot{\varphi} \sin \gamma + \dot{\psi} \right) \hat{j}_w + \left( \dot{\gamma} \sin \psi + \dot{\varphi} \cos \gamma \cos \psi \right) \hat{k}_w
\]

(6)
Angles of attack on a plane $y_S = \text{const}$ for different instants along the curve $\Theta$.

**Figure 7.** Anticlockwise 8 flapping cycle: visual representation of the kinematics.

(\textit{present laws})

<table>
<thead>
<tr>
<th>$\varphi(t)$</th>
<th>$\gamma(t)$</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\displaystyle \frac{\pi}{6} \cos \left( \frac{2\pi t}{T} \right)$</td>
<td>$\displaystyle -\frac{\pi}{18} \sin \left( \frac{4\pi t}{T} \right)$</td>
</tr>
<tr>
<td>$\displaystyle \frac{\pi}{12} \left[ \cos \left( \frac{2\pi t}{T} \right) + \cos \left( \frac{2\pi t}{T} \right) \right]$</td>
<td>$\displaystyle \frac{\pi}{18} \sin \left( \frac{4\pi t}{T} \right)$</td>
</tr>
<tr>
<td>$\displaystyle -\frac{\pi}{18} \sin \left( \frac{2\pi t}{T} \right)$</td>
<td>$\displaystyle -\frac{\pi}{18} \sin \left( \frac{2\pi t}{T} \right)$</td>
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Figure 8. Closed paths obtained by changing the temporal laws of the flapping and deviation angles.

in the wing coordinate system. An equivalent result, in the stroke plane is:

$\omega = \omega_{x_S} i_S + \omega_{y_S} j_S + \omega_{z_S} k_S = \left( \dot{\gamma} \cos \varphi - \dot{\psi} \cos \gamma \sin \varphi \right) i_S + \left( \dot{\gamma} \sin \varphi + \dot{\psi} \cos \gamma \cos \varphi \right) j_S + \left( \dot{\varphi} + \dot{\psi} \sin \gamma \right) k_S$

\hspace{10cm} (7)

Equations 6 and 7 are valid for any rigid wing planform. Besides the angular velocity, the forces and moments applied on the wing need to be considered when the power is calculated. The forces applied on a differential element $dm_w$ of wing are presented in Fig. 10. This figure depicts a rectangular wing, but the derivation is valid for any planform.

Point $B$ (the base) is kinematically constrained to maintain a constant horizontal velocity and to enforce symmetry. This means that there are three reactions $R_{x_S}$, $R_{y_S}$, and $R_{z_S}$ which change with time.

Rotations at point $B$ are completely allowed. Thus, no reaction moments are present. External moments
Figure 9. Angle of attack for a rectangular wing performing an anti-clockwise cycle (see also Figure 7)

Figure 10. Planar rigid rectangular wing and free body diagram (dynamic case).

\( M_{xS}, M_{yS}, \) and \( M_{zS} \) are applied to make the wing flap. In this work instead of applying input moments and determining the corresponding kinematics, the kinematics defining temporal variations for \( \varphi, \gamma, \) and \( \psi \) are prescribed (see Figs. 4-6). The prescription of the motion will create accelerations which are associated with inertial forces. Moreover, the motion in the fluid will also create other forces. The dynamic equilibrium between all these forces can be used to determine the support reactions \( R_{xS}, R_{yS}, \) and \( R_{zS} \).

The equilibrium equations are now written. The translational equilibrium (note the negative sign for the inertial forces is a consequence of the choices made in the free body diagram of Fig. 10) is defined as follows:

\[
R + F^{\text{aero}} - F^{\text{in}} = 0
\]

Similarly, the rotational equilibrium under dynamic conditions requires the satisfaction of the following vectorial equation:

\[
M + M^{\text{aero}} - M^{\text{in}} = 0
\]
where the **moment of the aerodynamic forces** is defined as

\[
M^{\text{aero}} = \int_{\text{wing}} r \times dF^{\text{aero}}
\]  

(10)

The infinitesimal aerodynamic force \( dF^{\text{aero}} \) is given by a load per unit of surface (normal and shear direction) \( P^{\text{aero}} \) multiplied by the wing’s infinitesimal area \( dA_w = dx_w dy_w \). The moment is then rewritten as

\[
M^{\text{aero}} = \int_{\text{wing}} r \times P^{\text{aero}} dA_w
\]  

(11)

In the practice, the CFD code provides the aerodynamic forces in a finite number of locations. If \( N_{\text{CFD}} \) is the number of locations at which the aerodynamic forces are given, equation 11 is replaced by the equivalent **discrete** counterpart:

\[
M^{\text{aero}} = \sum_{i=1}^{N_{\text{CFD}}} M^{\text{aero}}_i = \sum_{i=1}^{N_{\text{CFD}}} r_i \times F_i^{\text{CFD}}
\]  

(12)

where \( r_i \) is the position vector of the generic location where the generic computed aerodynamic force \( F_i^{\text{CFD}} \) is applied. The **power** \( P^{\text{aero}} \) associated with the aerodynamic forces is calculated by scalarly multiplying \( M^{\text{aero}} \) with the angular velocity vector:

\[
P^{\text{aero}} = M^{\text{aero}} \cdot \omega = M_{x_z}^{\text{aero}} \omega_{x_z} + M_{y_z}^{\text{aero}} \omega_{y_z} + M_{z_z}^{\text{aero}} \omega_{z_z}
\]  

(13)

or, using equation 7:

\[
P^{\text{aero}} = M_{x_z}^{\text{aero}} \left( \dot{\gamma} \cos \varphi - \dot{\psi} \cos \gamma \sin \varphi \right) + M_{y_z}^{\text{aero}} \left( \dot{\gamma} \sin \varphi + \dot{\psi} \cos \gamma \cos \varphi \right) + M_{z_z}^{\text{aero}} \left( \dot{\varphi} + \dot{\psi} \sin \gamma \right)
\]  

(14)

Now the focus is on the inertial forces and their contributions. The inertial force \( dF^{\text{in}} \) is given by

\[
dF^{\text{in}} = \mathbf{r} \cdot dm_w
\]  

(15)

where \( \mathbf{r} \) is the **position vector** of a generic point on the wing surface. The moment of the inertial forces about point \( \mathbf{B} \) is:

\[
M^{\text{in}} = \int_{\text{wing}} r \times dF^{\text{in}} = \int_{\text{wing}} r \times \mathbf{r} \cdot dm_w
\]  

(16)

To determine the moment \( M^{\text{in}} \) the position vector and its time derivative need to be calculated by considering the assumption of rigid wing. After some algebra and with the support of the symbolic software MATHEMATICA, the power \( P^{\text{in}} \) associated with the moments of inertial forces can be calculated (see equation 7):

\[
P^{\text{in}} = M^{\text{in}} \cdot \omega
\]  

(17)

or

\[
P^{\text{in}} = +I_{z_w z_w} \ddot{\varphi} \ddot{\varphi} \cos ^2 \gamma \cos ^2 \psi - I_{y_w z_w} \ddot{\gamma} \ddot{\varphi} \cos ^2 \gamma \cos \psi + I_{y_w y_w} \ddot{\gamma} \ddot{\varphi} \sin \gamma \sin \psi \cos \psi \\
- I_{y_w y_w} \ddot{\gamma} \ddot{\varphi} \sin \psi \cos \psi - I_{y_w y_w} \ddot{\gamma} \ddot{\varphi} \cos \gamma \sin \psi \cos \psi \\
- I_{y_w z_w} \ddot{\gamma} \ddot{\varphi} \sin (2\gamma) \cos \psi - I_{y_w z_w} \ddot{\gamma} \ddot{\varphi} \cos \gamma \cos \psi + I_{y_w y_w} \ddot{\gamma} \ddot{\varphi} \cos (2\gamma) \sin (2\gamma) \\
+ I_{y_w z_w} \ddot{\gamma} \ddot{\varphi} \cos \sin \psi + I_{y_w z_w} \ddot{\gamma} \ddot{\varphi} \sin ^2 \gamma \cos \psi - I_{y_w z_w} \ddot{\gamma} \ddot{\varphi} \cos ^2 \psi \\
+ \frac{1}{2} I_{y_w y_w} \ddot{\gamma} \ddot{\varphi} \sin (2\gamma) - \frac{1}{2} I_{z_w z_w} \ddot{\gamma} \ddot{\varphi} \cos \gamma \cos \psi - I_{y_w z_w} \ddot{\gamma} \ddot{\varphi} \cos \gamma \cos \psi \\
- I_{y_w z_w} \ddot{\gamma} \ddot{\varphi} \cos \gamma \sin \psi + I_{y_w y_w} \ddot{\gamma} \ddot{\varphi} + I_{z_w z_w} \ddot{\gamma} \ddot{\varphi} \\
- I_{y_w z_w} \ddot{\gamma} \ddot{\varphi} \sin \gamma \sin \psi + I_{y_w y_w} \ddot{\gamma} \ddot{\varphi} \cos ^2 \gamma \sin ^2 \psi \\
+ I_{y_w z_w} \ddot{\gamma} \ddot{\varphi} \cos \sin \psi + 2I_{y_w y_w} \ddot{\gamma} \ddot{\varphi} \cos \gamma \sin ^2 \psi + \frac{1}{2} I_{y_w y_w} \ddot{\gamma} \ddot{\varphi} \cos (2\psi) \\
- I_{y_w z_w} \ddot{\gamma} \ddot{\varphi} \sin \gamma \sin \psi + I_{y_w y_w} \ddot{\gamma} \ddot{\varphi} \sin ^2 \gamma + I_{y_w y_w} \ddot{\gamma} \ddot{\varphi} \cos ^2 \gamma \sin ^2 \psi \\
+ I_{y_w y_w} \ddot{\gamma} \ddot{\varphi} \sin \gamma - I_{y_w z_w} \ddot{\gamma} \ddot{\varphi} \sin \psi + I_{y_w y_w} \ddot{\gamma} \ddot{\varphi} \sin \gamma \\
- I_{y_w z_w} \ddot{\gamma} \ddot{\varphi} \cos \gamma \sin \psi + I_{y_w y_w} \ddot{\gamma} \ddot{\varphi}
\]  

(18)
where the mass moments of inertia have been introduced. Equation 18 is valid for any rigid planar wing (not necessarily of rectangular shape).

The equilibrium relation (see equation 9) can be scalarly multiplied by the angular velocity vector:

\[ M + M^{\text{aero}} - M^{\text{in}} = 0 \Rightarrow M \cdot \omega + M^{\text{aero}} \cdot \omega - M^{\text{in}} \cdot \omega = 0 \cdot \omega \] (19)

The mechanical power \( P^{\text{mech}} \) is represented by the product \( M \cdot \omega \). The other two contributions are \( P^{\text{aero}} = M^{\text{aero}} \cdot \omega \) (see equation 13) and \( P^{\text{in}} = M^{\text{in}} \cdot \omega \) (see equation 17 and the final expression reported in equation 18). Equation 19 is then rewritten as

\[ P^{\text{mech}} + P^{\text{aero}} - P^{\text{in}} = 0 \Rightarrow P^{\text{mech}} = -(P^{\text{aero}} - P^{\text{in}}) \] (20)

**Propulsive Efficiency**

The dynamic equilibrium of the forces is presented in equation 8. The summation of all the inertial forces \( F^{\text{in}} \) is given by the following relation:

\[ F^{\text{in}} = \int dF^{\text{in}} = \int \ddot{r} \cdot dm_w \] (21)

which can be calculated with some algebra considering the assumptions of a rigid wing. The explicit expression is omitted for brevity.

The resultant of all the aerodynamic forces, provided by the CFD software, is given by

\[ F^{\text{aero}} = \sum_{i=1}^{N_{\text{CFD}}} F^{\text{aero}}_{i} = F^{\text{aero}}_{xS} S_{xS} + F^{\text{aero}}_{yS} S_{yS} + F^{\text{aero}}_{zS} S_{zS} \] (22)

Once the aerodynamic and inertial forces are calculated, it is possible to find the reactions at \( B \) directly from the equilibrium relation (equation 8)

\[ R = F^{\text{in}} - F^{\text{aero}} \] (23)

Under the assumption of forward motion with constant speed, there cannot be an acceleration (at least in an average sense) in the direction of the flow \( (zS) \) on a complete flapping cycle. From a practical point of view, this means that the reaction \( R_{zS} \) in that direction has to be exactly zero. Consistency of the present model would require a selection of a mass distribution so that the condition is met during the flapping cycle. However, in his work the flapping wing is considered to be mounted on a wind tunnel testing set up. That is, the freestream velocity is blown on a flapping model. Thus, it is not required to have \( R_{zS} = 0 \).

The propulsive efficiency over a complete flapping cycle is calculated as the total power associated with the thrust (component of the aerodynamic forces in the \( zS \) direction) divided by the mechanical power required to maintain the wing’s flapping motion:

\[ \eta_{\text{prop}} = \frac{V_\infty \int_0^T \dot{F}^{\text{aero}}_{zS} (t) \, dt}{\int_0^T \dot{P}^{\text{mech}} (t) \, dt} \] (24)

**V. Aerodynamic Solver**

The computational approach employed for the aerodynamics solves the unsteady, three-dimensional Navier-Stokes equations using a well-validated and robust high-order solver\(^{17-19} \) (FDL3DI). A number of key features of the FDL3DI code make it most suitable for the present computational challenge. All spatial derivatives are computed through high-order compact or Pade-type\(^{20} \) difference methods. Schemes ranging from standard 2nd order to highly accurate 6th order methods are possible. To enforce numerical stability, which can be compromised by mesh stretching, boundary conditions and non-linear phenomena, a higher-order, low-pass filter is utilized. This discriminating up to 10th-order low-pass filter preserves the overall high-order accuracy of the spatial discretization while retaining stability for nonlinear applications. Careful attention to the treatment of the metric derivatives and the Geometric Conservation Law (GCL)\(^{21} \) ensures
higher-order accuracy on deforming and moving grids. The solver is embedded in a high-order overset-grid scheme which is utilized to provide flexibility for modeling complex geometries. It also serves as a domain decomposition mechanism for application of the high-order approach on massively-parallel, high-performance computing platforms.

**Implicit Large Eddy Simulation Methodology**

The ILES method to be used in the present computations was first proposed and investigated by Visbal et al.\(^22\) The underlying idea behind the approach is to capture with high accuracy the resolved part of the turbulent scales while providing for a smooth regularization procedure to dissipate energy at the represented but poorly resolved high wavenumbers of the mesh. In the present computational procedure the 6th-order compact difference scheme provides the high accuracy while the low-pass spatial filters provide the regularization of the unresolved scales. All this is accomplished with no additional sub-grid scale models as in traditional LES approaches. An attractive feature of this filtering ILES approach is that the governing equations and numerical procedure remain the same in all regions of the flow. In addition, the ILES method requires approximately half the computational resources of a standard dynamic Smagorinsky sub-grid scale LES model. This results in a scheme capable of capturing with high-order accuracy the resolved part of the turbulent scales in an extremely efficient and flexible manner.

**Aerodynamic Features of Flapping Flow**

Aerodynamic computations have been performed for the previously described rigid wing in isolation undergoing an anti-clockwise 8 cycle kinematics. The mesh system developed, Fig. 11, consists of 249 points in the streamwise direction, 289 points in the spanwise direction and 151 points above and below the wing. 61 points in the chordwise direction and 141 points in the spanwise direction are located on the wing with a maximum spacing of \(\Delta y = 0.03\) and \(\Delta z = 0.028\). A minimum spacing, \(\Delta z = 0.00025\), is specified at the wing surface. The streamwise spacing downstream of the wing was restricted to a maximum of \(\Delta z = 0.04\) for 1.85 chord lengths before the mesh was stretched to the downstream boundary. This grid was subdivided into 200 overlapping meshes for parallel processing.

The aerodynamic parameters specified for the flapping wing in forward flight consist of a Reynolds number based on the freestream velocity of \(Re = 1250\). The advance ratio specified is \(J = V_{\infty}/V_F = 0.5\) where \(V_F = 2.0\Phi R/T\), \(R\) is the distance from the flap axis to the wing tip and \(T\) is the period of the flapping motion. \(V_F\) indicates the so-called flapping velocity. A Reynolds number based on the flapping velocity is \(Re = 2500\). The frequency of the flapping motion correspond to a Strouhal number, \(St = 0.2387\) which gives a period for the motion of 4.19 characteristic times. A time step, \(\Delta t = 0.00025\) is employed based on the temporal resolution required for the fine scale flow features observed.

Fig. 12 presents the variation of the thrust coefficient, \(C_T\) and the lift coefficient, \(C_L\) over one period of the flapping motion where time \(t^* = 0\) corresponds to the top of the upstroke and \(t^* = 0.5\) corresponds to
the bottom of the downstroke. Distinct peaks in the two curves are noted at the midpoints of the downstroke and upstroke. An additional peak in the $C_L$ curve is seen near the bottom of the downstroke. The present flapping motion produces thrust over nearly the full flapping cycle whereas positive lift is produced during the downstroke and negative lift is produced during the upstroke. The mean lift generated during the flap cycle is $C_L = 1.69$ and the mean thrust is $C_T = 1.91$. The labeled circles in Fig. 12 correspond to temporal locations where flow visualizations will be presented subsequently.

The flow structure during one flapping cycle is shown in Figs. 13 (downstroke) and 14 (upstroke). Fig. 13a) is at the top of the upstroke. On the upper surface of the wing a small shallow separated vortical flow has formed outboard towards the tip of the wing. A very small tip vortex has also started to form. On the underside of the wing the remnants of the vortical structures previously created during the upstroke and the shed trailing vortex are seen. At this point in the cycle only very low levels of force are produced on the wing. As the wing progresses through the downstroke the remnants of the previous vorticity on the underside of the wing and the shed trailing edge vortex continue to be convected downstream and out of the system. On the upper surface a strong leading edge vortex and tip vortex develop as the downstroke commences. During this initial portion of the downstroke, Figs. 13a)-c), these vortices grow in strength and extent but remain pinned at the wing corners. Extensive regions of strong lift and thrust force are produced due to the low pressures that develop underneath these vortices on the upper surface. This is the cause of the first peak in the lift and thrust seen in Fig. 12.

As the downstroke continues several interesting new features emerge, Figs. 13c)-e). A trailing edge and wing root vortex are now clearly seen to have developed. Outboard on the wing the leading edge vortex breaks down and small scale vortical structures are observed in the shear layer that rolls up to form the leading edge vortex. These shear layer instabilities result from the eruption of vorticity from the wing surface due to the interaction of the leading edge vortex with the surface boundary layer and have been observed previously in delta-wing flows$^{23,24}$ and for revolving wings.$^{25}$ Proceeding towards the bottom of the downstroke the leading edge, wing tip and wing root vortices unpin from the two outboard corners and the trailing edge corner at the root. The vorticity is reoriented forming a ring-like structure that is pinned at the leading edge corner at the wing root. This detached vortex structure moves away from the wing surface and convects downstream with a corresponding reduction in both the lift and thrust forces. By the time the wing reaches the bottom of the downstroke the outboard portion of the vortex has convected further downstream and the inboard sweep of the upstream portion is terminated when the trailing edge is reached with the vortex being rapidly turned away from the wing surface, Fig. 14f). A shallow separation region over the outboard portion of the wing and a wing tip vortex also reform at this point. The reformed vortical flow from the leading edge and a region of increased force where the original leading edge vortex departs from the wing surface at the trailing edge results in a brief enhancement of the lift forces, Fig. 14f. This gives rise to the secondary peak in the lift force around $t^\star = 0.5$ noted in Fig. 12.
Figure 13. Flow structure and force distribution during downstroke portion of flapping cycle. Column 1 shows the flow on the upper surface, column 2 shows the flow on the lower surface and column 3 shows the distribution of the lift and thrust force at each time step. Negative values of force correspond to lift and thrust in this figure. a)-e) correspond to the points a)-e) in Fig. 12.
Figure 14. Flow structure and force distribution during upstroke portion of flapping cycle. Column 1 shows the flow on the upper surface, column 2 shows the flow on the lower surface and column 3 shows the distribution of the lift and thrust force at each time step. Negative values of force correspond to lift and thrust in this figure. f)-j) correspond to the points f)-j) in Fig. 12.
As the wing rotates and commences the upstroke portion of the motion the separated flow over the upper surface weakens and the remaining vorticity is shed and convected downstream, Figs. 14 g)-h). On the underside of the wing a process similar to that noticed on the upper surface for the initial portion of the upstroke is observed with the formation of a leading edge and tip vortex, which are initially pinned at the corners. In the wake of the wing a series of small scale vortices are shed, Fig. 14h)-i). As the upstroke progresses the tip vortex unpins from the trailing edge corner and connects with one of the shed vortices. The leading edge vortex breaks down as the upstroke continues, Fig. 14i)-j), and also unpins from the leading edge corner and reconnects with the tip vortex resulting in a complex separated flow region covering the outer portion of the wing. As the wing reduces in angle of attack at the top of the upstroke the leading edge separation weakens and ultimately the vorticity is shed and convected downstream.

This description of the unsteady vortex dynamics generated by this anti-clockwise 8 cycle is consistent with the features described by Viswanath and Tafti\textsuperscript{26} for a similar kinematics albeit for a higher Reynolds number. A number of the vortical features and dynamics noted by Garmann et al\textsuperscript{25} for a revolving wing have also been observed for the present case.

VI. Rigid Wing: Effects of the Inhomogeneity of Density Properties on the Inertial Forces and Power Expenditure

Dimensionless Quantities

The inertial forces, moment of inertial forces and power associated with the inertial forces can be written in dimensionless form by using the flapping period, $T$, the mass of the wing, $m_w$, and the wing chord, $C$. The generic dimensionless quantities is indicated with the symbol ($\tilde{\cdot}$).

The inertial forces, moment of inertial forces, mass moments of inertia and powers are written in terms of the corresponding dimensionless ones as follows:

\[ F_{\text{in}} = \frac{m_w C}{T^2} \tilde{F}_{\text{in}} \quad M_{\text{in}} = \frac{m_w C^2}{T^2} \tilde{M}_{\text{in}} \quad I_{\bullet\bullet} = m_w C^2 \tilde{I}_{\bullet\bullet} \]
\[ P_{\text{in}} = \frac{m_w C^2}{T^3} \tilde{P}_{\text{in}} \quad P_{\text{aero}} = \frac{1}{2} \rho_\infty V_F^2 C S \tilde{P}_{\text{aero}} \] (25)

where $S = \frac{7}{2} C^2$ is the wing area and $I_{\bullet\bullet}$ is the generic mass moment of inertia (for example it can be $I_{xw}$, $x_w$ etc.).

Homogeneous Case: Effect of the Pitch Axis Position

The kinematics earlier described (see Figs. 2-8) is adopted in this work to investigate the properties of the flapping motion. It is of interest to see what the effects are when the equations are maintained the same but the position of the pitch (or feathering) axis $y_w$ is modified. That is, the position of $y_w$ (in addition to the case shown in Fig. 2 where $y_w$ is located at the first quarter chord) is moved to the leading edge, mid-chord, third-quarter chord, and trailing edge respectively.

Fig. 15 presents this study. An interesting observation is that in the case of pitch axis located at the mid-chord position the maxima for the dimensionless inertial forces present the same value. The same happens for the dimensionless moments of inertial forces.

Inhomogeneous Mass Distribution

The total mass $m_w$ is kept constant. However the density is distributed with different patterns. First, it is observed that for the homogeneous case [$\rho(y_w, z_w) = \rho_w = \text{const}$]

\[ \begin{align*}
\tilde{I}_{y_w y_w} &= 7/48 \\
\tilde{I}_{z_w z_w} &= 73/12 \\
\tilde{I}_{y_w z_w} &= 9/16
\end{align*} \quad \text{and} \quad \begin{align*}
\tilde{y}_{wG} &= 9/4 \\
\tilde{z}_{wG} &= 1/4
\end{align*} \]
Figure 15. Rigid homogeneous wing: effects of different kinematics on the power expenditure.

Figure 16. Span-wise discrete redistribution of mass.

Span-wise Discrete Redistribution of Mass

The wing has been divided in \( n \) homogenous sections, having the same size, see Fig. 16. Each \( i \)-th section of the wing has a density

\[
\rho_i = i \Delta \rho
\]

where \( i \) increases with the distance from the wing tip. By imposing that the total mass has to be constant,

\[
\sum_{i=1}^{n} m_i = m_w
\]

it follows that

\[
\Delta \rho = 2 (n + 1)^{-1} \rho_w
\]

where \( \rho_w \) is the density of the homogeneous material.
The mass moments of inertia can be shown to be:

\[
\begin{align*}
\bar{I}_{z_w z_w} &= \frac{1}{12} \left( \frac{7}{2n} \right)^2 + \sum_{i=1}^{n} \frac{2i}{n(n+1)} \bar{y}_{wG}^2 i \\
\bar{I}_{y_w z_w} &= \sum_{i=1}^{n} \frac{i}{2n(n+1)} \bar{y}_{wG} i \\
\bar{I}_{y_w y_w} &= \frac{7}{48}
\end{align*}
\]

It is possible to deduce from Fig. 17 that a larger density near the wing root reduces the inertial forces and associated power for a given total mass \(m_w\). The maxima of the dimension-less power associated with the inertial contributions are reduced up to 50%.

![Figure 17. Span-wise discrete redistribution of mass and its effects on the inertial forces and power](image)

**Chord-wise Discrete Redistribution of Mass**

Following the same procedure previously described, the behavior of the physical quantities when the mass is redistributed and accumulated toward the wing trailing edge is investigated. That is, the density decreases for positions near the trailing edge. For this case the mass moments of inertia can be shown to be

\[
\begin{align*}
\bar{I}_{z_w z_w} &= \frac{73}{12} \\
\bar{I}_{y_w z_w} &= \sum_{i=1}^{n} \frac{9i}{4n(n+1)} \bar{z}_{wG} i \\
\bar{I}_{y_w y_w} &= \frac{1}{12} \left( \frac{1}{n} \right)^2 + \sum_{i=1}^{n} \frac{2i}{n(n+1)} \bar{z}_{wG}^2 i
\end{align*}
\]
From Fig. 18 it is possible to see that inertial forces and associated power are practically coincident with values relative to the homogeneous case. In other words, an inhomogeneous material with density distribution selected to have higher density at the leading edge, is not the best choice if the goal is to reduce the power associated with the inertia of the system.

Continuum Radial Distribution of Mass From a Corner of the Wing Root

In this case the mass is concentrated near one corner of the wing root. For simulating this configuration, a radial distribution of mass centered at the intersection between the root and the leading edge of the wing (coordinates \( \tilde{y}_w = 1/2 \) and \( \tilde{z}_w = -1/4 \)) is used:

\[
\rho \left( \tilde{y}_w, \tilde{z}_w \right) = \rho_w A e^{-B (\tilde{y}_w - 1/2)^2 + (\tilde{z}_w + 1/4)^2} = \rho_w A g \left( B, \tilde{y}_w, \tilde{z}_w \right)
\]  

(26)

where \( B \) is a coefficient that regulates the distribution of mass: for small values of \( B \) the wing is close to be homogeneous, while for high values of \( B \) the mass is mostly concentrated in the proximity of the center of the distribution. The value of \( A \) is selected such that the overall mass is equal to \( m_w \) thus it is possible to write:

\[
m_w = s_w \int_S \rho \left( y_w, z_w \right) d y_w d z_w = s_w C^2 \int_S \rho \left( \tilde{y}_w, \tilde{z}_w \right) d\tilde{y}_w d\tilde{z}_w = A \int_S g \left( B, \tilde{y}_w, \tilde{z}_w \right) d\tilde{y}_w d\tilde{z}_w
\]  

(27)

where

\[
\kappa \left( B \right) = \int_S g \left( B, \tilde{y}_w, \tilde{z}_w \right) d\tilde{y}_w d\tilde{z}_w,
\]

Thus, it follows from equation 27 that

\[
A = \frac{2}{\kappa \left( B \right)}.
\]
The center of mass and mass moments of inertia can be easily calculated (details omitted for brevity). When the center of radial distribution of mass is selected to be the intersection of the root chord with the trailing edge, the results are presented in Fig. 20. The maxima of the dimensionless power of the inertial forces is drastically reduced with respect to the inhomogeneous cases discussed in Figs. 17 and 18.

Homogeneous Case: Mechanical Power and Propulsive Efficiency

Fig. 21 shows the dimensionless distributions of the mechanical power associated with the inertial forces and aerodynamic actions. It can be observed that the relative importance of the contributions depends on the ratio of the freestream density and the density of the wing. Other parameters, such as the flapping velocity, period of oscillation, and wing chord also influence the power of the aerodynamic forces (see the boxed equation reported in Fig. 21). The propulsive efficiency, calculated as shown in equation 24 is 0.18. This value agrees well with the result (0.1959) presented in Reference [26].

VII. Elastic Wing: Effects of the Inhomogeneity of Density and Stiffness Properties

For the previous discussions the wing was considered rigid. This was perfectly consistent with the aerodynamic forces which were computed via a CFD analysis of a rigid wing moving with the prescribed kinematics. The assumption of a rigid structure is now removed. This is a preliminary investigation and at this stage the aerodynamic forces, independently calculated for the rigid case are transferred to the structural mesh of the elastic wing. The aerodynamic loads computed with this method are clearly not affected by the deformation of the structure. That is, the intrinsic nature of the dependence of the aerodynamic forces on the deformation (the local deformation changes the angle of attack with respect to the rigid case) is not included in this first study. The true aeroelastic capability will be added in subsequent works.

It is of interest to assess the effects of inhomogeneity of both density and stiffness distributions for the elastic case. In this work, several investigations are carried out with prescribed kinematics and no aerodynamics. Other analyses involve the aerodynamic loads as well. To maintain a meaningful and consistent...
Figure 20. Continuum radial distribution of mass from a corner of the wing root

The distribution of density is:

$$\rho (\vec{y}_w, \vec{z}_w) = A r_w e^{-B[(\vec{y}_w - 1/2)^2 + (\vec{z}_w - y)^2]}$$

where $\rho_w$ is the density of the homogeneous material.

Figure 21. Rigid homogeneous wing: comparison between the inertial and aerodynamic power.

The approach, the reference material (subscript $\text{ref}$ is adopted to indicate the corresponding quantities) is selected to be steel, so that the deformations are not excessive and the overall kinematics of the wing will not differ substantially from the rigid case. The adoption of steel as reference material does not imply that a flapping wing should be built with a rectangular lamina made of steel. The present study intends only to provide physical insights of a wind-tunnel-like model of a flapping unmanned aerial system.

Starting from the reference material, the density is gradually varied following several patterns (see also
the previous discussion valid for the rigid case) as indicated in Fig. 22. Other investigations will modify

\[ \rho_1 = 0.270 \cdot \rho_{ref} \]
\[ \rho_2 = 0.541 \cdot \rho_{ref} \]
\[ \rho_3 = 0.811 \cdot \rho_{ref} \]
\[ \rho_4 = 1.082 \cdot \rho_{ref} \]
\[ \rho_5 = 1.352 \cdot \rho_{ref} \]
\[ \rho_6 = 1.622 \cdot \rho_{ref} \]

Figure 22. Inhomogeneous case: different density distribution

the elastic modulus distribution as shown in Fig. 23. In some cases the density will be kept equal to the reference value. In others, the ratio \( E/\rho \) will be maintained constant. The numerical solutions are carried

\[ E_{ref} = 210 \, GPa \]
\[ E_1 = 1/6 \cdot E_{ref} \]
\[ E_2 = 1/5 \cdot E_{ref} \]
\[ E_3 = 1/4 \cdot E_{ref} \]
\[ E_4 = 1/3 \cdot E_{ref} \]
\[ E_5 = 1/2 \cdot E_{ref} \]
\[ E_6 = E_{ref} \]

Figure 23. Inhomogeneous case: different stiffness distribution

out with the software NX NASTRAN\textsuperscript{27} The solution is the \textit{Advanced Nonlinear Transient Analysis} (SOL 601.129) described at page 328 of Reference [27] and in Reference [28]. The structural mesh adopts 1008 CTRIA3 elements for a total of 559 nodes corresponding to 3354 degrees of freedom. The boundary and initial conditions are imposed as follows. The positions and velocities of all nodes are imposed at the initial instant to be consistent with the values of the rigid case (the same laws are used). At subsequent instants, the kinematics of the rigid case is imposed only at the nodes on the root of the wing.
Effect of Density Distribution

The density is a function of the spatial coordinates $y_w$ and $z_w$ as described in Fig. 22. The total mass is kept constant and equal to the reference value. The reference elastic modulus is also selected.

Fig. 24 shows the trajectory of the tip of the wing (the point has coordinates $y_w = 4C$ and $z_w = -1/4C$ in the wing coordinate system) for the first cycle. The first observation is that the anticlockwise 8 is not coincident with the one obtained with the rigid analysis (see previous discussions). Now the loop is not perfectly closed. If more cycles are studied, similar patterns are observed. The subsequent cycles are all contained in a small neighborhood of the first one.

Changing the density chord-wise provides a similar pattern to the homogeneous case. In other words, the inertial forces are not changed substantially. The results earlier demonstrated for the rigid case are then confirmed for the elastic wing. As expected from the rigid case, having the mass concentrated near the root reduces inertial forces and the trajectory of the wing tip is closer to the rigid case. If the simulation is conducted by changing the density as done in Fig. 24, but the ratio $E/\rho$ is maintained constant and equal to the reference value, then the resulting trajectory for the tip of the wing is plotted in Fig. 25. Based on these results, the analysis of Fig. 24 also applies for this case.

Effects of the Stiffness Distribution

Fig. 23 shows different ways in which the magnitude of the elastic modulus is distributed over the wing. The density is not modified. Fig. 26 presents the trajectory of the tip of the wing during the first cycle. The reduction of the stiffness in some areas of the wing as a consequence of the inhomogeneity of the stiffness properties, dramatically affects the wing response and the resulting motion is far from the rigid case. The aerodynamic forces are not considered in this case.

Effect of Inhomogeneity of the Density Properties in the Presence of Aerodynamic Forces

The aerodynamic forces are transferred to the structural mesh with an in-house subroutine based on an energetic method. The virtual work done by the aerodynamic forces acting on a generic triangular finite
Figure 25. Trajectories of the tip node during the first cycle for different distributions of densities keeping $E \rho = \text{const}$

Figure 26. Trajectories of the tip node during the first cycle for different distributions of Young modulus
element is imposed equal to the virtual work done by the equivalent nodal forces. Fig. 27 shows the nodal forces equivalent to the loads derived via CFD analysis. The forces shown in the Figure are evaluated at the beginning of the stroke ($t^* = 0$). The aerodynamic forces applied to the structure (Fig. 27) during the entire cycle have a very little effect on the deformation of the wing as shown in Fig. 28 and Fig. 29.

Figure 27. Aerodynamic forces applied at the structural grid points on the upper (left) and lower (right) faces of the wing in the beginning of the stroke

Employing the distributions of density depicted in Fig. 22, Fig. 28 demonstrates that the effect of the aerodynamic forces is small when compared to the case in which only the inertial forces are present. This is not a general finding: as discussed in References [13, 14] the relative importance of aerodynamic and inertia loads in flapping wings is a configuration-dependent property. Fig. 29 shows a similar study for the case in which the ratio $E/\rho$ is maintained constant.

VIII. Elastic Wing: Effects of the Material Anisotropy

In an aeroelastic simulation the bending-torsion coupling is one of the determinant factors that needs to be considered. Changing how the angle of attack is modified during the deformation is a powerful design possibility which can enhance the performance of the unmanned aerial system. In the present study, the aerodynamic forces are calculated off line with a CFD solver. Thus, the effects of anisotropy cannot really be appreciated to its full extent. However, it is still interesting to observe the nonlinear structural response and how it changes when the material is not isotropic (earlier, in this work, the material was at most inhomogeneous but still isotropic).

To investigate the effects of anisotropy, the material properties of the composite laminate are varied (the thickness $s_w$ is maintained 0.2 mm). For simplicity, only a single lamina is considered (no practical implications on the building of composite wings are implied). An angle of zero degrees means that the fibers are oriented along the wingspan. An angle of 90 degrees means that the fibers are parallel to the wing chord.

Fig. 30 shows the displacements on the stroke-plane coordinate system $x_S$, $y_S$, $z_S$ for the different angles. Stiffness in the wingspan direction is crucial to maintain a tip trajectory similar to the rigid case. In fact, when the fibers are oriented with a relatively large angle ($60^\circ$ or $90^\circ$) the trajectory is dramatically affected.
Figure 28. Comparison of loaded and unloaded trajectories for different densities distributions.
Figure 29. Comparison of loaded and unloaded trajectories for different densities distributions keeping $\frac{E}{\rho}$ costant
Figure 30. Anisotropic case: deformation on a cycle
IX. Conclusions

This work presented high-fidelity structural and aerodynamic investigations of the anticlockwise eight flapping cycle, which simulates a forward flapping motion at constant speed. First the kinematic laws were derived and interpreted. Then, the aerodynamic CFD analysis was introduced and a complete stroke was presented, with particular emphasis on the complex vortex structure that is formed during the downstroke and upstroke phases. Formation of the leading edge, trailing edge, tip, and root vortices were observed. Their evolution during the flapping of the wing was discussed. Thrust is produced for both the downstroke and upstroke motions, whereas a positive lift is obtained only during the downstroke. The magnitude of the aerodynamic lifting forces are of the order of magnitude of $mN$, showing the challenging design requirements for a flapping unmanned aerial system: it should be able to sustain its own weight and carry a payload. Thus, the choice of wing structural material and its properties is a crucial steps towards efficiency.

In regards to the energetic efficiency required to maintain forward flight, the mechanical, inertial, and aerodynamic powers were analytically derived for the case of rigid wing. Several investigations demonstrated that the inhomogeneity of the density distribution could be an effective resource to reduce the required mechanical power. In particular, a concentration of the mass towards the root of the wing is highly beneficial, whereas a higher density near the leading edge provides no practical benefit.

The effect of the pitch axis location on the power expenditure was also investigated for homogeneous wings. In some cases it is possible to achieve a reduction of the power associated with the inertial forces. This aspect should be fully analyzed from an aeroelastic point of view and is the subject of future investigations.

Part of the mechanical power can be imagined to be spent to advance the aerial system with an assigned (constant in this work) speed. How this process is effective for the case investigated in this work, was investigated with the introduction of the propulsive efficiency. It was demonstrated that for the rectangular geometry analyzed in this effort that efficiency was about 18%.

This research also showed the relative importance between the aerodynamic and inertial contributions to the mechanical power. However, a general indication cannot be given since, as known from the literature, the relative importance is a configuration dependent problem.

In an effort to understand the effects of deformability of the wing, a reference case of a wind tunnel-like wing model made of steel has been dynamically analyzed with a geometrically nonlinear CSD simulation. The inhomogeneity of the mass distribution played a role similar to the rigid case. However, changing the stiffness, especially in the wing span direction (anisotropy of the material) seems to have major effects on the response of the wing. This analysis was carried out by taking aerodynamic forces evaluated from off line CFD computations. A true aeroelastic coupling may dramatically modify this picture. Future studies will address this effect of aeroelastic coupling. Material anisotropy/inhomogeneity will be exploited to increase the propulsive efficiency and reduce the power expenditure required to maintain the motion and sustain the system weight.

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