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DEVELOPMENT OF AIRCRAFT IN-FLIGHT LOADS SOFTWARE WITH UNSTEADY VORTEX LATTICE METHOD (UVLM)

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ABSTRACT

This paper describes the development of an in-flight loads calculation software which, in addition to calculating the aerodynamic and structural loads, is capable to solve a fully integrated aeroelastic model. The software contains an unsteady vortex lattice Method (UVLM) based aerodynamics solver, developed in FORTRAN, to compute the aerodynamic loads on the aircraft, while dedicated structural dynamics and flight dynamics solvers are also included, to calculate the modal characteristics and rigid-body dynamics of the aircraft respectively. The solver modules are wrapped in a user-interface and graphics engine module, especially developed in C++. The UVLM model also includes vortex shedding and free-wake relaxation routines to better capture the unsteady motions. Its potential applications include design and efficient analysis of novel aircraft configurations ranging from small unmanned air vehicles to regional transport aircraft. The results are verified by theory as well as published experimental data and show that the software tool developed, and the methodology adopted, is capable of accurately simulating the key features of unsteady flight and predicting aerodynamic loads.

INTRODUCTION

Every commercial aircraft structural design has to pass through a minimum acceptable means of compliance of aerodynamic and structural loads as per applicable aviation regulations such as FAR and CS. Hence, the need for fast and efficient computation of aircraft loads is highly desired, especially during the conceptual design phase. Aerodynamic and structural non-linearities are known to play an instrumental, and often counter-intuitive, role in the dynamics of flexible aircraft, but the computationally expensive nature of aeroelastic analysis and lack of high-fidelity quick analysis tools dictates that aeroelastic requirements are not taken into consideration until later in the design process. Design of modern, highly flexible aircraft requires that aeroelastic requirements must be considered in the

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beginning of the design phase in order to avoid the expensive redesign process later on.

The development and overview of a novel medium-fidelity aero-structural design and analysis software tool is presented herein with unsteady vortex lattice method (UVLM) coupled with finite element structural mass and stiffness model. The software suite (known as DynaFlight) consists of three primary modules; the graphics and UI module, flight dynamics module, and the aerodynamic solver module. The tight coupling between the flight dynamics and aerodynamic modules, allows the user to simulate diverse loading conditions that arise during the steady and unsteady flight, as well as solve static aeroelasticity problems. The interconnectivity of the various modules is shown in figure 1.

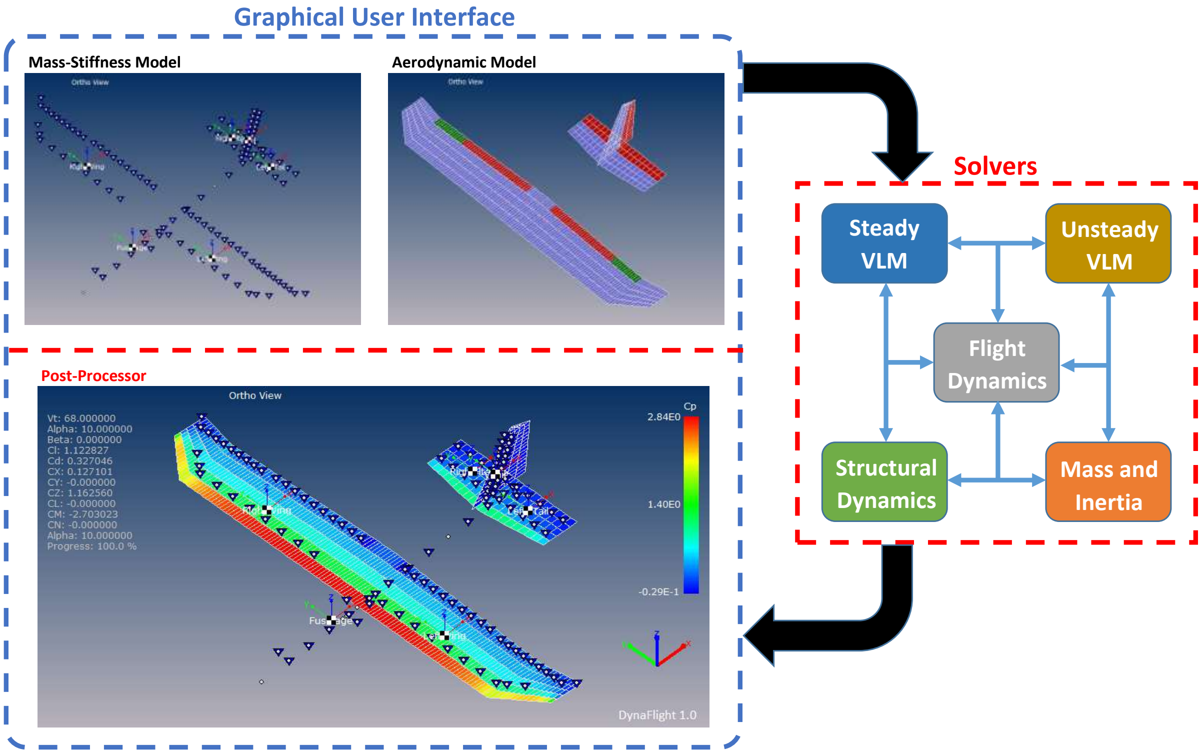


Figure 1: Integration of various modules

In order to study the aeroelasticity of flexible fixed-wing aircraft, the main purpose of unsteady aerodynamic modeling is the efficient and accurate prediction of transient flow fields in the vicinity of the aircraft and the resulting aerodynamic forces acting on that aircraft. The software presented allows the user to model and solve for a variety of unsteady motions. A few examples of flow unsteadiness, which may be solved separately or together, are listed below:

1) Motion-induced unsteadiness, caused by movement of the geometry over which the air is flowing; for example flapping flight and vibration due to structural deformation.

2) Externally-induced unsteadiness caused by interferences in the incident flow; for example gust loads and wake vortices.

3) Flow unsteadiness, caused by flow fluctuation periodically over time; for example turbulence and buffeting.

Historically, one of the most intriguing unsteady problem for researchers has been the pitch-plunge motion of the wing. This is due to the fact that many creatures, in nature, use oscillatory and undulatory motions to generate lift and thrust with and efficiently exploit unsteady aerodynamic effects in flight. Numerous research efforts and wide range of investigations have been conducted in the past, to study and formulate computational models for flapping flight aerodynamics. Successful attempts have recently been made in applying UVLM methodologies to unsteady flight [Han, 2016; Long, 2004], but the application of these novel methodologies to efficient and predictive, design analysis software tools has been very limited. Although these research efforts have considerably improved our understanding of unsteady aerodynamics for the design and development of flapping wing micro-aerial vehicles and modern flexible aircraft, they are typically implemented for classical maneuvers of specific geometries.

The DynaFlight software suite aims to employ the unsteady vortex lattice method in a comprehensive framework, to grant user the freedom to model, simulate, and analyze various unsteady motions for any arbitrary geometry.

Software Architecture and Parallel Computing

The overall software framework is highly accelerated using multicore processor, GPUs, and MPI Cluster based parallel processing. The custom designed 3D graphics engine is built without OpenGL, DirectX or any other 3rd party 3D graphics library, and it makes use of multicore and GPU based execution to accelerate 3D rendering. The unsteady VLM and structural dynamics modules on the other hand employ multicore, GPUs, as well as MPI Cluster based acceleration to effectively deal with the compute-intensive nature of their mathematical models. The software framework employs an intelligent scheme for overall processing load balancing and data partitioning. This scheme automatically determines parallelizable nature of a computational task based on some preset metrics and employs most suitable execution scheme for it, for example it prefers multi-core execution over GPUs based acceleration for highly serial (non-parallelizable) computational tasks or when processing data partition is significantly small to exclusively launch a GPUs or Cluster based execution kernel. The overall performance gain has been observed to be more than 10 times for typical test cases.

UNSTEADY VLM MODELING

The vortex lattice method (VLM) is a numerical method based on the theory of Potential flow. It is a well understood and widely used method in computational fluid dynamics, which has been developed and refined several times over the past 70 years [Anderson, 1991; Bertin, 1998; Falkner, 1946; Katz, 2001]. VLM was primarily developed for the steady-state flight, however it is also well suited to the unsteady aerodynamic problems (such as flapping flight) because time-marching techniques can easily be applied to it and it can accurately account for the changing vortex circulation over the wing, the time-dependent velocity potential, and the movement of the circulatory wake. Although UVLM does not allow for the modeling of the complete complex physics of unsteady flow like some of the other high-fidelity methods, it is able to efficiently predict the key features and phenomena of unsteady aerodynamics with sufficient accuracy. Therefore, vortex lattice models are widely used in the design and analysis of aircraft and rotorcraft, because of their high efficiency and robustness. Higher order CFD methods are not feasible for unsteady aerodynamic analysis in the early stages of the design process, simply because of the compute-intensive nature of the process.

UVLM analysis begins with the modeling of the lifting surface(s). The DynaFlight software allows the user to model any arbitrary shape of the aerodynamic surface through commonly used design parameters, such as the root/tip chord length, leading-edge sweep angle, span length, and dihedral angle. In addition, multiple surfaces (for example wing-tail configuration) and control surfaces can also be added. The aerodynamic surface is then discretized into a mesh of vortex ring panels, which are arranged in a grid on the lifting surface. User is able to control the mesh density (no. of panels per chord/span) as well as the distribution of the grid lines (for example a finer mesh may be used for the wing tip and coarse mesh for the root). Definition of aerodynamic surface for UVLM analysis is given in figure 2. The analysis is controlled by the flight dynamics module, which simulates the user-defined unsteady motion of the aircraft. As the surface (or aircraft) moves through the fluid, the circulation of the vortex rings is calculated and a row of panels is shed from the surfaces trailing edge at each time step. This process is shown graphically in figure 3. Once the panels are shed, they become part of the wake, and their circulation is changed according to the wake dissipation theory. Shape of the wake is sometimes prescribed in advance based on empirical data, or simply left as and where it was at the time of shedding. However, in reality the wake does not carry any load and it should move with the local velocity of the fluid. Therefore, in the software presented, the wake model is force-free and includes wake stretching, aging (dissipation), and free-relaxation algorithms which move the wake nodes with the local induced velocities and update its strength accordingly. This local velocity is a combination of the velocity induced by the other rings in the wake and the rings bound to the surface.

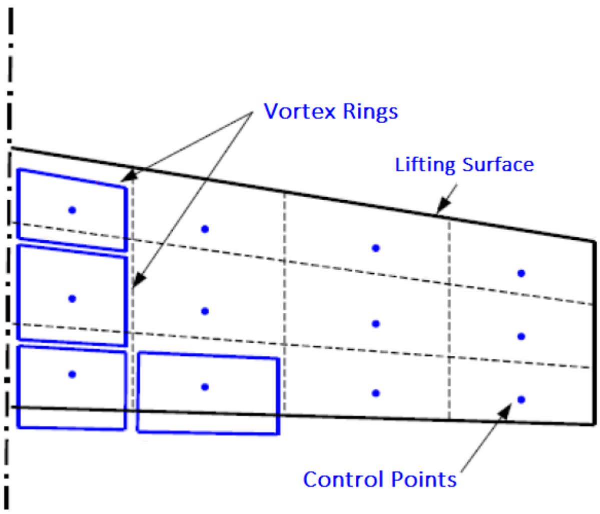


Figure 2: Definition of Aerodynamic Surface in VLM

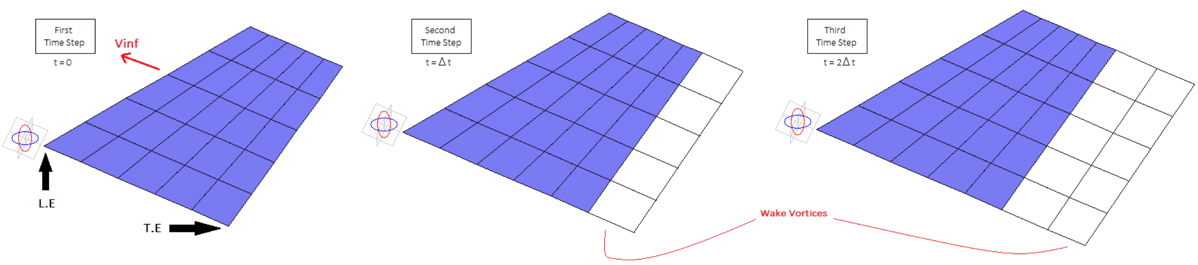


Figure 3: Wake shedding procedure

Boundary Condition

The strength of the circulation (Γ) of the rings bound to the surface is determined by imposing the boundary condition on the surface. The boundary condition applied is of the Neumann type which states that the velocity's component normal to the surface must be zero. Mathematically, this is represented as equation 1 and can be re-written as equation 2.

$$\nabla(\phi + \phi_\infty) \cdot n = 0 \tag{1}$$

$$V_i \cdot n_i = (V_\infty + \sum A_{ij}\Gamma_j) \cdot n_i = 0 \tag{2}$$

In the above equation, V_∞ is the freestream velocity, A_{ij} is the aerodynamic influence coefficient matrix, Γ is the vortex strength, and n is the normal vector of the panel. This boundary condition is also known as the flow tangency condition, and is applied at the control point of each surface panel.

Aerodynamic Influence Coefficients (AIC)

AICs are defined as the velocity induced on a specific control point due to all the vortex rings elements present on the surface. Therefore, the matrix element a_{11} (refer to equation 5) represents the influence of the four elements of the first vortex ring on the first collocation point and so on. The process is carried out for each control point on the surface until the whole matrix is populated. Mathematically, the first element of the AIC matrix is given by equation 3. Where, (u, v, w) are the components of the induced velocity.

$$a_{11} = (u, v, w)_{11} \cdot n_1 \tag{3}$$

The equation for induced velocity is derived from the Biot-Savart law, which states that the incremental velocity $\boldsymbol{v}$ induced by a vortex filament of length $\boldsymbol{l}$ and strength Γ, on a point at distance $\boldsymbol{r}$ is given by:

$$d\boldsymbol{v} = (\Gamma/4\pi)[(d\boldsymbol{l} \times \boldsymbol{r})/r^3] \tag{4}$$

It must be noted that the influence coefficients are calculated using a unit vortex strength. By adding up the effects of each of the four segments in a vortex ring, the influence of every ring on every other ring can be computed to populate the AIC matrix. Equation 5 is then solved at each time step calculate the circulation strength (Γ) of vortex rings bound to the surface.

$$\begin{bmatrix} a_{11} & a_{12} & \cdots & a_{1m} \\ a_{21} & a_{22} & \cdots & a_{2m} \\ a_{31} & a_{32} & \cdots & a_{3m} \\ \vdots & \vdots & \ddots & \vdots \\ a_{m1} & a_{m2} & \cdots & a_{mm} \end{bmatrix} \times \begin{bmatrix} \Gamma_1 \\ \Gamma_2 \\ \Gamma_3 \\ \vdots \\ \Gamma_m \end{bmatrix} = \begin{bmatrix} RHS_1 \\ RHS_2 \\ RHS_3 \\ \vdots \\ RHS_m \end{bmatrix} \tag{5}$$

Here, the right-hand-side (RHS) vector is given by

$$RHS_i = -[U(t) + u_W, V(t) + v_W, W(t) + w_W] \cdot n \tag{6}$$

where, $(U(t), V(t), W(t))$ are the components of the time-dependent kinematic velocity, and $(u, v, w)_W$ are the components of the velocity induced by the entire wake at ith panel. This system of equations is then resolved at each time step according to the time-marching algorithm given in figure 5.

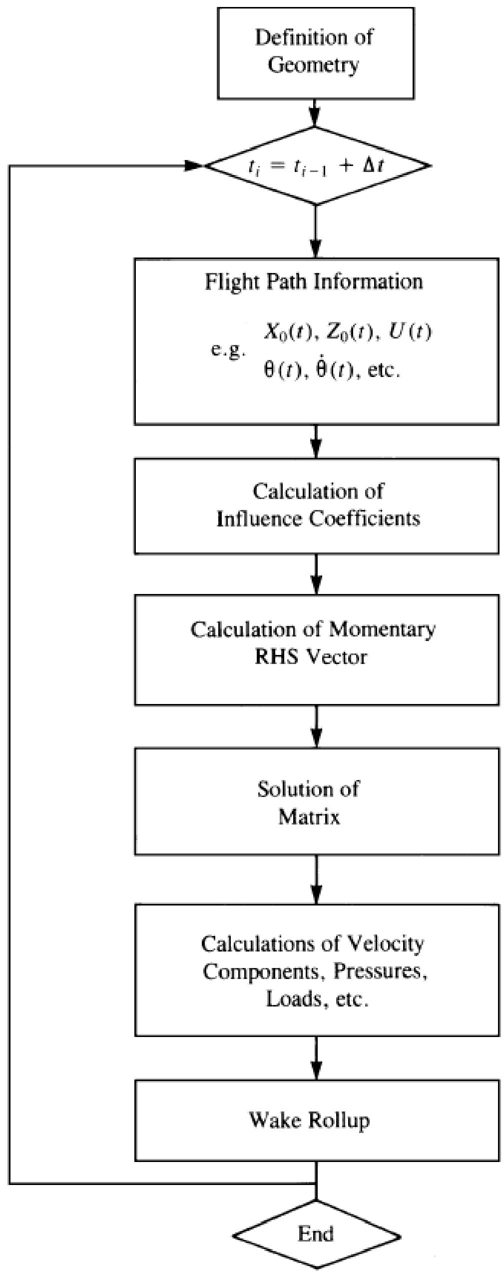


Figure 4: Time-loop Algorithm of UVLM (Adopted from [Katz, 2001])

The resolution of equation 5, requires the inversion of the AIC matrix at every time step. Arbitrary and complex geometries require large number of panels and potentially very high mesh density, thus making the AIC matrix very large and computationally very expensive to solve. Therefore, an efficient matrix inversion algorithm becomes mandatory in unsteady aerodynamics analysis. The DynaFlight software makes use of the Intel® MKL and IMSL® libraries to carry-out the linear algebra operations efficiently. It is useful to note here that if the surface geometry does not change with time, the user may opt to exclude the inversion of AIC matrix from the time loop, thus saving valuable computation time.

Once the vortex strengths are calculated, aerodynamic loads can be determined at the time step. The differential pressure across each panel, derived from the unsteady Bernoulli equation, is given by

$$\Delta p_{ij} = \rho([U(t) + u_W, V(t) + v_W, W(t) + w_W] \cdot \tau_i \frac{\Gamma_{ij} - \Gamma_{i\text{-}1,j}}{\Delta c_{ij}} + [U(t) + u_W, V(t) + v_W, W(t) + w_W] \cdot \tau_j \frac{\Gamma_{ij} - \Gamma_{i,j\text{-}1}}{\Delta b_{ij}} + \frac{\delta}{\delta t}\Gamma_{ij}) \tag{7}$$

where τ_i and τ_j are the panel's tangential vectors in x and y directions, c_{ij} is the panel's chord, and b_{ij} is the panel's width (or span). Once the pressure is determined over an individual panel, the contribution of this panel to unsteady loads is calculated using equation 8. Where (ΔS) is the panel's surface area. The total force acting on the surface is then determined by adding the force of each panel on the surface.

$$\Delta \mathbf{F} = -(\Delta p_{ij} \Delta S_{ij}) \cdot n \tag{8}$$

Since the vorticity of wake, shed from aerodynamic surfaces due to unsteady motion, can significantly affect the aerodynamic forces on the aircraft, the dynamics of the flow field have been studied extensively for a broad range of kinematics and geometries [Freymuth, 1988; Lewin and Haj-Hariri, 2003; Rival et. al, 2009, 2011, 2013; Visbal, 2009; Wang, 2012]. It is important to understand that wake vorticity not only has a significant impact on aerodynamic loads on the surface that sheds the wake but also on other bodies that are present in the wake, such as tandem wings or fins and other aircraft in the vicinity. Therefore, understanding the evolution of the shed vorticity can provide valuable insight into flow interactions between multiple surfaces of the same aircraft, as well as within formation flight of aircraft, UAVs, and MAVs. Although leading-edge vortex (LEV) shedding is an important phenomenon in studying the unsteady aerodynamics of flapping flight, it is not as simple and well-understood as the trailing-edge wake modeling. Hence, LEV modeling is left to more experienced users in the DynaFlight software, whereby the user can specify individual surface panels which will shed wake vortices, before simulating the unsteady motion of the aircraft.

In order to achieve free wake relaxation, each node of every wake panel is updated (or relaxed) according to equation 9. Where, (u, v, w) is the velocity induced by all wake and surface panels on the wake node location (x, y, z).

$$(\Delta x, \Delta y, \Delta z) = (u, v, w)\Delta t \tag{9}$$

Implementing the unsteady vortex lattice methodology described above in an object-oriented software, results in a robust framework which gives user the freedom to model aerodynamic surfaces of any arbitrary geometry, define a custom flight path and structural deformations, and calculate the unsteady loads with physically accurate wake. An example of wake relaxation behind an accelerating wing is given in figure 5.

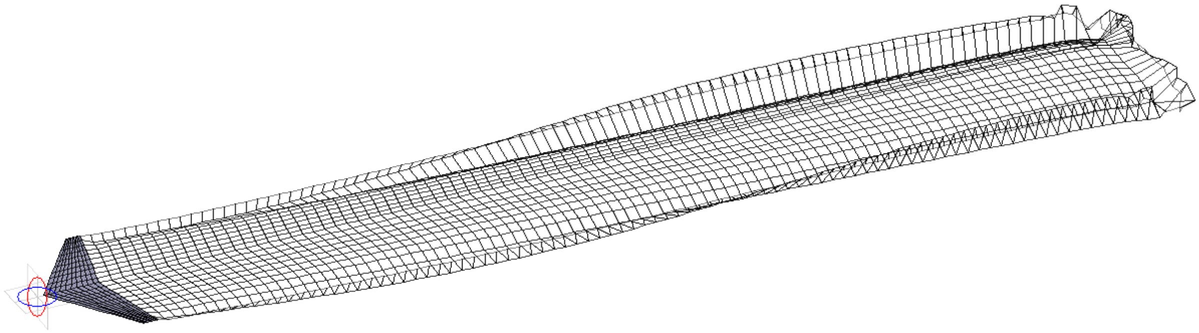


Figure 5: Unsteady Wake behind an accelerating wing

CP Correction from Wind Tunnel or CFD Data

The aerodynamic loads calculated through UVLM procedure are exhaustively verified and sufficiently accurate. However, in principle, it occasionally requires a correction to be made with, wind tunnel test (WTT) measurements or solutions using higher-order computational methods, especially for non-linear analysis. The presented software employs a polynomial integration technique to allow the user to interpolate pressure coefficients calculated through WTT or CFD analysis over the VLM surface mesh, for subsequent loads and stability analysis.

RESULTS

Steady-State Flight

In order to validate the software, a steady flight test case was solved for the F-104 Starfighter aircraft model. Steady VLM is solved without time-marching and with a rigid prescribed wake extending to infinity behind the lifting surface. Various loads and stability coefficients are calculated for the F-104 aircraft model and the results show very good correlation with those published in Appendix B of the Flight Stability and Automatic Control textbook [Nelson, 1998].

Vortex lattice model of the aircraft and its pressure distribution is given in figure 6, while the analysis parameters are listed in table 1. Load coefficients, and lateral and longitudinal stability derivatives, are summarized and compared with reference values in table 2.

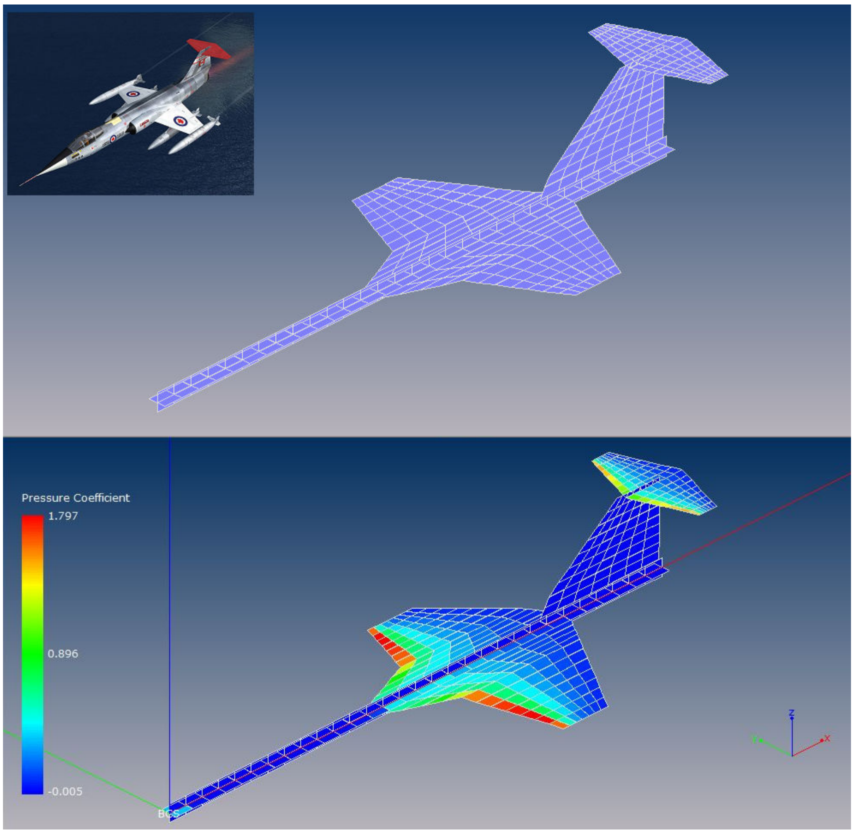


Figure 6: F-104 Starfighter Model and Results in DynaFlight

Table 1: Analysis Parameters

Parameter	**Value**
Mach	0.257
Altitude	*Sea Level*
Wing Span (b)	6.68 m
Reference Area (S)	18.22 m^2
Reference chord length ($\bar{c}$)	2.91 m

Table 2: Loads and Stability derivatives of the F-104 Aircraft

Parameter	**Symbol**	**Computed**	**Reference**	**Difference**
Lift coefficient ($\alpha = 10°$)	C_L	0.709	0.735	3.5%
Drag coefficient ($\alpha = 10°$)	C_D	0.182	0.263	30%
Lift curve slope	C_{L_α}	3.6	3.44	4.6%
Drag curve slope	C_{D_α}	0.66	0.45	46.67%
Pitching moment slope	C_{M_α}	-0.73	-0.64	14%
Dihedral effect	C_{l_β}	-0.164	-0.175	6.2%
Weathercock stability	C_{N_β}	0.52	0.50	4%
Yawing moment stability	$C_{N_{\delta_r}}$	-0.152	-0.16	5%
Pitching moment stability	C_{M_q}	-6.0	-5.8	3.45%

Transient Analysis of a Rectangular Wing

In order to validate the unsteady methodology, a transient analysis of a rectangular wing in a suddenly accelerated plunging motion is conducted. The analysis is carried out for wings of aspect ratios (AR) 4, 8, and 20 and their lift coefficients (CL) are plotted for comparison. Impulsive motion of an AR=4 wing with wake relaxation is shown in figure 7. Katz & Plotkin [Katz, 2001] reported a comprehensive comparison of impulsively started rectangular wings of several different aspect ratios, and the results of present study matches those very well.

Figure 8 shows the transient lift coefficients of the three wing cases analyzed, where $\alpha = 5deg$, $\Delta t = 0.036s$, and the non-dimensional time constant is defined as $\Delta\tau = (V\infty * \Delta t)/c = 1/16$. The shape of the curves clearly depict the characteristic superposition of steady and unsteady part. Since the motion is largely steady, the unsteady part quickly decays and the solution converges to the steady-state solution as the time progresses.

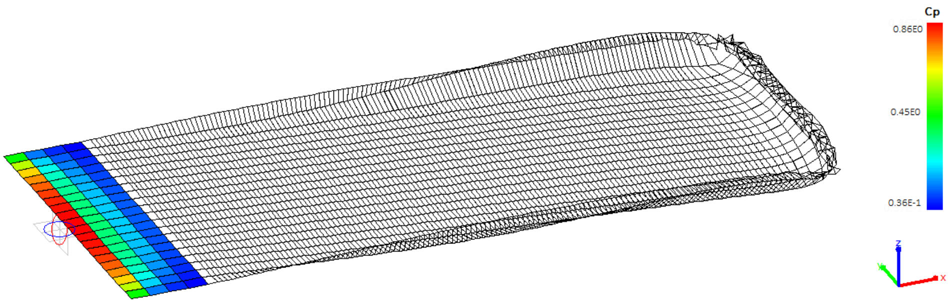


Figure 7: Flight of an impulsively started wing of AR=4

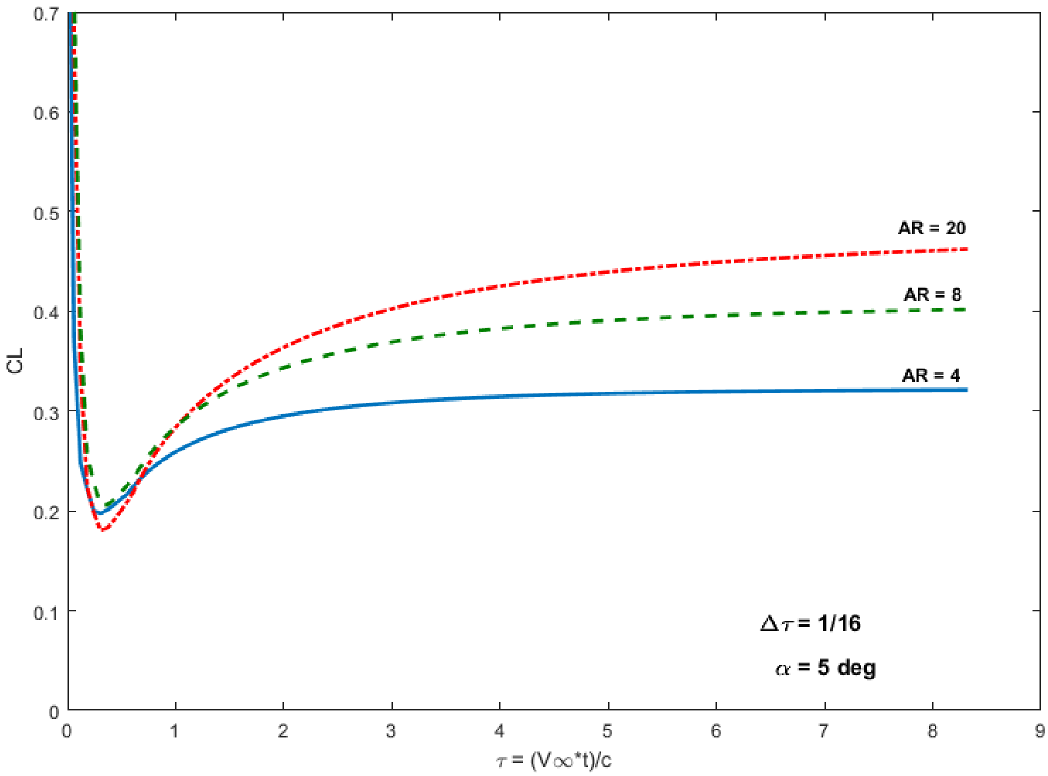


Figure 8: Transient lift coefficient of a rectangular wing

Flapping Wing

Following the validation of the transient loads on a rectangular wing, flapping flight was simulated and analyzed in the DynaFlight software. Flapping flight has been of great interest to scientists and engineers, and it still remains the least well-understood phenomenon of aerodynamics today. A simple tapered wing of aspect ratio 2.66, with relatively low mesh density, is modeled for this analysis. Flapping simulation was carried out with a stroke angle (ψ) of 50 degrees and the reduced frequency was set to 0.5. Reduced frequency is an important non-dimensional parameter, used to study the performance of a flapping (or vibrating) wing. It is given by

$$k = (\omega * b)/V \tag{10}$$

where ω is the circular frequency, b is the wing semi-span, and V is the flow velocity. Flows below $k = 0.05$ is generally considered to be quasi-steady with little influence of unsteady aerodynamics, while value of k greater than 0.2 is considered to be highly unsteady, and the contribution of the unsteady flow cannot be ignored in analysis. The profile of the flapping flight, with free-wake relaxation, is given in figure 9. For this simulation, the wing was set into motion with constant forward speed and fixed base angle of attack (α) of 5 degrees.

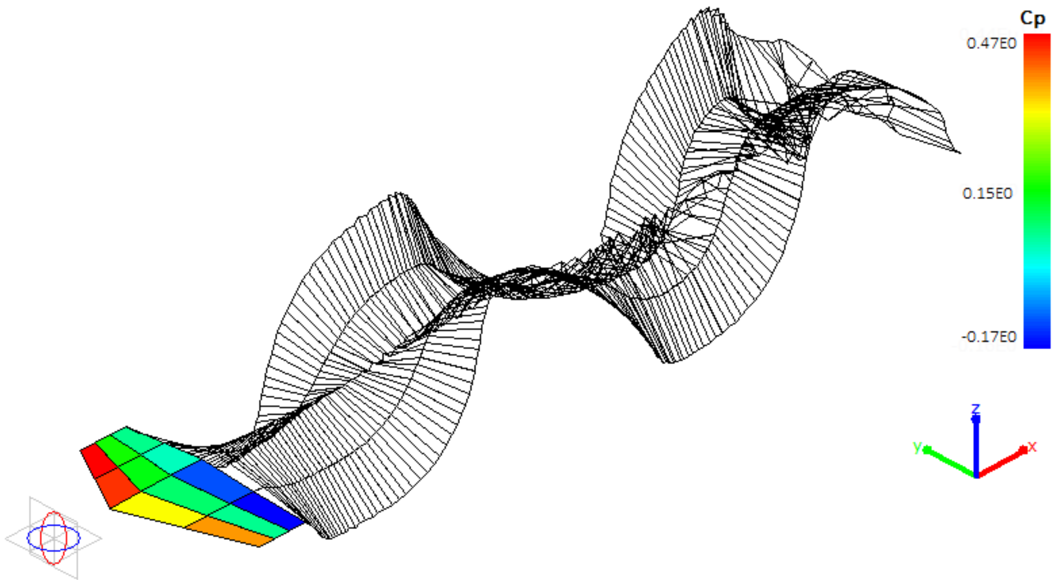
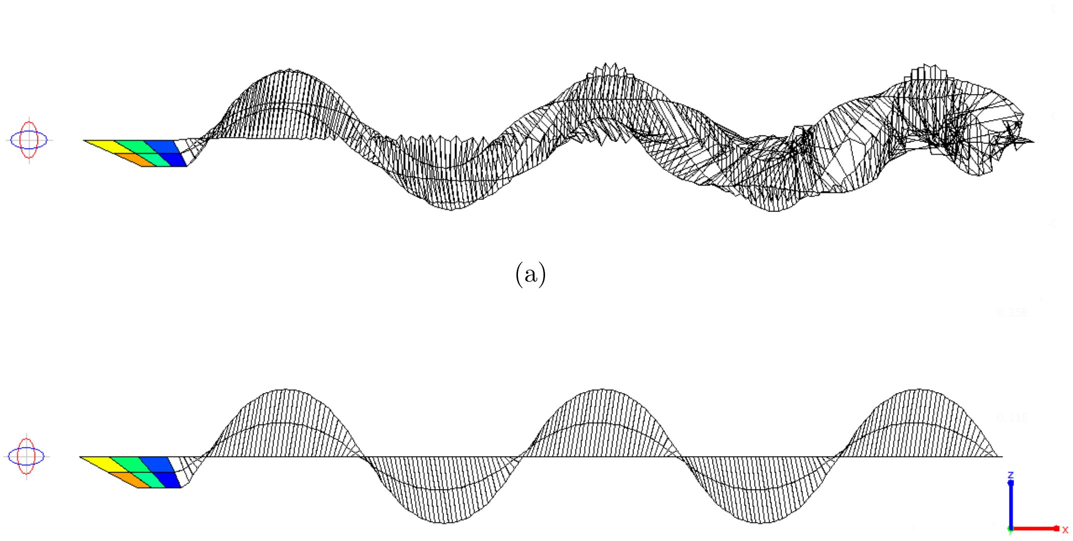


Figure 9: Flapping wing flight: Stroke angle (ψ) = 50 deg, Reduced frequency (k) = 0.5

Wake relaxation is a computationally expensive process, especially for high mesh-density models. Figure 10 provides a comparison of wake panels behind a flapping wing with and without free-wake relaxation. It is clearly seen that the shape of the wake changes significantly as the simulation is progressed, hence affecting the loads acting on the wing. Therefore,it is considered as a trade-off between computational efficiency and solution accuracy.



(a)

(b)

Figure 10: Flapping flight simulation (a) with free-wake relaxation (b) without wake relaxation

Lift (CL) and moment (CM) coefficient plots, for this analysis, are given in Figure 11. The plots provide an insight into the unsteady aerodynamic effects during the flap cycles. It is clear from the CL data that although, most of the lift is produced during the downstroke, the negative peaks during the upstroke are much smaller, thus resulting in net positive lift over a complete flap cycle. The results show good qualitative agreement in amplitude and curve shape for both upstroke and downstroke, with theoretical predictions and simulation results presented by Fritz and Long [Long, 2004].

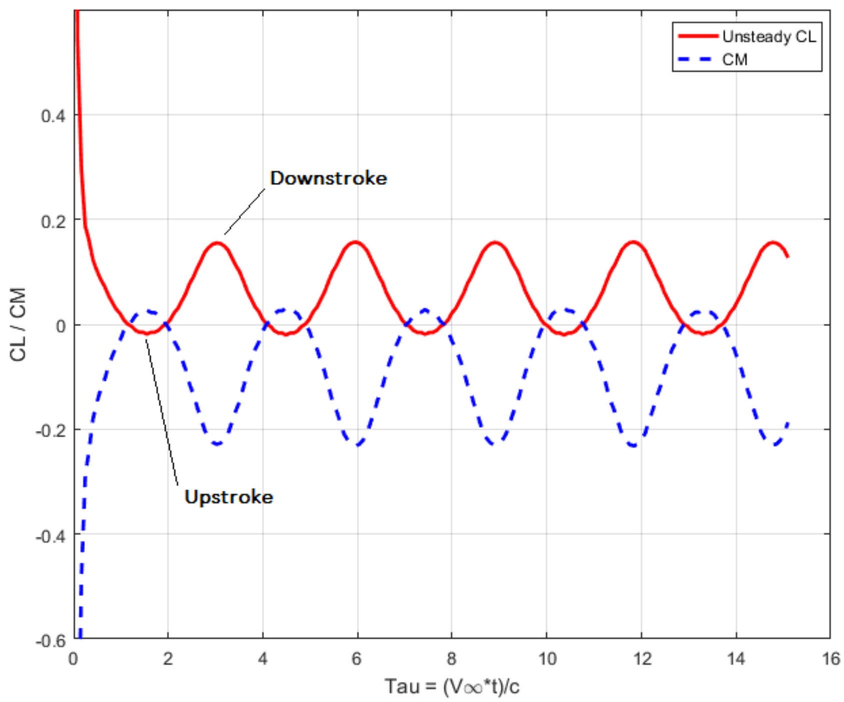


Figure 11: Unsteady CL

Next, a survey of various flapping frequencies was conducted to determine its effect on the lift of flapping wings. For this case, the flight speed was kept constant, while the flapping frequency was steadily increased. Average lift coefficient (CL) vs flapping frequency (ω) plot is given in figure 12. As expected, average lift produced by the upstroke and downstroke of the flap cycle increases as the flapping frequency (and reduced frequency) increases.

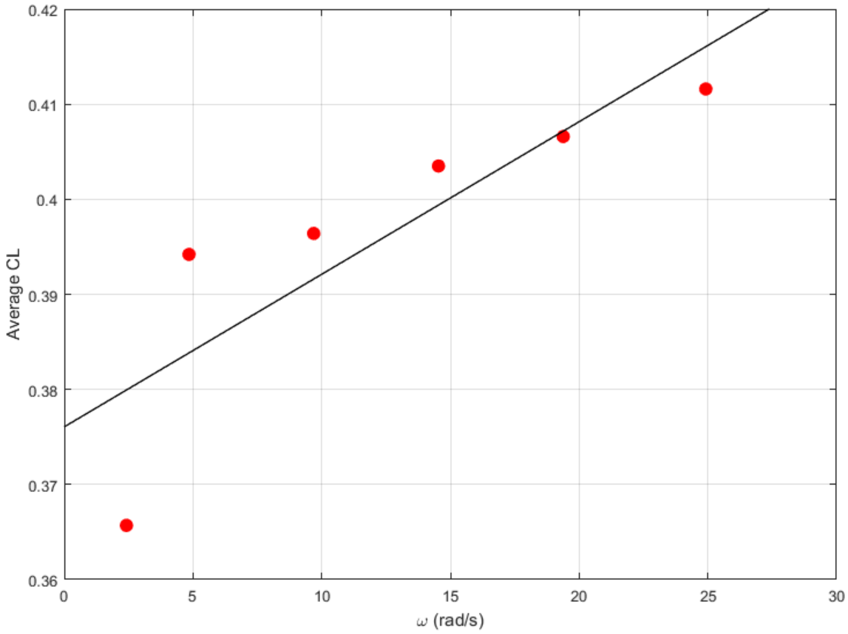


Figure 12: Average lift coefficient for various flapping frequencies

CONCLUSION

A computational model for the prediction and study of unsteady aerodynamics, for arbitrary geometries and a range of flight conditions, is developed within the framework of a medium-fidelity aero-structural optimization capability software. The software presented here is intended to be used for a wide variety of challenges faced by design engineers. Strong coupling between the flight dynamics, structural dynamics, and aerodynamic analysis modules allow for the rapid design and analysis of fully flexible large aircraft, while also having the provision to study and optimize the wingbeat patterns and geometry of micro aerial vehicles (MAV). UVLM model is implemented in the unsteady aerodynamic solver, which includes free-wake relaxation, aging, and dissipation to achieve good accuracy with minimum vortex ring panels. The complete software suite is developed using object-oriented C++ and FORTRAN code, utilizing the IMSL® and Intel® MKL libraries for most efficient computation of mathematical equations.

The overall encouraging results of the test cases, motivates us to further develop the software. Future enhancements include solution of dynamic aeroelasticity problems as well as development of the stability and control module for the analysis of aeroservoelastic problems.

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