The unsteady airflow over a Pitching wing using a 3-D Panel Method

Mark Ashish K

March 30, 2012

The unsteady airflow over a Pitching wing using a 3-D Unsteady Panel Method

A thesis submitted towards partial fulfilment of

BS-MS Dual Degree Programme

by

MARK ASHISH K

under the guidance of

DR. L. VENKATAKRISHNAN

HEAD, EXPERIMENATAL AERODYNAMICS DIVISION

CSIR-NATIONAL AEROSPACE LABORATORIES

INDIAN INSTITUTE OF SCIENCE EDUCATION AND RESEARCH PUNE

30th March, 2012

Certificate

This is to certify that this thesis entitled 'The unsteady airflow over a Pitching wing using a 3-D unsteady Panel Method' submitted towards the partial fulfilment of the BS-MS dual degree programme at the Indian Institute of Science Education and Research Pune represents original research carried out by Mark Ashish K at National Aerospace Lab-Bangalore, under the supervision of Dr.L.Venkatakrishnan from 2nd August,2011 to 30th March,2012.

Supervisor

Dr.L.Venkatakrishnan

Head, EAD

Dr.C.Divakar

Joint Head,KTMD

Acknowledgements

The candidate would especially like to thank Mr. Shyam Chetty (Director, National Aerospace Laboratories), Dr. C. Divakar (Joint Head, KTMD), Dr. L. Venkatakrishnan (Head, Experimental Aerodynamic Division), Dr. V. Y. Mudkavi (Head, CTFD division) and Dr. K. R. Srilatha (Scientist, CTFD division) for their constant guidance and support throughout the project. I extend my gratitude to the other project students in the CTFD division for their help. Lastly, I thank Dr. Sudarshan Ananth for being the local coordinator at IISER.

Abstract

An unsteady 3-D Panel method has been employed to study the aerodynamics of a finite span wing. The wing is unhinged and is pitching about the quater chord line as it is moving in the forward direction. The panel method is used to extract the loads on the wing and also to capture the wakes generated by the wing as it executes pitching. The advantages of the panel method visa vie other methods are also summerized. The forces on the wing are plotted with respect to time, from which the propulsive efficiency is calculated. The free wake is simulated by a lattice of shed and trailing vortices. The wakes are plotted in *tecplot* and compared with plots given in literature. The wake shapes generated by the code for the wing in time dependent motion agrees well with the results obtained in literature.

Contents

1 Introduction

Studying the unsteady aerodynamics of a wing in time dependent motion is essential to understand the aerodynamic performance. For wings in time-dependant motion, unsteadiness in the flow is caused by the shed vorticity and temporal change of the wake geometry. The cyclic variation of the free stream velocities relative to the wing due to the cyclic modulation of the blade pitch introduces an unsteadiness in the flow.

Some of the methods used to study unsteady aerodynamics are : momentum method, blade-element method, hybrid momentum (or vortex) method, lifting-line method, 2-D thin airfoil method and lifting-surfaces(or vortex lattice) method. Unfortunately, existing methods of the unsteady aerodynamics of the flapping wing are constrained. Describing each of the methods in detail will encompass the thesis, but in broad strokes current theory and research on flapping flight(in general)is characterized by rapid reversals in stroke direction and in wing rotation which result in gross movements of lifting surfaces, and produces the necessary aerodynamic forces for flight in a highly efficient manner. The need of the hour is a method which accommodates both trailing vortex effects and wing force resolution in a detailed manner.[1] A general problem, therefore, with existing methods is that while some can detail vortex effects and others can accommodate wing force resolution, not one of the methods reviewed above is capable of detailing both. For example, the hybrid method has no detailed wake or detailed force resolution, the lifting-line method has no detailed wake resolution, is valid only for small displacements and has no detailed force resolution, and the prevailing lifting-surface method has no detailed free-wake analysis. Given the need to model the relavent aerodynamic forces on pitching/flapping wings, and the disadvantages of prevailing aerodynamic methods, the present study advances a type of liftingsurface method known as an unsteady aerodynamic panel method.

2 Theory

The theory underlying my work is as follows:

Fundamental principles of aerodynamics : definitions, variables, dimensional analysis

2.1 Fundamentals in Aerodynamics

Aerodynamics describes the dynamics of gases, especially atmospheric interactions with moving objects. The quantities which are most frequently used in aerodynamics are as follows

- Pressure
- Density
- Temperature
- Velocity
- Shear stresses

Pressure is the normal force per unit area exerted on a surface due to the time rate change of the momentum of the gas molecules impacting on (or cross section) that surface.

$$p = \frac{dF}{dA} \tag{1}$$

Density is the mass per unit volume. It is a point property that can vary from point to point.
ν = element volume around a point in a fluid
dm = mass of the fluid element

$$\rho = \frac{dm}{d\nu} \tag{2}$$

Temperature is the average kinetic energy of the molecules of the Fluid.It plays an important role in high speed aerodynamics.

Flow Velocity is the velocity of a flowing gas at any fixed point in space ai the infinitesimally small fluid element as it sweeps through the point. *Shear Stress* is defined as follows.

$$\tau = \frac{dF}{dA} \tag{3}$$

F is the tangential force on a particular streamline

2.2 Aerodynamic Forces

The aerodynamic forces are due to 2 main basic sources:

1. Pressure distribution over the body surface(p)

2. Shear stress distribution over the body surfac(τ)

Pressure acts normal to the surface whereas the shear stress acts tangential to the surface. The net effect of the p and tau distributions integrated over the complete body surface is a resultant aerodynamic force R and moment M on the body. In turn, the resultant R can be split into components, Lift(L) and Drag(D). If V is the relative wind defined as the flow velocity far away from the body(free stream). Hence V_∞ is called the *free stream velocity*.
Lift is the component of R perpendicular to V_∞.
Drag is the component of R parallel to V_∞.
Moment is the torque due to the aerodynamic forces, which is generally taken about a point on the airfoil.

2.3 Dimensional Analysis

This section is mainly intended to enunciate the various dimensionless quantities in aerodynamics. The focus is on how we go about deriving them and also sheds light on the importance of each in aerodynamics.

The aerodynamic forces and moments on a body have been described above, we now need to determine the physical quantities that determine the variation of these forces and moments. The same is done through *dimensional analysis*.
Consider a body of a given shape at a given angle of attack. The resultant aerodynamic force is R. On a physical intuitive basis, we expect R to depend on:

1. Freestream Velocity V_∞.

2. Freestream density ρ.

3. Viscosity of the fluid μ_∞.

4. The size of the body, represented by some chosen reference lenght. The convenient lenght used here is the chord lenght c.

5. The compressibility of the fluid. The compressibility of the fluid is related to the speed of sound a_∞.

We can (without any a priori knowledge) write the variation of R as the following

$$R = f(\rho, V_\infty, c, \mu_\infty, a_\infty) \tag{4}$$

Measuring the variation of R due to $\rho, V_\infty, c, \mu_\infty, a_\infty$ will take a very long time. For this reason the method of dimensional analysis is employed, the method defines a set of dimensionless parameters that govern the aerodynamic forces and moments. This will considerably reduce the number of independent variables.

2.3.1 Buckingham pi Theorem

Let K equal the munber of fundamental dimensions required to describe the physical variables (K=3 as in mechanics all physical variables can be expressed in terms of mass, lenght and time), Let $P_1, P_2, P_3, ..., P_N$ represent the N physical variables in the physical relation.

$$f_1(P_1, P_2, P_3, ..., P_N) = 0 \tag{5}$$

the above physical relation may be reexpressed as a relation for N-K dimensionless products called Π products,

$$f_2(\pi_1, \pi_2......, \pi_N) = 0 \tag{6}$$

As K = 3 and N=6, we get N-K = 3 dimensionless numbers. These numbers describe all that we have to know regarding the aerodynamic forces. Thus, we have reduced the number of independent variables from 5 to 2. The 3 dimensionless numbers derived from this theorem are

1. Reynolds number$(Re = \rho * V_\infty * c/\mu_\infty)$

2. Mach number$(M = V_\infty/a_\infty)$

3. Force coefficient $(C_R = R/0.5 * \rho * V_\infty^2 * S)$

Now, if we wish to run a series of wind-tunnel tests for a given body at a given angle of attack, we need only to vary the Reynolds and Mach numbers in order to obtain data for the direct formulation of R through the equation

$$C_R = f(Re, M). \tag{7}$$

This analysis was for a given shape at a given angle of attack $\alpha. If \alpha$ is allowed to vary, then C$_L$, C$_D$, C$_M$ will in general depend on the value of α, hence

C$_L$ = f(Re, M_∞, α) where $(C_L = L/0.5 * \rho * V_\infty^2 * S)$

C$_D$ = f(Re, M_∞, α) where $(C_D = D/0.5 * \rho * V_\infty^2 * S)$

C$_M$ = f(Re, M_∞, α) where $(C_M = M/0.5 * \rho * V_\infty^2 * S * l)$

Much of theoretical and experimental aerodynamics is focussed on obtaining explicit expressions for the above quantities The above are strictly NOT equations. They define the quantities such as C_L, C_D etc. They depend primarily on angle of attack once the shape is fixed. In any one experiment, it would be possible to get complete dependence of these quantities with angle of attack α by merely varying speed, for instance. Once the dependence curve is obtained, it represents behaviour for a large number of combinations of speed, density etc.(for specific body shapes).

2.4 Governing equations for irrotational, incompressible flow

Continuity Equation:

The continuity equation is given by:

$$\frac{\partial \rho}{\partial t} + \nabla(\rho \cdot V) = 0 \tag{8}$$

If ρ is constant then the equation becomes,

$$\nabla \cdot V = 0 \tag{9}$$

From continuity equation, we get,

$$\nabla \cdot V = 0 \tag{10}$$

For an irrotational flow, $\nabla \times V = 0$. Hence, a velocity potential ϕ can be defined such that

$$V = \nabla\phi \tag{11}$$

Both the above equations combine to give

$$\nabla^2\phi = 0 \tag{12}$$

The above equation is called the Laplace equation. By the above equation, we conclude that

1.Any irrotational, incompressible flow has a velocity potential that satisfies the Laplace's equation.

2.Conversely, any solution of Laplace's equation represents the velocity potential for an irrotational and incompressible flow.

2.5 Superposition of elementary flows

In this section, we present the first of a series of elementary incompressible flows that later will be superimposed to synthesize more complex incompressible flows. Some elementary flows used in this presentation are given below,

1. Uniform Flow

If ϕ is the velocity potential, then ϕ for a uniform flow is given by:

$$\phi = V_\infty x + const. \tag{13}$$

2. Source Flow

Source flow is a 2-D incompressible flow where all the streamlines are straight lines emanating from a central point. Source flow is a physically possible incompressible flow, that is ∇. V = 0 at every point except origin and is irrotational at every point.

$$\phi = \frac{\Lambda}{2\pi} ln(r) \tag{14}$$

3. Doublet Flow

A source-sink pair that leads to a singularity is called a doublet flow.

$$\phi = \frac{k}{2\pi}\frac{cos\theta}{r} \tag{15}$$

4. Vortex Flow

Consider a flow where all the streamlines are concentric circles about a given point. Moreover, let the velocity along any given circular streamline be constant, but let it vary from one streamline to another inversely with distance from the common center. Such a flow is called a vortex flow. A vortex flow is a physically possible flow such that ∇. V = 0 at all points and $\nabla \times V = 0$ at all points except origin.

$$\phi = -\frac{\Gamma}{2\pi}ln(r). \tag{16}$$

2.6 Circulation

Circulation is the line integral of velocity around a closed curve in the flow. It is denoted by Γ. If V is the fluid velocity on a small element of a defined curve, and dl is a vector representing the differential length of that small element, the contribution of that differential length to circulation is :

$$\Gamma \equiv \oint V \cdot dl \tag{17}$$

2.7 Kutta-Joukowski Theorem

The lift force acting per unit span on a body in a two dimensional invisid flow field can be expressed as the product of circulation(Γ) about the body, the fluid density(ρ) and the speed of the body relative to the free stream(V).

$$lift = \rho V \Gamma \tag{18}$$

2.8 Biot-Savart Law

If the strength of the vortex filament is defined as Γ(circulation). Consider a directed segment of the filament dl. The radius vector from dl to an arbitrary point P in space is r. The segment dl induces a velocity at P equal to

$$dV = \frac{\Gamma}{4\pi}\frac{dl \times r}{|r|^3} \tag{19}$$

2.9 Helmholtz's vortex theorem

In an inviscid, incompressible flow:

1. The strength of a vortex filament is constant along its lenght.

2. A vortex filament cannot end in a fluid; it must extend to the boundaries of the fluid (which can be $\pm\infty$)or form a closed path.[4]

2.10 Propulsive efficiency

For a wing which is flapping, the propulsive efficiency is given by:

$$\eta = \bar{C_T}/(\dot{z}.\bar{C_L} + \bar{C_M}.\bar{\dot{\alpha}}) \tag{20}$$

Unsteady aerodynamics: Thrust generation, lift generation, propulsive efficiency[6,7,8,9]

3 Description of the Method: Unsteady Panel Method

Introduction

The unsteady aerodynamic panel method or a classical boundary element method, relies on developing a distribution of source and doublet singularities on wing and body surfaces, and doublet singularities to represent the lifting surface. To date, the panel method has been used, almost exclusively, to analyze the aerodynamic forces on aircraft(Ashley and Landahl, 1985;Ashby et al. 1988;Katz and Plotkin, 1991). The unsteady panel method is based on potential theory which assumes non-viscous flow(see Katz and Plotkin, 1991). Engineers have established that the advantages of the panel method are that it accommodates the detailing of the trailing wake, includes dynamic effects and includes those effects in a distributed manner (in resonable time). Moreover, the panel method is also capable of accommodating flexibility and interference effects. Such advantages render the panel method more useful than other prevailing methods when analyzing the aerodynamic forces on flapping wing in the preliminary design phase of aerospace configurations.[1]

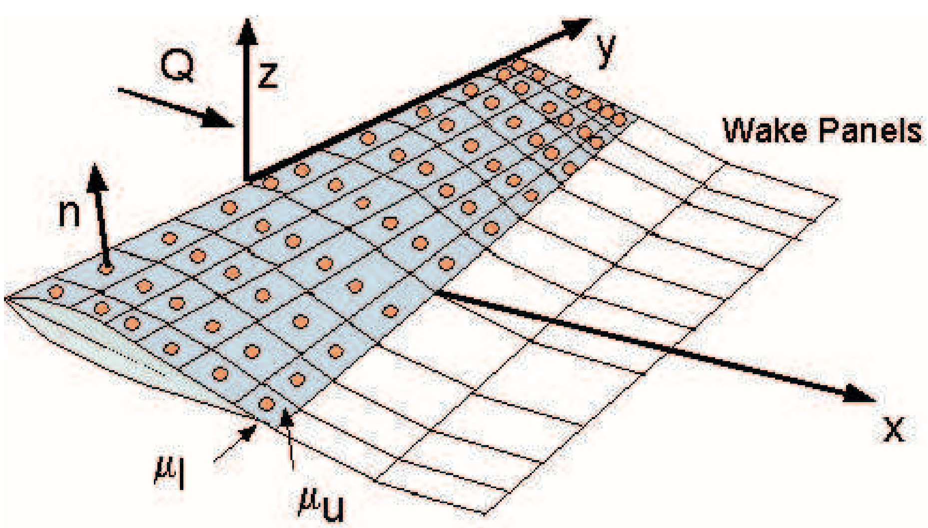

Figure 1: Panels on the wing surface[1] 1

3.1 Governing equations and boundary conditions

Taken from-S.R.Ahmed, V. T. Vidjaja: Numerical simulation of subsonic unsteady flow around wings and rotors. AIAA-94-1943-CP[2]

Consider a body moving through an incompressible and inviscid flow, with respect to a body fixed frame of reference, let the body surface be given by:

$$S_b = 0 \tag{21}$$

It is assumed that the flow remains attached everywhere on the body surface expcept at known and well defined locations such as the trailing edge. The separated flow containing vorticity is confined to sheets of zero thickness which commence at the trailing edge. If the motion of the body starts from a state of rest, then the fluid external to the body and wake remains irrotational for all subsequent times. The governing equations for the velocity potential is the Laplace equation.

$$\nabla^2\Phi(r,t) = 0 \tag{22}$$

which is always linear. The velocity potential is determined subject to the following boundary conditions

1. Far away from body or the wake, the fluid is undisturbed so that

2. The body surface S_b is a stream surface, i.e. the normal component of the total velocity on the body surface at any instant of time is equal to zero.

$$[\nabla\Phi(r,t) - V_b(r,t)].n(r,t) = 0 on S_b(r) = 0 \tag{23}$$

3. Kutta-Joukowski condition of finite fluid velocity at the trailing edge at all times expect the starting instant.

4. Since the wake cannot sustain any load, the pressure across it must br continuous.

From Green's theorem[5]it follows that any solution of Laplace's equation can be expressed in an integral form over the boundary surface S_b, where the surface is replaced by singularity distribution of unknown strength:

$$\Phi_P(t) = -1/4\pi[\int\int \frac{1}{r(S,P)}\frac{\partial\Phi(S,t)}{\partial n}dS - \int\int \Phi(S,t)\frac{\partial}{\partial n}(\frac{1}{r(S,P)})dS] \tag{24}$$

Here Φ_p is the potential at any point P.

The term $\frac{\partial \Phi}{\partial n}$ under the first integral in the above equation is the velocity normal to the surface S. The argument of the first integral can thus be interpreted as the potential of a three-dimensional source of strength $\sigma = \frac{\partial \Phi}{\partial n}$:

$$\Phi_s(S,t) = -\frac{\sigma(S,t)}{4\pi r(S,P)} \tag{25}$$

Similarly the argument of the second integral in equation(32) can be interpreted as the potential of a 3-D doublet of strength $\mu = \Phi$:

$$\Phi_d(S,t) = \frac{\mu(S,t)}{4\pi}\frac{\partial}{\partial n}(\frac{1}{r(S,P)}) \tag{26}$$

$$\Phi_P(t) = -1/4\pi[\int\int \sigma(S,t)\frac{1}{r(S,P)}dS - \int\int \mu(S,t)\frac{\partial}{\partial n}(\frac{1}{r(S,P)})dS] \tag{27}$$

The above equation represents a general solution of the Laplace equation. To render the solution unique, additional conditions for physics of the flow under study have to be invoked.

Consider now for example the flow around a wing, a source/sink distribution can be placed over the wing surface(Figure1). This simulates the finite thickness of the wing profile. The wing profile is extended by a small amount at the trailing edge along the tangent to the mean line. If the thickness of this extension is collapsed to zero, then the source/sink distribution on this extended trailing edge is transformed into a doublet distribution. Following the concept of Kraus[9], this doublet distribution is continued inside the profile along the mean line, its strength decreasing to zero at the leading edge. The variation of the doublet strength between zero at the leading edge and a finite, as yet unknown value at the trailing edge is arbitrary. With this artifice large gradients in singularity strengths at the trailing edge and associated problems during the numerical solution are avoided.[2]

3.2 Numerical Interpretation

The numerical procedure is as follows: the wing is divided into finite surface elements(we call them panels)such that the σ(S, t) and μ(S, t) are finite but constant in each panel. The

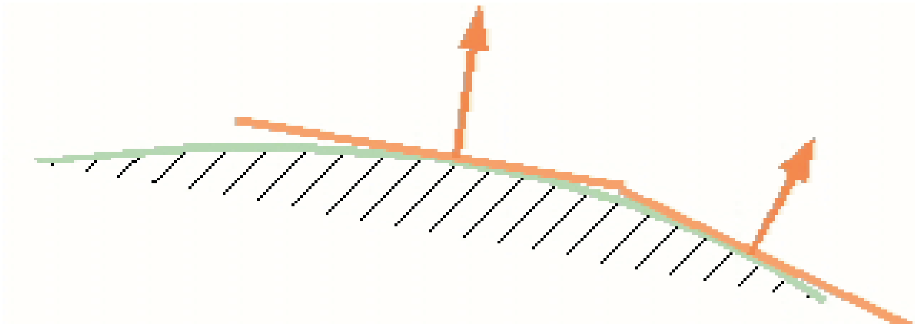

Figure 2: Surface Panel[1]1

strengths vary from panel to panel depending on the boundary conditions. Additionally, the equivalence of constant strength doublet panels and vortex rings is used to replace the doublet panels of strength μ by vortex rings of same strength μ placed on the perimeter of the doublet panels. Induced velocities for a quadrilateral doublet panel are then , for example, calculated from the four vortex line filaments at the panel edge using Biot-Savart law. Substituting equation(35) in equation(31) one obtains

$$\nabla(-1/4\pi[\int\int \sigma(S_b,t)\frac{1}{r(S,P)}dS - \int\int \mu(S_w,t)\frac{\partial}{\partial n}(\frac{1}{r(S,P)})dS])\cdot n_p(r) = V_P(r,t)\cdot n_P(r) \tag{28}$$

If the wing surface S_b is discretized by m plane surface elements and the internal surface S_w by m' plane surface elements, it follows from the equation (36)

$$\nabla(-1/4\pi[\sum_{j=1}^{m}\sigma(t)\int\int\frac{1}{r(S,P)}dS_j - \sum_{j=n}^{k}\mu(t)\int\int\frac{\partial}{\partial n}(\frac{1}{r(S,P)})dS_j])\cdot n_p(r) = V_P(r,t)\cdot n_P(r). \tag{29}$$

Here S_j denotes the area, σ and μ the source/sink or circulation strength of an element j respectively. The wing surface S_b and internal mean surface S_w have been approximated as

$$S_b \approx \sum_{j=1}^{m} S_j \tag{30}$$

$$S_w \approx \sum_{j=n}^{k} S_j \tag{31}$$

with

n = m+1

k = m+m‘

The velocity V_p is the net velocity experienced by the point P due to all the motions of the wing. For example for a wing in forward motion V_p has basically two components:

$$V_P(r,t) = V_\infty(t) + V_c(r,t) \tag{33}$$

where $V\infty$ is the translational velocity and V_c is the resultant of the pitching and plunging velocity. For a rigid blade in forward flight, based on known geometry, flight conditions and control inputs, the terms on right hand side of the equation(41) are known. Futher, with known blade geometry and the chosen surface discretization, the integrals on the left hand side of equation(36) as well as the direction of unit normal vectors ncan be evaluated.Imposing equation(36) at a number of discrete points P on the body and Kutta panel surface leads to a system of linear algebraic equations whose iterative solution gives the strength of the singularities σ and μ for each of the generic points P at a time instant t. As mentioned earlier, the variation of circulation strength on the profile mean line is prescribed. The circulation strength of the Kutta panel is set equal to that at the trailing edge. Once the Kutta panel strength is known the relative strengths of all vortex panels on the mean surface are known. The number of unknowns is thus equal to the number of surface elements (panels)on the body surface plus the number of kutta panels. Equation(35) is satisfied at one collocation point per panel.

The calculation proceed in the following manner: At time t = 0, the body or the rotor is impulsively started from rest from a given initial position. This means that the right hand side of equation(36) is evaluated with full values of V_∞ and V_c. At this instant there is no wake present. With the solution of the system of equations (36) and (37), the strength of the singularities on the wing surface and circulation strength of vortex rings on Kutta panels is known. Using these singularities, the induced velocities at the downstream corner points of the Kutta panel are evaluated. A straight vortex element is released now from the trailing edge of each Kutta panel. The ends of this vortex element are moved with calculated induced velocities plus the velocity components due to translation, rotation and other motions of the body or the wing over a time interval Δt. This vortex filament, together with the so created downstream segments and the Kutta panel trailing edge, comprises a quadrilateral ring vortex. This row of vortex panels released from the kutta panels is the first increment of the blade wake. The distortion of the wake is effected by the differing velocities with which the end points of the released vortex filaments move. Once a row of vortex panels has been assigned a certain spanwise variation of circulation, the circulation distribution for this particular row of panels remains constant as the wake panels move and distort in space. For the next time step, the system of equations generated by equation(37) is solved anew, taking into consideration now the induction of the first row of vortex panels at all collocation points. This process is repeated until the aerodynamics converge to a desired behaviour.With each time step the number of vortex panels comprising the wake grows and with it the number of Biot-Savart operations to compute the wake induced velocities.[2]

3.3 Pitching wing

An unsteady 3-D panel code is modified to simulate an unhinged rotor wing with symmetric NACA 0012 airfoil is translating and pitching simultaneously. The airfoil and wake coordinates are rotated in a prescribed manner aboout the quater cord line of the wing. The wing is positioned with an 6°. The wing oscillated about the given angle of incidence with an amplitude of 2.75°. The angle of attack at a given instant is given by:

$$\alpha(t) = \alpha_0 + \tilde{\alpha} sin(ft) \tag{34}$$

with α_0 the initial incidence, $\tilde{\alpha}$ the amplitude of angular oscillation, f the oscillation frequency and t the time. To validate the code, the unsteady results for f = 0 were compared with results computed by a seperate steady solver. The results match closely suggesting that the code is accurate. After validating the code, the computations were done for two frequencies f = 0.21 and f = 0.42. The computation was run for 500 time steps with a step size of 30°. The resulting wake develpoment was plotted in tecplot. After the data was extracted the propulsive efficiency was calculated in each case.

3.4 Results

First, to validate the code the results were computed for zero pitch(no pitching-steady case), the same were compared with another program which computes data for steady cases. In short, an unsteady solver is compared to a steady solver. Both the results are supposed to match as the unsteady solver is forced to compute for zero pitch(same as a steady case). As table 1(page 20) suggests, data aquired from both the sources match validating the unsteady code.

In the next few pages, the results of the computed results are documented. The contents of the data extracted are as follows:

1. The plots of C_A, C_W, C_M vs time.
2. The propulsive efficiency calulated for each case.
3. The wake plot generated in tecplot.

Table 1: comparison between steady and unsteady code

	unsteady	unsteady	steady	steady
angle of attack	C_X	C_Z	C_X	C_Z
1.0	0.00089	0.0789	0.00101	0.08
2.0	0.00378	0.0789	0.00101	0.08
-1.0	0.000904	0.0799	0.00102	0.081
-2.0	0.00379	0.15923	0.004	0.162

As seen in the table, the unsteady data agrees very well with the steady data. Thus, validating the code.

3.4.1 Propulsive Efficiency

For a wing which is flapping, the propulsive efficiency is given by:

$\eta = \bar{C_T}/(\dot{z}.\bar{C_L} + \bar{C_M}.\bar{\dot{\alpha}})$

For a wing which is only pitching, $\dot{z}$=0

(a)Propulsive Efficiency for frequency = 0.21

$\bar{C_T}$=0.042(Figure 3)

$\bar{C_M}$=0.125(Figure 4)

η=52.36

(b)Propulsive effiency for frequency = 0.42

$\bar{C_T}$ = 0.042

$\bar{C_M}$ = 0.125

η = 26.18

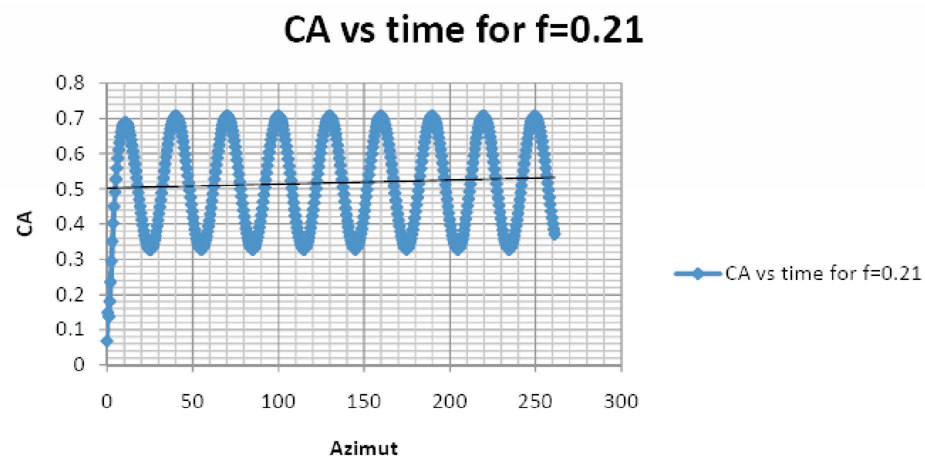

Figure 3: C_A vs time for f=0.21

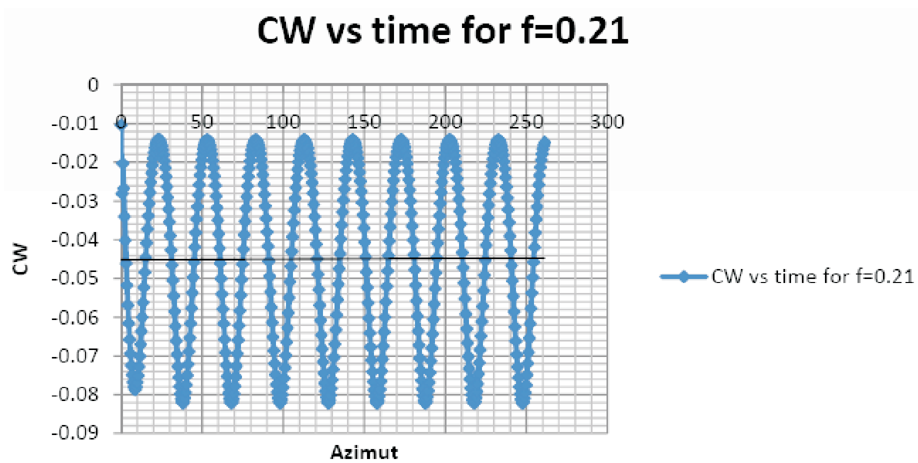

Figure 4: C_W vs time for f=0.21

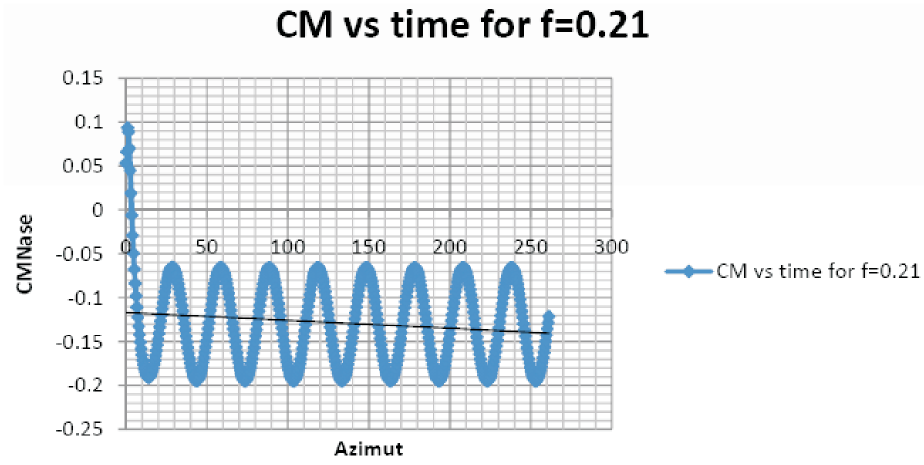

Figure 5: C_M vs time for f=0.21

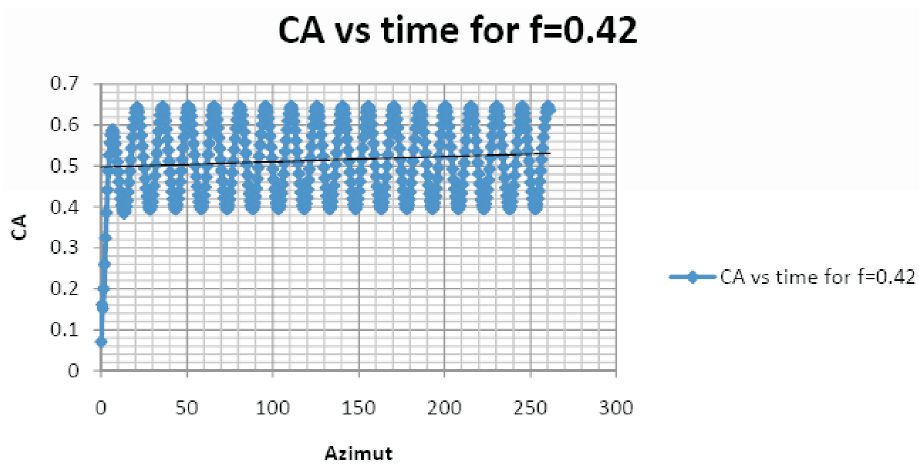

Figure 6: C_A vs time for f = 0.42

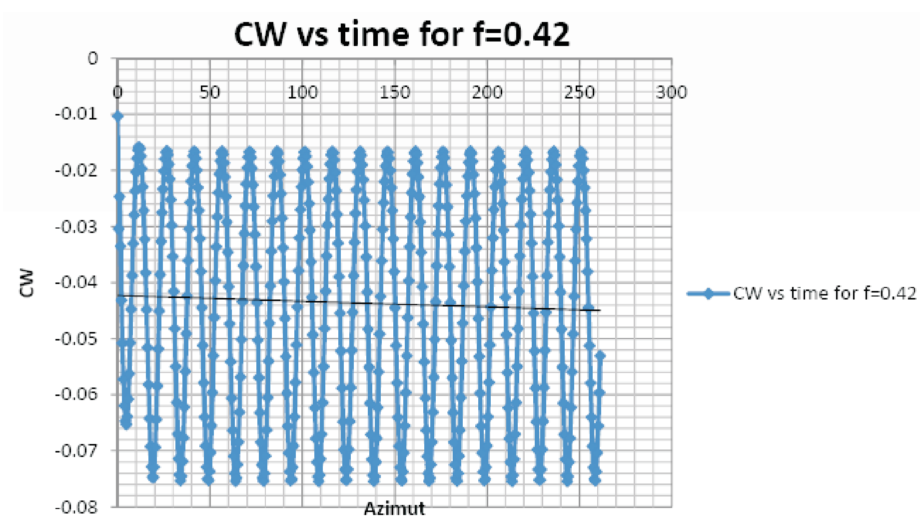

Figure 7: C_W vs time for f = 0.42

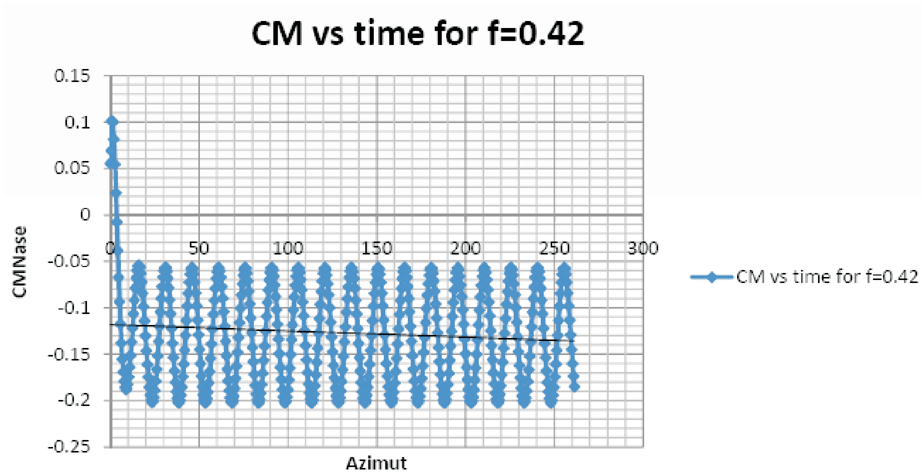

Figure 8: C_M vs time for f = 0.42

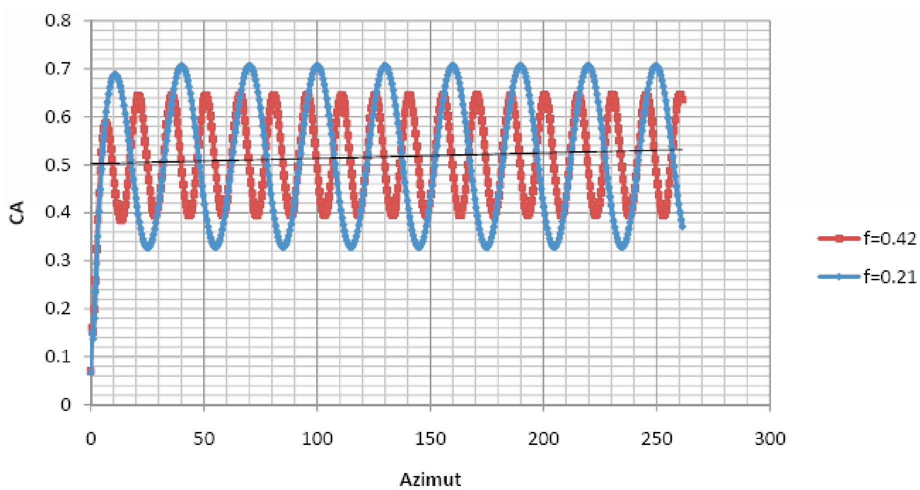

Figure 9: C_A vs time superposition of f = 0.21 and f=0.42

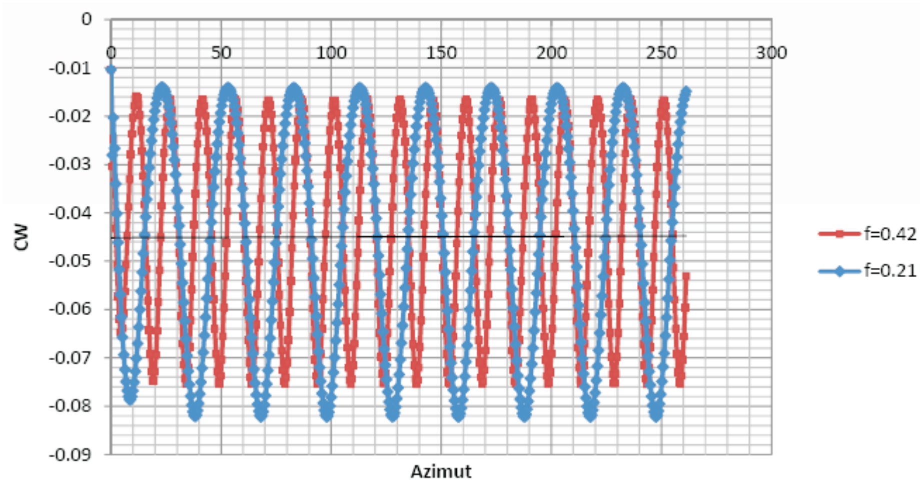

Figure 10: C_W vs time superposition of f=0.21 and f=0.42

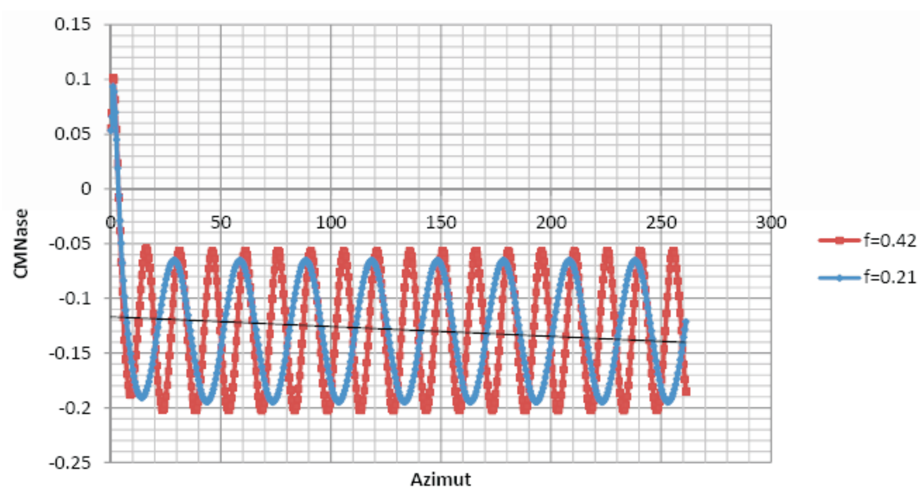

Figure 11: C_M vs time for f = 0.21 and f = 0.42

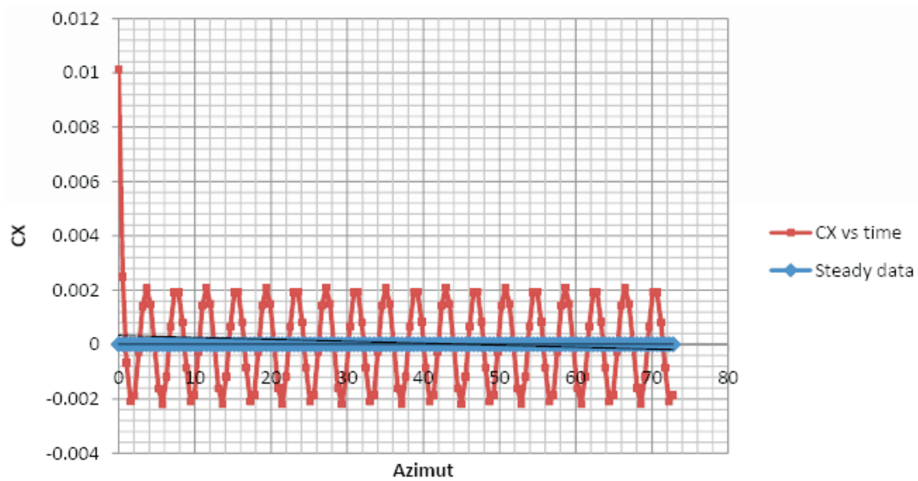

Figure 12: C_X vs time : comparison with the steady code

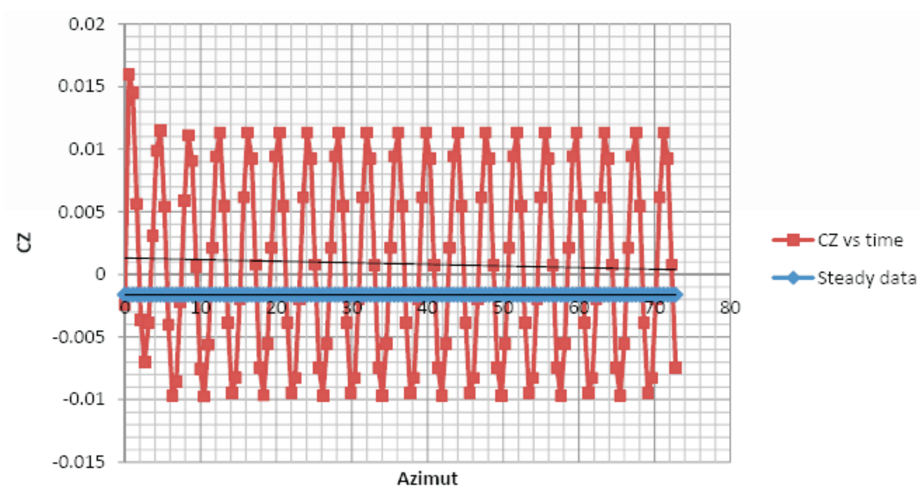

Figure 13: C_Z vs time : comparison with the steady code

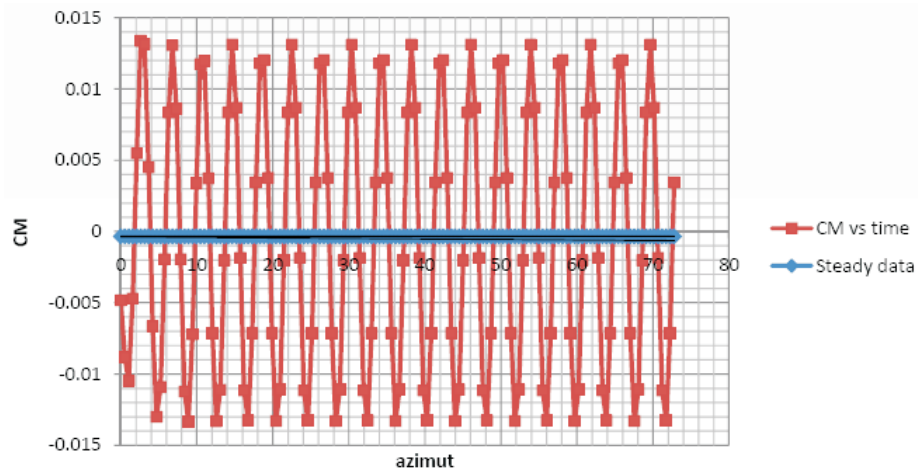

Figure 14: C_M vs time : comparison with the steady code

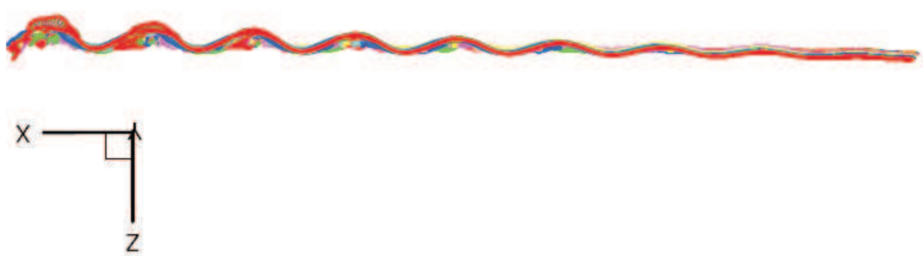

Figure 15: wake plot for f = 0.21

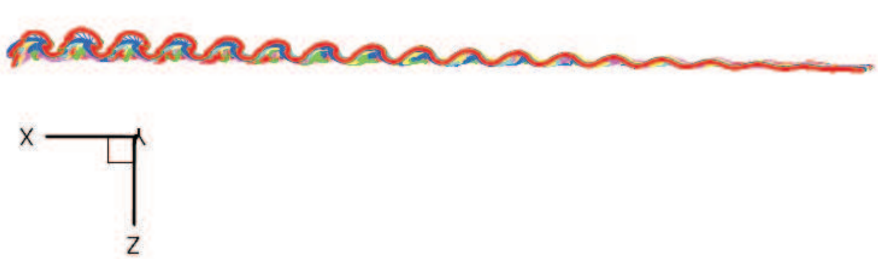

Figure 16: wake plot for f = 0.42

4 Summary

1. An airfoil which is exposed to steady wind and which executes sinusoidal pitching or plunging can generate thrust similar to a flapping bird wing or a pitching fish tail.

2. The vortical signature of this thrust is a karman vortex street in reverse.

3. When an airfoil executes pitching, it produces a net angle of attack. As it executes pitching, there is a constant creation and shedding of vortices. This results in a normal force vector with both thrust and lift components, the behavior of vortices determine the lift and thrust forces.

4. The unsteady panel code is advantages compared to the other methods as it can capture both the loads on the wing and the wake vortices generated in reasonable time

5. why study wakes: The wakes impinge objects(trailing helicopter blade etc.) and thus affect the aerodynamics and inturn the propulsive efficiency of the trailing airfoil.

6. The change in the vortex dynamics over/around the airfoil will determine how the Lift and thrust coeff. change with time.

7. The wakes plotted in tecplot agree well with those found in literature.

5 Future Study

In future, I plan to investigate the aerodynamics of the plunging airfoil. I also plan to study a 2 wing system, in which we study the effects of the vortices generated by the leading wing on the trailing wing(Blade vortex interaction). By studying the above, other phenomenon like **hovering** (with regard to dragon fly flight) can be looked at.

6 References

(1)Michael J.C.Smith, Peter J.Wilkin and Marc H.Williams:The advantages of an unsteady panel method in modelling the aerodynamic forces on rigid flappign wings.

(2)S.R.Ahmed, V.T.Vidjaja: Numerical simulation of subsonic unsteady flow around wings and rotors.

(3)Fundamentals of Fluids: I. G. Currie

(4)Fundamentals of aerodynamcics by J. D. Anderson Jr.

(5)Lamb H:Hydrodynamics

(6)Insect normal hovering flight in ground effect: Tong Gao and Xi-Yun Lu, Physics of fluids 20, 087101(2008)

(7)Numerical Study of two-winged insect hovering flight: Yanjun Liu, Nansheng Liu and Xiyun Lu, Adv. Appl. Math. Mech., Vol.1, No.4, pp.481-509.

(8)Aerodynamics of Low Reynolds Number Flyers: Wei Shyy, Yongsheng Lian, Jian Tang, Dragos Viieru, Llao Liu.

(9)An experimental and numerical investigation of flapping-wing propulsion:K.D.Jones, M. F. Platzer, AIAA-99-0995

(10)Kraus, W:Das MBB-UFE Unterschall-Panalverfahren. Messerschmitt-Bolkow-Blohm GmbH, Munchen, Germany, Report MBB-UFE 633-70.1970

(11)Low-Speed Aerodynamics - Joseph Katz, Allen Plotkin