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Flight Dynamic Modelling and Simulation of Large Flexible Aircraft

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Abstract

The drive for aircraft efficiency and minimum environmental impact is requiring the aerospace industry to generate technologically innovative and highly integrated aircraft concepts. This has changed the approach towards conceptual design and highlighted the need for modular low fidelity aircraft simulation models that not only capture conventional flight dynamics but also provide insight into aeroservoelasticity and flight loads. The key aspects that drive the need for modularity are discussed alongside integration aspects related to coupling aerodynamic models, flight dynamic equations of motion and structural dynamic models. The details of developing such a simulation framework are presented and the utility of such a tool is illustrated through two test cases. The first case focuses on aircraft response to a gust that has a spanwise varying profile. The second investigates aircraft dynamics during control surface failure scenarios. The Cranfield Accelerated Aeroplane Loads Model (CA^{2}LM) forms the basis of the presented discussion.

Keywords: modelling, simulation, flight dynamics, flexible aircraft, aeroelastic coupling

1. Introduction

Today's concerns regarding growth in the demand for air transport and the environmental impact of aviation has resulted in active efforts by airframe manufacturers to design more efficient aircraft. They have adopted a strategy that sees an incremental introduction of novel technologies, where at each stage the components that constitute the aircraft become more integrated with each other. This effectively provides the opportunity to build the multi-disciplinary design tools and experience needed to develop radical configurations. As a result, the technical disciplines in aircraft design which have traditionally been relatively independent,

such as aeroelasticity and flight dynamics, must now integrate. This chapter aims to present the methods used for developing modelling and simulation tools that are needed to facilitate such an integrated approach, especially focusing on large flexible aircraft.

The traditional approach to modelling and simulation of aircraft flight dynamics has framed the problem in the form of the equations of motion (EoM) that couple nonlinear inertial components with quasi-linear aerodynamic models [1, 2]. This has been found to be satisfactory when modelling the flight dynamics of rigid aircraft, but the assumptions of linearity in the method used to formulate the aerodynamic model remains the primary limitation of this approach. Typically, this limitation is the cause of significant uncertainty early in the aircraft design process where engineers can only resort to either empirical methods or panel based methods. For conventional tube and wing configurations, the civil aviation industry has developed and modified these methods based on extensive testing and operational data. On the other hand, the radical configurations seen in the military domain rely on significant effort put towards the identification of aerodynamic characteristics and validation of models during the expensive flight test phase. The latter may often span the entire service life of the aircraft [3, 4].

Accurate modelling and simulation of novel concepts aimed to address today's societal concerns is needed to enable the multidisciplinary approach necessary for design. However, it cannot resort to the knowledge gained either from significant operational data or extensive flight test data. As a result it can only rely on a physics based approach and moreover, this approach needs to be modular if it is to assist in the necessary multidisciplinary design process. Within this chapter, a brief review of past methods for modelling and simulation of flexible aircraft is presented before the physics based modular approach is discussed. This is followed by details of the methods needed to integrate aerodynamics, structural dynamics and flight dynamics within a single simulation framework. Finally, the reader is presented with two test cases that demonstrate the use of such a framework in aircraft design. The Cranfield Accelerated Aeroplane Loads Model (CA^{2}LM) [5, 6] forms the basis of the discussion presented in this chapter.

2. Review of past methods

An extended version of the Collar's triangle shown in **Figure 1** highlights the physical phenomena that need to be integrated for accurate modelling and simulation of flexible aircraft. Traditionally the flight dynamics community has focused on the link between inertial dynamics and aerodynamics and it assumes structural dynamics to occur at far higher frequencies than those of rigid-body dynamics. The vice versa is true for the structural dynamics community who have mainly focused on specific loads cases for sizing airframe components. The development of aircraft such as the Boeing 747 [7], which was exceptionally large, and the Rockwell B-1 [8] with its flexible fuselage made it necessary for flight dynamics and structural dynamics to be integrated. The work done by Schmidt and Waszak [9] is an early example of such an integrated modelling approach carried out from a flight dynamicist's perspective. The approach retains the inertial components of the classical nonlinear six degree of freedom (6-DoF) equations [1, 2].
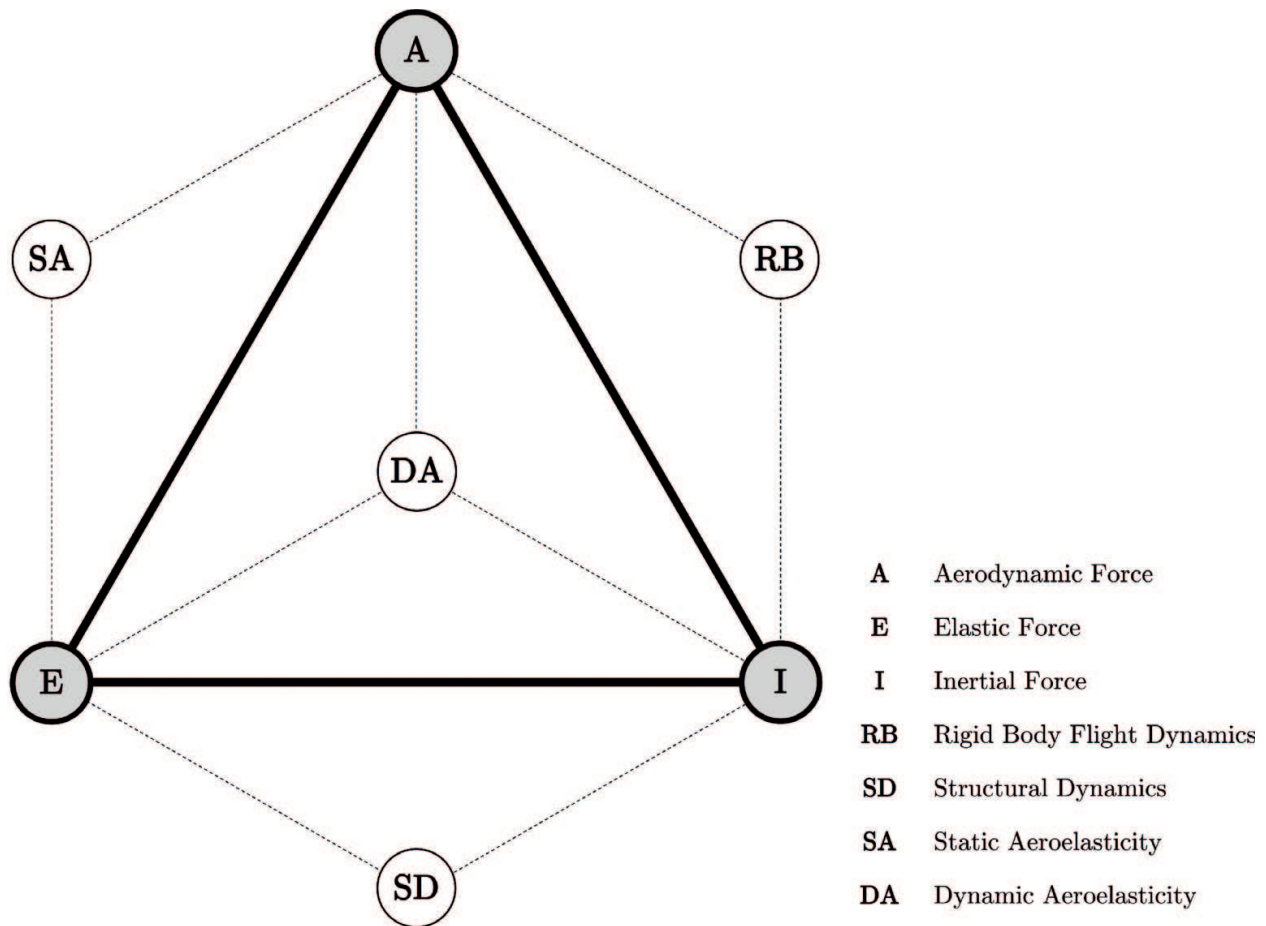
A	Aerodynamic Force
E	Elastic Force
I	Inertial Force
RB	Rigid Body Flight Dynamics
SD	Structural Dynamics
SA	Static Aeroelasticity
DA	Dynamic Aeroelasticity

Figure 1. Extended Collar's triangle.

However, the aeroelastic effects are introduced by the addition of states related to each aeroelastic mode. Assuming that the free vibration modes are available, these make a set of orthogonal functions. The modal representation of the airframe is often obtained through the use of beam element models of the structure and the use of structural analysis software such as NASTRAN. Thus the airframe deformation $e(x,y,z,t)$ can be described in terms of the mode shape $\phi_i(x, y, z)$ and the general displacement coordinate $\eta_i(t)$, as follow:

$$e(x, y, z, t) = \sum_{i=1}^{\infty} \Phi_i(x, y, z)\eta_i(t) \tag{1}$$

The sum of the mode shapes is theoretically infinite but in practice, a finite number of mode shapes are selected in order to investigate the coupling of aeroelastic modes with rigid-body dynamics. The coupling between the rigid-body motion and elastic motion takes place through the forces and moments. The generic force and moment term can be described as function of the inputs (as in the general rigid equations of motion) and the generalised displacement η and its first derivative $\dot{\eta}$, as follow:

$$F = f(u, \alpha, \delta, \ldots, \eta, \dot{\eta}) \tag{2}$$

A new equation is then introduced to account for the elastic dynamics as:

$$\ddot{\eta} + \omega^2 \eta_i = \frac{Q_{\eta i}}{M_i} \tag{3}$$

where $Q_{\eta i}$ and M_i are the generalised force and mass terms, respectively. This formulation allows the application of stability analysis and flight control methods that have been developed based on traditional aircraft models.

Since the work done by Waszak and Schmidt, modelling frameworks of varying complexity have been developed both in industry and academia. Industrial frameworks are highly complex and aimed at supporting certification activities. These often couple Computational Fluid Dynamics (CFD) with Computational Structural Modelling (CSM) and result in processes that provide the desired insight, but at a very high computational cost [10–12]. Much research has been carried out to reduce the computational cost and the effort needed to integrate CFD solvers with CSM packages. However, more often the approach has depended on the specific technical challenge faced by the designer. For example, a few CFD-CSM simulations may be carried out to provide a means of validation for Reduced Order Models (ROMs). The various methods for aerodynamic and structural analysis are summarised in **Figure 2**.

Academic research has shown the capability to link aeroelasticity with flight control and develop novel approaches to aeroservoelastic analysis of highly flexible configurations [13–15].
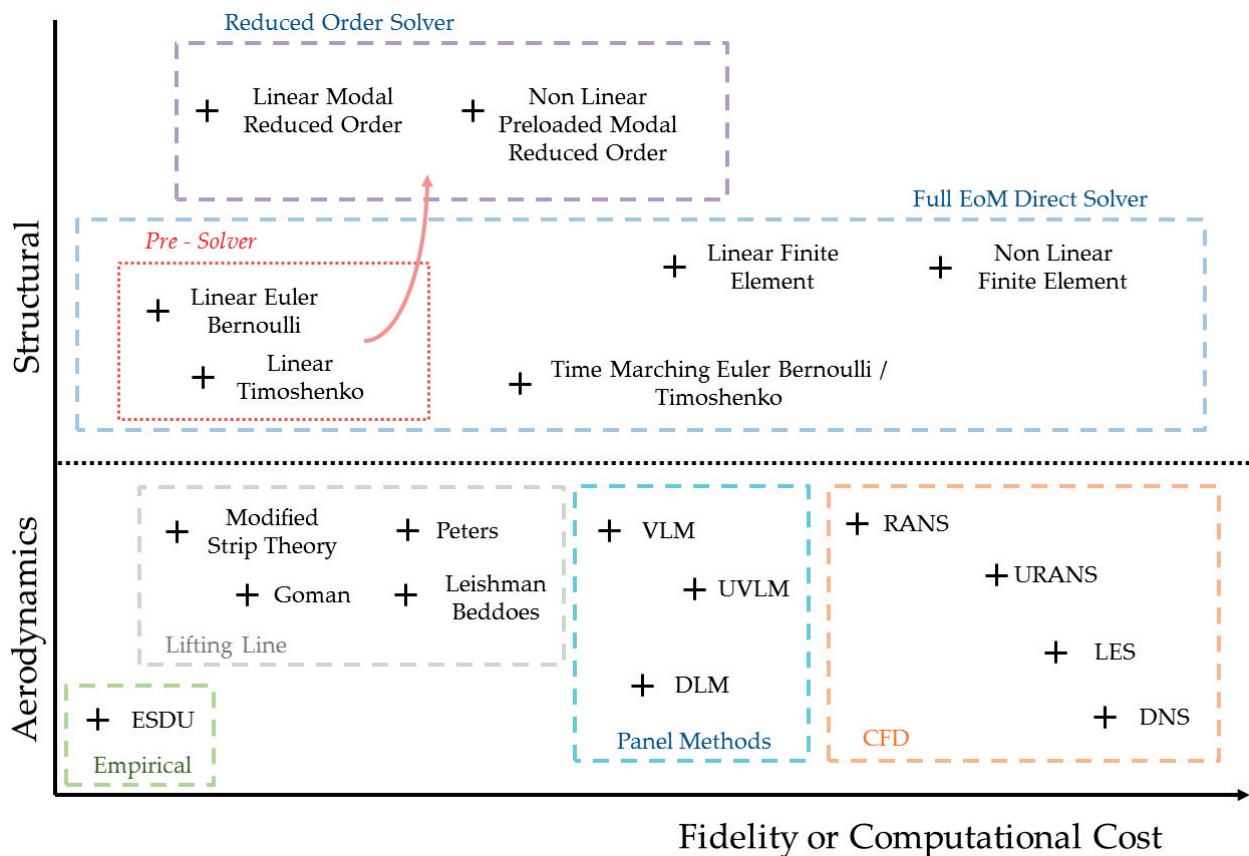

Figure 2. A non-exhaustive list of modelling methods ranked by complexity and fidelity.

Structural flexibility effects have been modelled through the implementation of a nonlinear structural dynamics formulation and aerodynamic contributions have been captured by means of an Unsteady Vortex Lattice Method (UVLM) code. Solving the geometrically-nonlinear beam equations in three different ways, Palacios et al. concluded that the intrinsic beam element model is more efficient regarding the computational time than the classical displacements and rotations based model. It has been shown that for certain geometries the intrinsic model required two times less operations per iteration due to simpler algorithms.

With regards to aerodynamic modelling Palacios et al. [14] showed that an indicial response based on the usual Pade approximation to Wagner's step response performs better at low reduced frequencies than the model based on a Glauert's expansion of the inflow velocity field. Three models—strip theory, strip theory with wingtip effects correction and UVLM—have been compared for different reduced frequencies and wingtip deflections. It has been shown that at low reduced frequency wingtip effects is of high importance both for low and high aspect ratio wings. However, for the case of increased reduced frequencies there has been no agreement of results for low aspect ratio wing. On the other hand, for high aspect ratio wing the agreement between the UVLM and the strip theory without wingtip correction has been shown. Such an agreement has been expected as increasing wing aspect ratio tends to reduce the 3D effect over the wing. The dynamic stall effects have not been modelled in the examples, nevertheless they may be of a great importance for a highly flexible wing. It is important to notice at this point that, if such a dynamic stall model is required by the user, empirical methods are much easier to implement within 2D strip theory than within the UVLM. Palacios and Cesnik [13] included aerofoil deformations in both the structural and the aerodynamic models: A Ritz (finite-section) expansion includes cross-sectional structural deformations, while a Glauert's expansion accounts for deformations of the aerofoil camber line. Integration of both expansions into a single methodology provides a simple alternative to more complex two-dimensional and three-dimensional models for preliminary active aeroelastic analysis of High Aspect Ratio Wings (HARW).

Although the approach adopted by Palacios is computationally cheaper than coupled CFD-CSM, real time simulation is still not possible. The need for real time simulation of flexible aircraft arises from the concern that low frequency aeroelastic modes can potentially couple with rigid-body modes such as the aircraft's short period pitch oscillation and result in poor handling qualities due to unwanted aircraft-pilot coupling [16]. Furthermore, novel concepts for future aircraft, such as those based on blended-wing-body configurations, need detailed stability and control analysis early in the design stage. A real time pilot-in-the-loop simulation environment is therefore needed to identify and solve stability and control problems. The development of such a simulation model requires a trade-off between model fidelity and computational cost.

3. Physics based modular approach

3.1. Aspects of physics based modelling

The case for developing physics based simulation models and the motivation to move away from the classical formulations that rely on stability and control derivatives stems from the

need for flight dynamic insight at the early conceptual design of highly integrated concepts. For such concepts, a database of stability and control derivatives such as Heffley and Jewell [17] does not exist. Moreover, these concepts integrate numerous technologies, such as active folding wingtips for flight and loads control [18] for which empirical methods also do not exist. The modelling and simulation of airframe aerodynamics alone can be complex, but a further layer of complexity is added when considering flexible aircraft for which, the inertial, aerodynamic and structural models need to be coupled. Multiple calculation points, known as structural nodes and aerodynamic panels, must be defined around the airframe and used to capture local flow physics. The structural model must be coupled with the aerodynamics model so that aerodynamic forces and moments acting on the structure modify the effective shape of the aircraft. To complete such an aeroelastic coupling, the updated shape is used to compute the aerodynamic loading for the next iteration.

This additional layer of complexity and iteration process requires a clear definition of methods used when investigating aircraft flight dynamics. These can be broadly divided into two categories:

a. Low fidelity models used in particular for flight simulation and preliminary design studies. These allow for a rapid flight dynamic analysis and may allow parameters to be modified for identifying and quantifying possible optimised solutions.

b. High fidelity computationally expensive models which are used to consolidate the results obtained via low fidelity simulations and help in the investigation of specific problems where low fidelity simulation is not accurate.

For a given problem, multiple approaches can be adopted depending on the needs of the user or the key characteristics of the simulation framework. For example, the structural dynamics of the aircraft can be captured through the integration of a full Finite Element (FE) model with high fidelity, or with a simple beam, or 'stick' model. Within the latter method, multiple sub-layers of complexity can be added depending on the mathematical formulation being used. A direct solving method, which is the most intuitive as it is based on discrete structural loads and nodes, will also be the most laborious and computationally heavy for a high number of structural elements. Alternatively, the modal approach restricted to frequency ranges of interest will be more efficient for linear deformations. In High Altitude Long Endurance (HALE) aircraft or HAR Wing concepts, structural nonlinearities can also become a physical phenomenon that must be captured by the model. Nonlinearities may be relevant only for specific modes and parts of the structure so that optimal solving methods can be identified as well.

Similarly, centre of gravity (CG) position and inertial terms will vary with structural flexibility and displacement. Therefore, acceptable or desired fidelity must be identified. For example, assuming a fixed CG and inertia can lead to significant simplifications in the EoM. However, this may be incorrect for HALE configurations where most of the mass lies in the flexible wing that undergoes large deformations.

Multiple methods to capture the aerodynamic loads acting on the aircraft have also been developed for different levels of fidelity; from simple lifting line theory, use of Engineering Science Data Unit (ESDU) to more complex UVLM and further to more expensive CFD based

processes. The desired accuracy and performance can be optimised depending on the purpose of the framework. Dynamic stall models can also be added for a more accurate simulation of high angle of attack or flow detachment scenarios [19]. CFD simulations are at the higher fidelity end of the spectrum and can be used for construction of the aerodynamic databases [20].

3.2. Modular simulation

The objectives and scope of the problem being considered will undoubtedly dictate which mathematical formulation is selected. For instance, the aerodynamic forces can be calculated using either a Modified Strip Theory (MST) or a UVLM method [21] depending on the fidelity requirements and the available computational power. The structural deflection of the wing can be assumed either linear through an Euler-Bernoulli model or nonlinear with a Timoshenko model [22]. Various atmospheric disturbance models [23] are also implemented so that flight simulations with or without gusts and turbulence are possible for specific gust loads and flight control research. Flight control laws and actuation models of a variety of control surfaces can be used if the user wishes to investigate and develop optimal control or loads alleviation laws. The gravity and navigation model allows for trajectory and autopilot if required. Specialised hardware can be used to accelerate the model and reach real time performances suitable for pilot in the loop simulations at 50 Hz, paving the way for handling quality analysis of flexible aircraft concepts. So far a number of different modelling approaches towards flight dynamics modelling of flexible aircraft have been introduced. This section focuses on the possible problems and issues that emerge when integrating the various elements of such a framework and discusses the need for modularisation.

The basic components required for building a simulation framework are as follows:

1. A structural dynamics model that outputs airframe deformation. This should require forces and moments acting on the structure as inputs, and provide the corresponding displacements, velocities and accelerations as outputs.

2. An aerodynamic model that provides aerodynamic forces and moments as a function of the flight conditions, rigid-body attitudes and structural deformations.

3. An EoM block which uses the total forces and moments acting on the aircraft to compute the vehicle acceleration, velocity, attitude and position in the various reference frames. This will require a clear definition of aircraft mass properties.

4. Atmosphere model that outputs parameters such as Reynolds number required to calculate aerodynamic forces and moments.

5. A gravity model to compute the gravitational forces acting on the aircraft.

6. External atmospheric disturbances based on external velocity fields through which the aircraft is flying. This can be used for carrying out gust/turbulence simulations.

7. Control surface and flight control systems to simulate controlled flight.

Figure 3 illustrates the links between each of the modules and their relative dependencies.
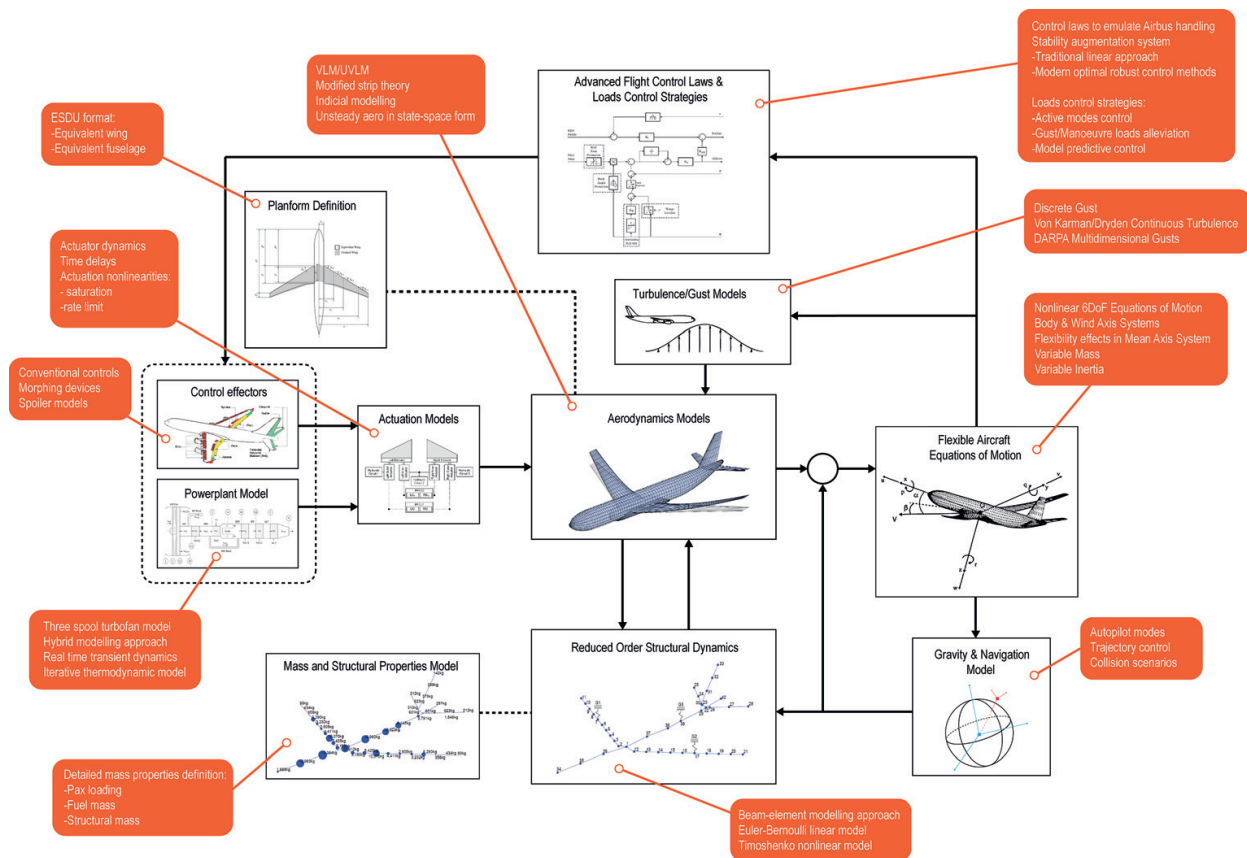
Figure 3. Links between each modules of the simulation framework.

Adopting a modular approach allows for a more versatile framework that can be used to study different configurations and scenarios. Moreover, it allows the adoption of multiple approaches to solve particular mathematical or physical problems. The overhead effort required to develop a modular framework, which primarily takes the form of software engineering, is justified by the end result. If carefully managed a versatile framework that allows solvers and models to be treated in a plug-and-play fashion is achievable. An example of a modular framework is given in **Figure 4**. The CA^{2}LM framework offers the user multiple options in most of the different mathematical models. The modular approach was considered at the early stages of framework development, and has allowed continuous development aiming for a versatile academic research tool.

4. Framework setup for CA^{2}LM

4.1. Wing aerodynamic modelling

There are numerous ways in which wing aerodynamics can be modelled for flexible wings, such as directly via CFD using RANS simulations or steady or unsteady VLM. However, given that there can be thousands of cases that need to be considered for flight loads, computationally cheap alternatives are needed. Within the CA^{2}LM framework, the aerodynamics module
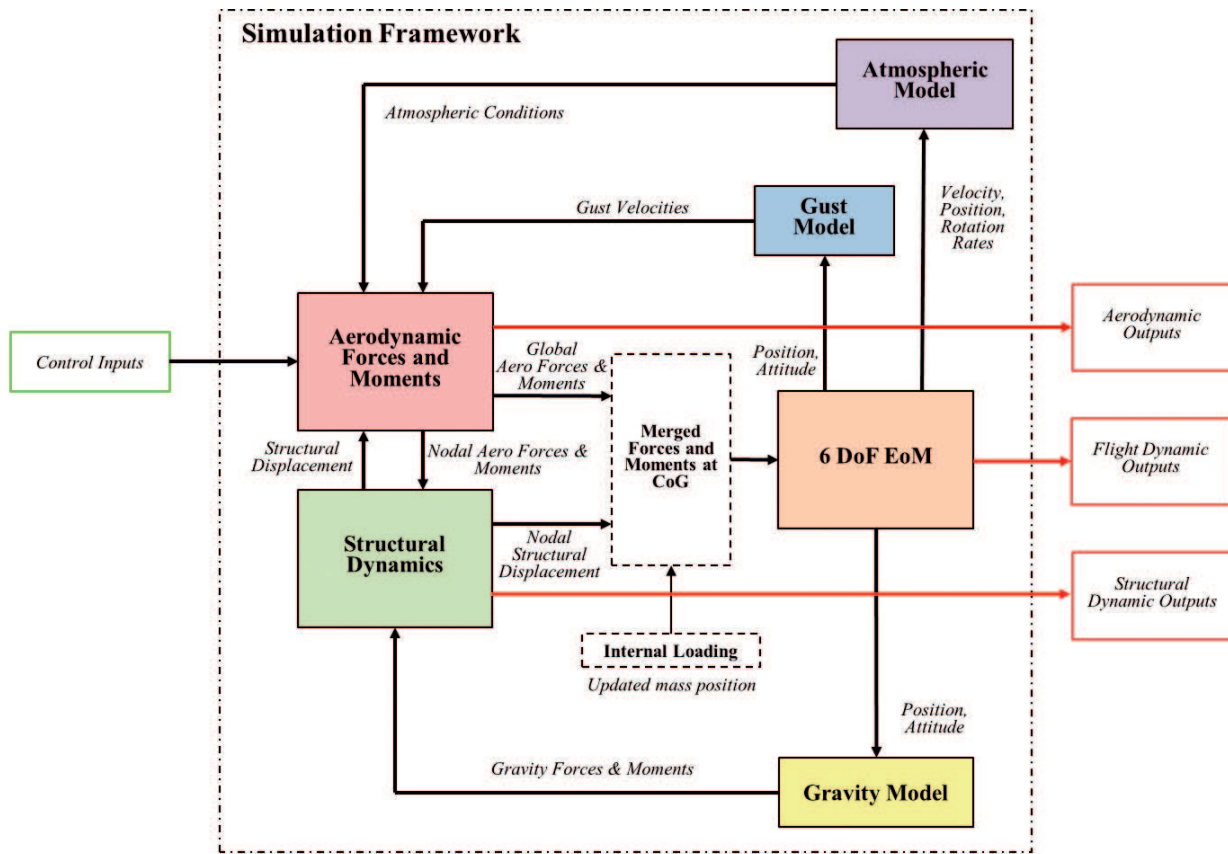
Figure 4. CA^{2}LM framework overall modular architecture.

contains the implementation of the MST based steady aerodynamics coupled with unsteady aerodynamic models [24].

To model the unsteady build-up of lift due to changes in angle of attack and airspeed, a state-space representation of the unsteady aerodynamics of the aerofoil has been implemented following the work done by Leishman and Nguyen [25]. This assumes an arbitrary motion of the aerofoil as combination of the indicial lift response and the superposition principle applying the well-known Duhamel's integral [26]. The following general two-pole approximation of the Wagner function has been adopted in CA^{2}LM:

$$\Phi(\lambda) \approx 1 - A_1 e^{-b_1\lambda} - A_2 e^{-b_2\lambda} \tag{4}$$

where $\lambda = 2V_t/c$ is the relative distance travelled by the aerofoil in terms of semi chords whilst A and b are the indicial response parameters that depend on the boundary conditions. Using the two-pole representation, Leishman and Nguyen developed the lift response to a change in angle of attack $\alpha(t)$ as follow:

$$\begin{bmatrix} \dot{x}_1 \\ \dot{x}_2 \end{bmatrix} = \frac{2V}{c} \begin{bmatrix} -b_1 & 0 \\ 0 & -b_2 \end{bmatrix} \begin{bmatrix} x_1 \\ x_2 \end{bmatrix} + \begin{bmatrix} 1 \\ 1 \end{bmatrix} \alpha(t) \tag{5}$$

and the output equation of the normal force coefficient is given by:

$$C_N(t) = 2\pi \frac{2V}{c} [A_1 b_1 A_2 b_2] \begin{bmatrix} x_1 \\ x_2 \end{bmatrix} \tag{6}$$

Coefficients A_i and b_i have been derived by Leishman in order to obtain the indicial response approximation for a two-dimensional subsonic flow [27]. However, since the Wagner indicial response cannot be applied to compressible flows, a correction introduced by Leishman and Beddoes [28], has been used including the Prandtl-Glauert coefficient $\beta = \sqrt{1 - M^2}$. The full equation of unsteady aerodynamics is then described as:

$$\phi(\tau) \approx 1 - 0.918 e^{-0.366\beta^2} - 0.082 e^{-0.102\beta^2} \tag{7}$$

Increasing the number of poles of the Wagner function allows a closer approximation to be obtained, but at the cost of an increased number of states.

In the CA^{2}LM framework the two-pole representation is used to find lift and pitching moment response with respect to a change in angle of attack α and pitch rate q for each section. The generic total normal force coefficient is given by [29, 30]:

$$C_N(t) = C_N^c(t) + C_{N\alpha}^{nc}(t) + C_{Nq}^{nc}(t) \tag{8}$$

where the superscripts c and nc represent the circulatory and non-circulatory terms respectively. Once aerodynamic characteristics are obtained at each aerodynamic node, the results are extended along the wingspan applying the method defined by DeYoung and Harper [31]. This approach considers the lift line and its trailing vortex as continuous. The circulation strength, however, can be discretized in as many control points as desired. In the CA^{2}LM framework the control points are assumed to be at the aerodynamic nodes. DeYoung and Harper stated that a number of seven control points is enough to correctly represent the span loading without any sharp discontinuities. As the lifting line is discretized in m nodes, the method allows the calculation of the aerodynamic coefficients as follows [29]:

$$c C_{lv} = \sum_{n=1}^{m} A_{vn} G_n \alpha_n, \qquad n = 1, 2, \ldots, m \tag{9}$$

where A_{vn} is the influence matrix which defines the effect of the circulation in the node v to the downwash at node n. The load coefficient G is dimensionless circulation and describes the strength of the circulation at any node n. When the aerodynamic forces and moments at each node are obtained, the loads are transposed from nodal-axis to body-axis and summed to give the overall lift, drag and moment acting on the aircraft structure.

Following the same methodology used for the calculation of the drag, the pitching moment is comprised of circulatory and non-circulatory term, described as follow:

$$C_M = C_{M_\alpha}^c + C_{M_q}^c + C_{M_\alpha}^{nc} + C_{M_q}^{nc} \tag{10}$$

The drag is instead modelled as the sum of the zero-lift drag coefficient, C_{D_0}, and the pressure drag coefficient, C_{D_P}. The unsteady drag force has been defined by Leishman as:

$$C_D = C_{D_0} + C_N \sin \alpha_e(t) - \eta_c C_c \cos \alpha_e(t) \qquad (11)$$

where the effective angle of attack α_e is function of both the states and it is described as:

$$\alpha_e(t) = \beta^2 \frac{2V}{c}(A_1 b_1 x_1 + A_2 b_2 x_2) \qquad (12)$$

and the chord force term is:

$$C_c(t) = \frac{2\pi}{\beta} \alpha_e^2(t) \qquad (13)$$

As a real flow is unable to be fully attached in any real flow, the coefficient η_c is used to account for the properties of the real flow.

4.2. Structural modelling

Now all aerodynamic forces have to be applied to the structures of the aircraft. This is done in the structural dynamics modelling block.

Aerodynamic forces and moments, along with forces and moments due to gravity, are converted to modal forces F through modal transformation matrix $\boldsymbol{\Theta}_m^T$:

$$F_i = \boldsymbol{\Theta}_m^T \boldsymbol{F}_{aero} \qquad (14)$$

The next step is to solve the following structural equation of motion:

$$\frac{F_i}{m_i} = \ddot{x}_i + 2\zeta\omega_n\dot{x}_i + \omega_n^2 x_i \qquad (15)$$

where F_i represents the modal forces, m_i the modal masses, $\omega_{n,i}$ the modal natural frequencies, ζ the modal damping ratios, i is the modes number, $x_i, \dot{x}_i, \ddot{x}_i$ are the modal displacements, velocities and accelerations. To obtain the structural dynamics in modal form, the Normal Modes analysis solver SOL 103 from the NASTRAN finite element analysis program is used. Its output (modal masses, natural frequencies and modal transformation matrix) are used in the CA^{2}LM framework to calculate structural deflections. The displacements, velocities and accelerations of each structural node can then be obtained using the transformation matrix.

As these deflections, velocities and accelerations are applied to aerodynamic frame, the interpolation between structural and aerodynamic nodes is executed.

The first 12 structural modes are considered in the CA^{2}LM framework because the tool is designed to investigate interactions between aeroelasticity effects and flight dynamics phenomena that are typically at low frequencies. An illustration of an aircraft first four modes is given in **Figure 5**.

It is important to note that only small wingtip deflections (less than 10% of a wing semi-span) are modelled within CA^{2}LM framework as linearly varying beam properties are assumed. However, recent developments in highly flexible aircraft [32] have introduced wingtip
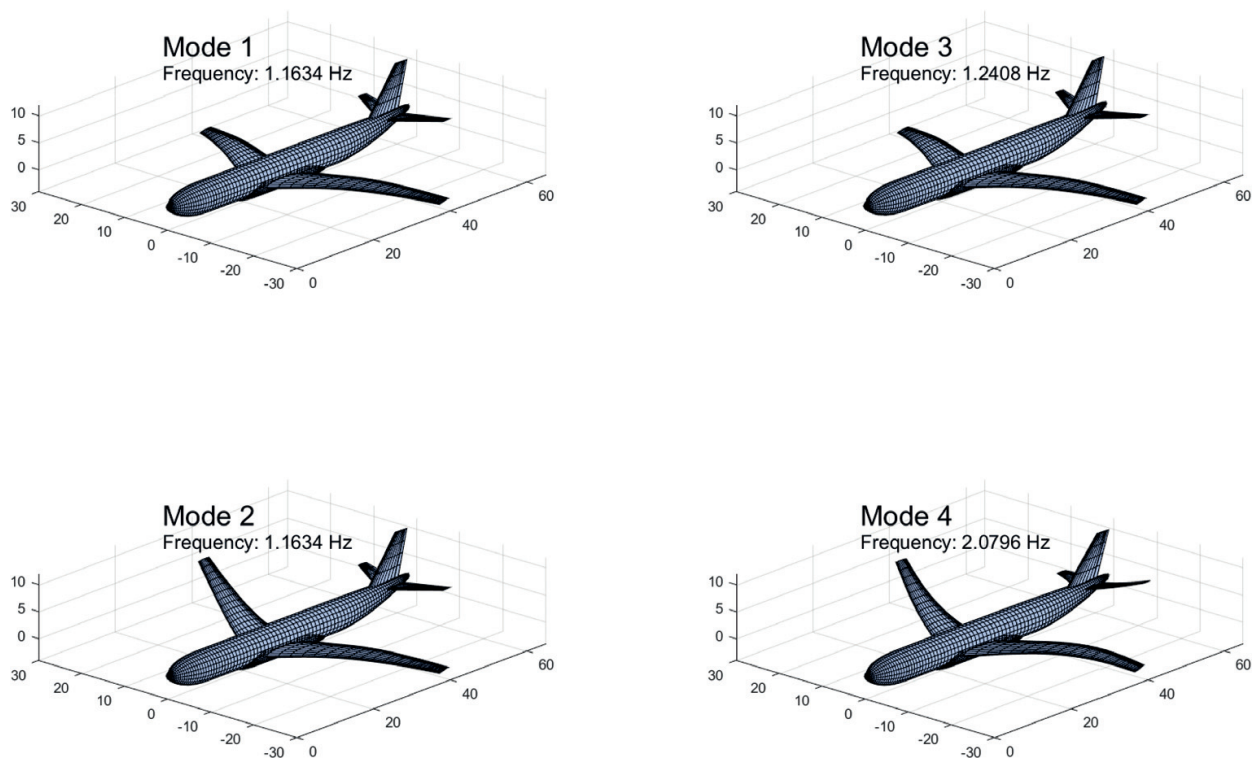
Figure 5. First four modes of the AX-1 aircraft implemented in CA²LM.

deflections of more than 25% of a wing semi-span. To investigate the effects of such high structural deformations on flight dynamics, a structural dynamics model capable of capturing the nonlinear phenomena due to large deformations is needed.

4.3. Equations of motion

For large flexible aircraft, the centre of gravity (CG) position may vary significantly as a function of structural deformation. This is typically ignored in the classical EoM formulation for rigid aircraft [1, 2]. This issue together with continuously deforming aerodynamic and structural stations requires the careful definition of the axes systems for each module of the simulation framework. The selection of an appropriate axes system has been extensively discussed for many years [8, 33, 34]. Effectively there are two approaches that may be adopted: (1) use an arbitrary point on the aircraft also called the body axes centre (BAC) or, (2) adopt the mean axes system which has a floating point as the reference centre [35]. The latter has seen widespread application in research [9, 36] because its formulation minimises the coupling between rigid-body dynamics and aeroelastic modes. On the other hand, the axes system centre is often collocated with the CG which moves in phase with the flexible airframe, making the application of traditional flight dynamics analysis techniques more difficult. The idea of the mean axes system's inertial decoupling and complexity of its formulation has been questioned [34].

The CA²LM framework uses a fixed BAC as a reference centre for its flight dynamic axis system. This allows the framework to be used in both flexible and rigid modes and more importantly, it allows the integration of classical flight dynamics post-processing tools.
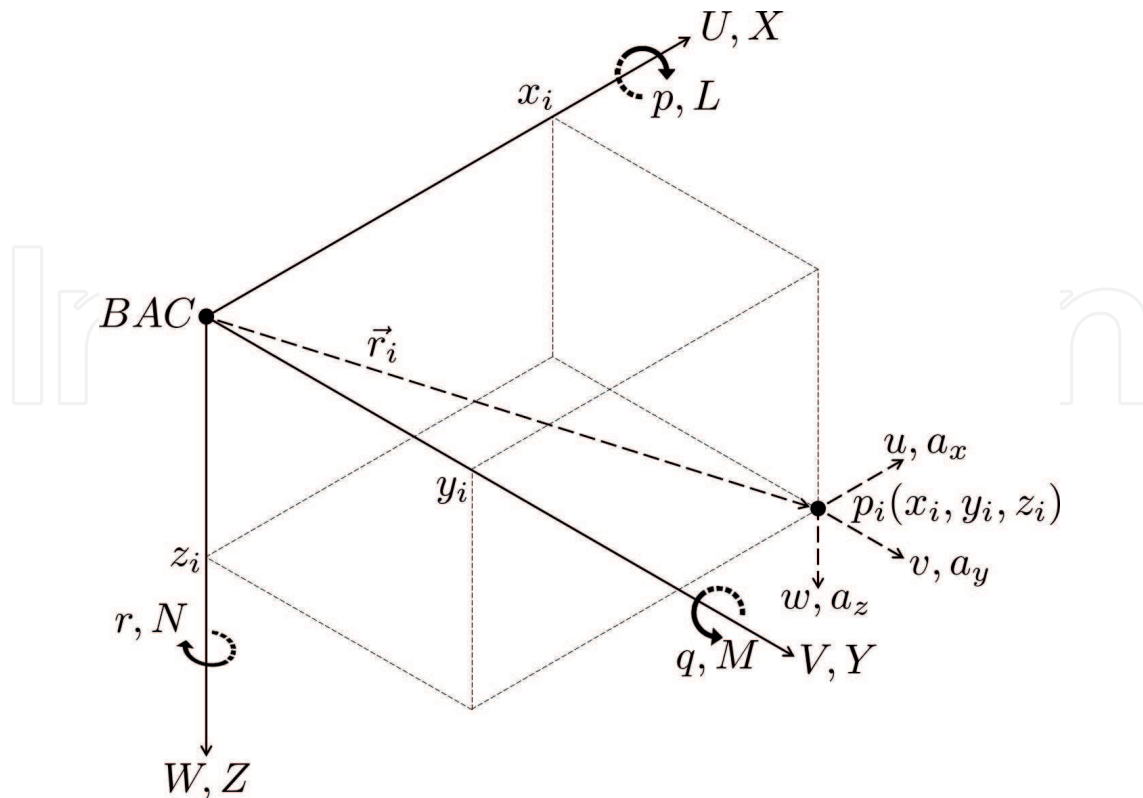
Figure 6. Motion of a body and its particle within the frame.

The derivation of the EoM begins by considering a fixed node which is located away from the BAC, as shown in **Figure 6**. The velocities of this point can be expressed as:

$$\begin{bmatrix} u \\ v \\ w \end{bmatrix} = \begin{bmatrix} U + \dot{x} \\ V + \dot{y} \\ W + \dot{z} \end{bmatrix} - \begin{bmatrix} x \\ y \\ z \end{bmatrix} \times \begin{bmatrix} p \\ q \\ r \end{bmatrix} \tag{16}$$

And therefore, the following accelerations can be obtained:

$$\begin{bmatrix} a_x \\ a_y \\ a_z \end{bmatrix} = \begin{bmatrix} \dot{u} \\ \dot{v} \\ \dot{w} \end{bmatrix} - \begin{bmatrix} u \\ v \\ w \end{bmatrix} \times \begin{bmatrix} p \\ q \\ r \end{bmatrix} \tag{17}$$

The velocities U, V and W express the motion of the BAC, while x, y and z express the position of the node. The angular rates p, q and r represent the angular velocities of the overall aircraft. Merging both equations gives following accelerations expressions:

$$a_x = \dot{U} - rV + qW - x(q^2 + r^2) + y(pq - \dot{r}) + z(pr + \dot{q}) + \ddot{x} - 2r\dot{y} + 2q\dot{z} \tag{18}$$

$$a_y = \dot{V} - pW + rU + x(pq + \dot{r}) - y(p^2 + r^2) + z(qr - \dot{p}) + \ddot{y} - 2p\dot{z} + 2r\dot{x} \tag{19}$$

$$a_z = \dot{W} - qU + pV + x(pr - \dot{q}) + y(qr + \dot{p}) - z(p^2 + q^2) + \ddot{z} - 2q\dot{x} + 2p\dot{y} \tag{20}$$

Now applying Newton's second law with a nodal mass of δm the EoM can be obtained as follows:

$$\begin{bmatrix} X \\ Y \\ Z \end{bmatrix} = \sum_{i=1}^{N} \delta m_i \begin{pmatrix} a_x \\ a_y \\ a_z \end{pmatrix} = \overbrace{\sum_{i=1}^{N} \delta m_i \dot{v}_0 + \sum_{i=1}^{N} \delta m_i \omega \times v_0}^{\text{Rigid body dynamics force}}$$

$$\overbrace{+ \underbrace{\sum_{i=1}^{N} \delta m_i \omega \times (\omega \times r_i)}_{\text{Centrifugal force}} + \underbrace{\sum_{i=1}^{N} \delta m_i \dot{\omega} \times r_i}_{\text{Euler force}}}^{\text{Axes reference point offset}}$$

$$+ \underbrace{\sum_{i=1}^{N} \delta m_i a_{rel,i}}_{\text{Inertial force}} + \underbrace{2 \sum_{i=1}^{N} \delta m_i \omega \times v_{rel,i}}_{\text{Coriolis force}}$$

$$\underbrace{\hphantom{+ \sum_{i=1}^{N} \delta m_i a_{rel,i} + 2 \sum_{i=1}^{N} \delta m_i \omega \times v_{rel,i}}}_{\text{Flexibility effects}} \tag{21}$$

$$\begin{bmatrix} L \\ M \\ N \end{bmatrix} = \overbrace{I\dot{\omega} + \omega \times (I\omega)}^{\text{Rigid body dynamics}} + \overbrace{\sum_{i=1}^{N} \delta m_i r_i \times \dot{v}_0 + \sum_{i=1}^{N} \delta m_i r_i \times (\omega \times v_0)}^{\text{Axes reference point offset}}$$

$$\underbrace{+ \dot{I}\omega + \omega \times \sum_{i=1}^{N} \delta m_i (r_i \times v_{rel,i}) + \sum_{i=1}^{N} \delta m_i r_i \times a_{rel,i}}_{\text{Flexibility effects}} \tag{22}$$

The forces and moments on the left hand side of the above equations are the sum of the forces and moments obtained from the structural dynamics, aerodynamics and gravitational modules.

4.4. Aeroelastic coupling and equations of motion integration

The previous sections have shown that each module within the simulation framework requires the definition of its own axis system and a separate means of modelling the aircraft, whether it is through a set of structural nodes or aerodynamic panels. This presents two issues that must be addressed before scenarios can be simulated: (1) node and panel distributions and densities need to be optimised based on the scope of the research and, (2) the structural nodes must be linked to aerodynamic nodes.

As seen in the previous section, the structural loads calculations rely on a set of structural nodes. Displacements, velocities and accelerations of each node are calculated in all 6 degrees of freedom.[1]

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[1]It is possible to constrain specific degrees of freedom to reduce model complexity after a comparison study with the 6 DoF model. For stiff wings, structural rotation around the vertical axis can be neglected for example.
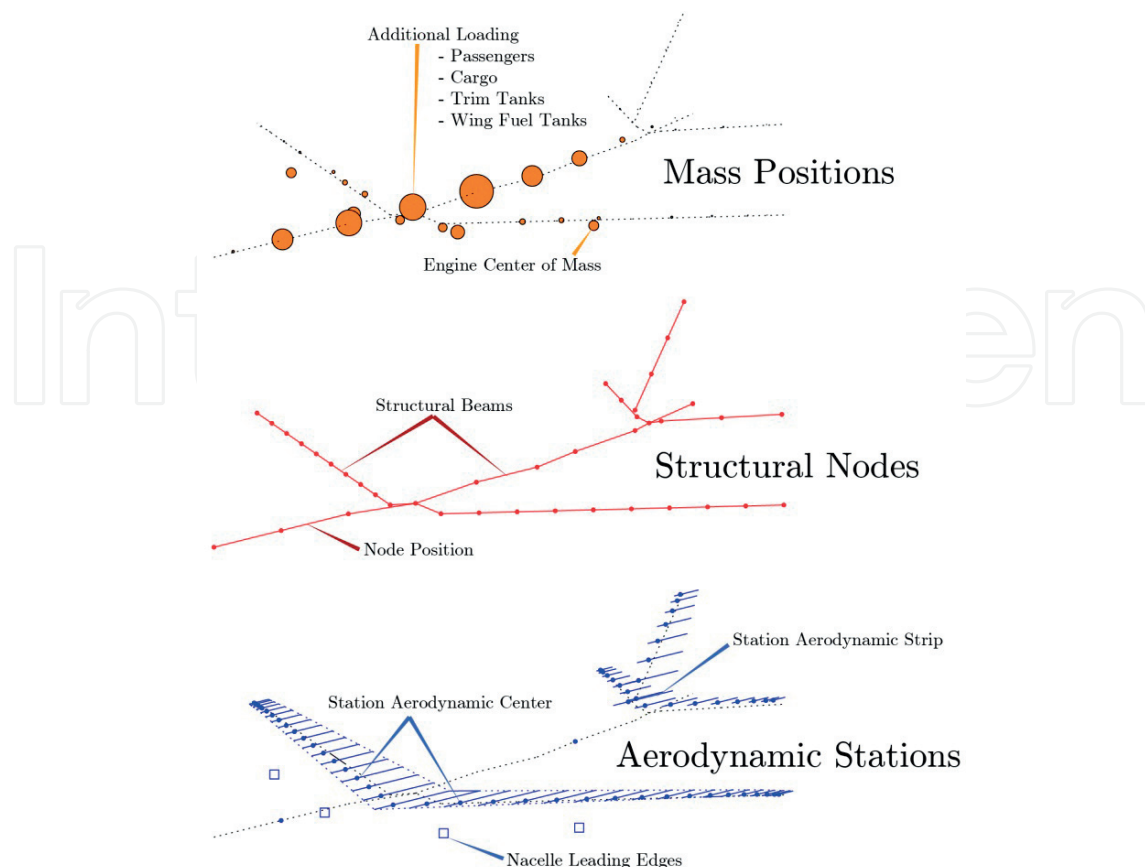
Figure 7. Illustration of the different mass, structural node and aerodynamic station positions for the AX-1 aircraft.

Appropriate balance between accuracy and computational cost must be obtained using a convergence study to identify the optimal number of structural nodes and aerodynamic panels or strips. This number can vary with aircraft configuration and the type of flight dynamics being considered. However, the number of structural nodes may be different from the optimal number of aerodynamic stations. A modular simulation environment such as CA^{2}LM allows the definition of different numbers of aerodynamic strips and structural nodes. The aerodynamic forces and moments calculated at the aerodynamic stations must then be transferred to the structural set of nodes using various interpolation methods. Similarly the structural displacements, velocities and accelerations calculated from the structural model must be transferred to the aerodynamic stations in order to calculate the local forces and moments with structural flexibility. An example of this coupling can be found in **Figure 7** where both the structural node and aerodynamic station layout is illustrated for an example aircraft.

The EoM rely on the aircraft total forces and moments, acting around the centre of gravity of the vehicle. Therefore, the updated CG position due to structural deformation must be used to calculate the new global set of moments acting on the aircraft. Aerodynamic loading calculated at each aerodynamic station is merged and calculated at the temporary CG position. Only then can the coupling between the aerodynamic and structural block be made with the EoM.

The output of the EoM such as aircraft position, attitude and velocity can then be used by conventional atmospheric models to compute the dynamic pressure and other aerodynamic
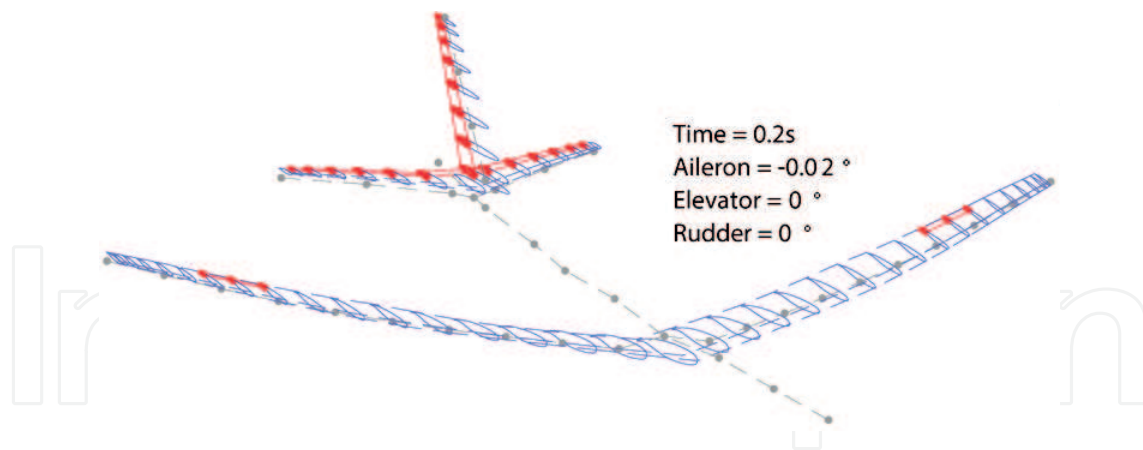
Figure 8. Aircraft flexible structure overlaid with aerodynamic profiles and control surfaces for pilot input visualisation.

parameters used by the aerodynamic model, closing the main calculation loop. Similarly, the adequate gravity contribution can be computed with position (or altitude) and applied to the structural model.

Appropriate inputs, usually on aircraft control surface and thrust, should be linked to the model in the correct format. Control surface dynamics can be implemented for higher fidelity.

As each module is included in the simulation framework, correct integration testing must be conducted to verify that each modules are behaving as expected. Therefore, as the complexity of the framework increases, thorough testing also requires more effort. It can also be really helpful to have visual aids and illustrations of the simulation. For example, an illustration of aerodynamic station and structural node positions updated with structural flexibility at each time step can be found in **Figure 8** and is very useful to visualise the modelled aircraft.

5. Framework test cases

5.1. Multidimensional discrete gust loads simulation

The aim of this test case is to demonstrate the use of simulation frameworks such as CA^2LM for assessing the impact of multidimensional discrete gust modelling on conventional gust loads practices seen in industry. The prediction and control of aircraft gust loads is a key step in aircraft design development and certification. The methodology to model realistic discrete and continuous atmospheric disturbances has been derived based on many years of flight testing and operational data [37]. Hoblit [23] covers a concise but thorough overview of the historical development of gust and turbulence modelling in whereas a detailed discussion of current industry practices can be found in [35]. However, the methods to date simplify the process of calculating gust loads by neglecting spanwise variations in the gust/turbulence fields. This case study demonstrates the application of the CA^2LM framework for studying gust profiles that have spanwise variations. Atmospheric disturbances are usually added through the use of velocity fields. For each aerodynamic station, the wind or gust velocities can be added to the rigid-body translation, rotation and elastic structural dynamics in a local

nodal axis system to compute local changes to angle of attack and flow velocities. If gusts are defined as a velocity field, the gust model should also use the aerodynamic station layout and aircraft attitude to apply a penetration effect.

With the development of HALE UAV aircraft, the lack of spanwise non-uniform velocity distributions was identified as critical both for realistic and theoretical modelling purposes. The gust profiles specified in certification requirements [37, 38] implicitly assume that a uniform velocity distribution causes the highest internal loads and therefore, are the only cases that need to be investigated. Therefore, Defense Advanced Research Projects Agency (DARPA) focused on the derivation of a modified discrete gust model to account for the extra dimensional term and led to the expression of the discrete gust velocity V_{dg} to be defined by:

$$V_{dg}(x_d, y_d) = V_{do}f_x(x_d, H_x)f_y(y_d, H_y) \tag{23}$$

where:

$$f_x = \frac{1}{2}\left(1 - \cos\left(\frac{\pi x_d}{H_x}\right)\right) \tag{24}$$

and f_y is the corresponding sinusoidal function. V_{do} is the gust intensity, H_x and H_y are the longitudinal and lateral gust gradients respectively and x_d and y_d are the longitudinal and lateral positions of the interest point in the discrete gust reference frame. Specifications to the range of both gust gradients can be made using similar hypothesis as before, ranging from 9 to 107 m.[2] An illustration of the multidimensional discrete gust velocity field is given in **Figures 9** and **10**.

This type of model was implemented as a feature within the CA^{2}LM framework and applied to a conventional long range flexible aircraft configuration known as the AX-1. A study investigating the impact of such an approach to gust loads prediction for conventional aircraft was then carried out [39] using a sinusoidal lateral distribution as follows:
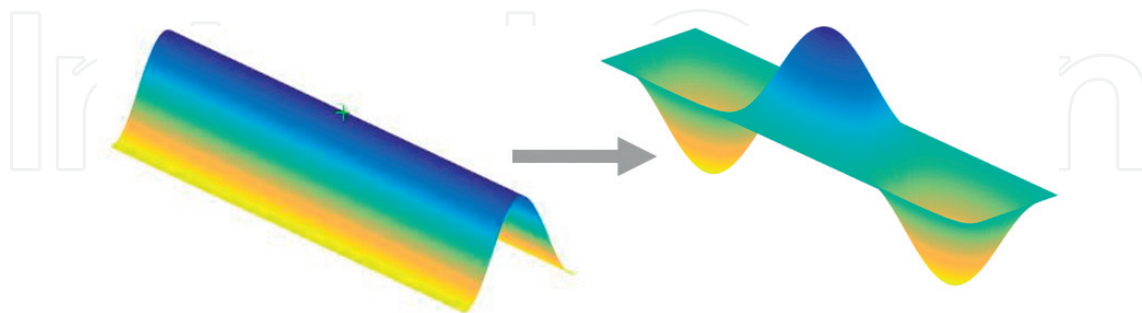

Figure 9. From a 1D to 2D discrete gust definition using coupled sinusoidal variation functions.

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[2]In fact, it is necessary to push the higher end of the gradient spectrum so as to reach a minimum of 12.5 times the maximum aerodynamic chord of the vehicle and/or reach the peak maximum of the evaluated quantity with respect to the various conditions.
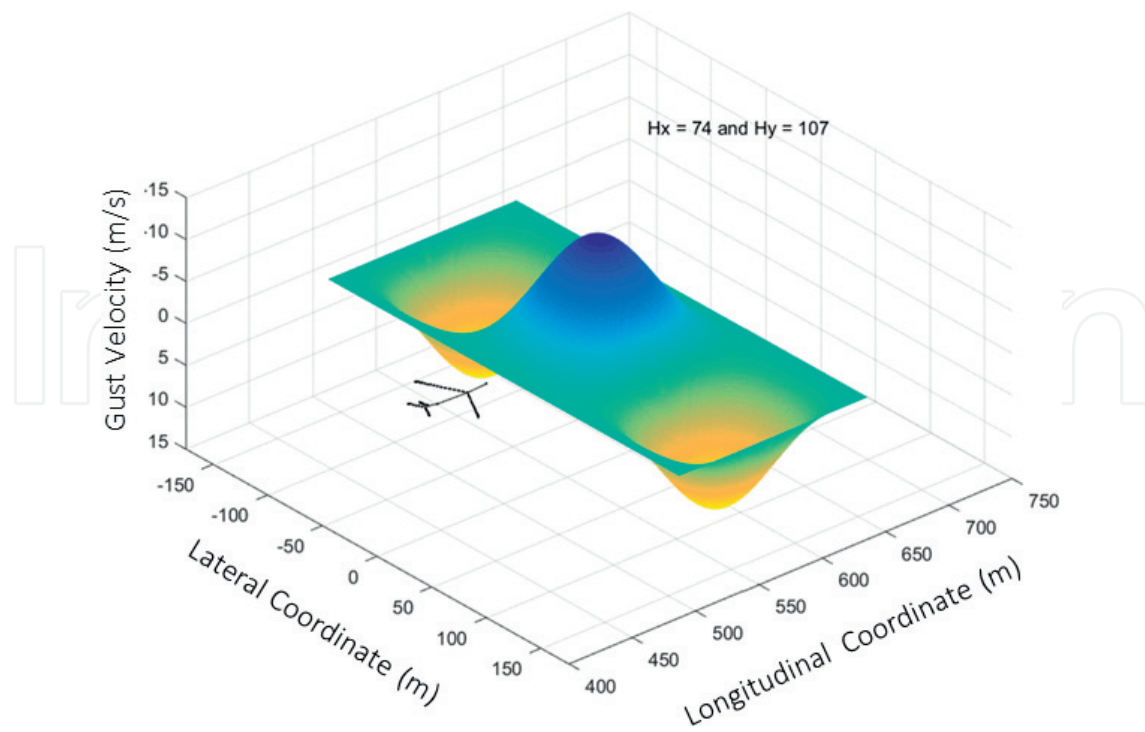
Figure 10. Visual display of the discrete gust velocity field for a given set of gust gradients used in the loads prediction loop.

$$f_y = \cos\left(\frac{\pi y_d}{H_y}\right) \tag{25}$$

A sufficiently large number of realistic flight points compatible with the framework and implemented aircraft were used for this study. A number of gust gradients were used to allow a comparison between the conventional spanwise uniform velocity field and the multi-dimensional model of interest with enough fidelity. All simulations were made in an open loop system, where no correction to aircraft attitude is made. Two different approaches were used to scale the maximum gust intensity, keeping the core hypothesis of the certification requirements. This is justified by the very nature of the derivation of the original model, based on flight testing and loads data and not actual mapping of the gust velocity fields.

In both cases, the use of a multidimensional model led to lower gust structural wing root loads and vertical loads for an equivalent longitudinal gust gradient, as illustrated in **Figure 11**. In one case of velocity tuning methodology, some local loads extrema were higher than with the conventional model, possibly leading to higher occurrence numbers of specific load values. This also came to a cost in computation time, increasing by an order varying with H_y discretisation size the number of simulations required for a complete gust loads loop process.

Overall, these results were to be expected with the chosen spanwise distribution. Maximal gust intensity was centred on the fuselage in this study. But these results can vary quite dramatically with the selected f_y distribution. If focused on matching the vertical load factor whilst keeping wingtip loads to the highest, this could lead to:
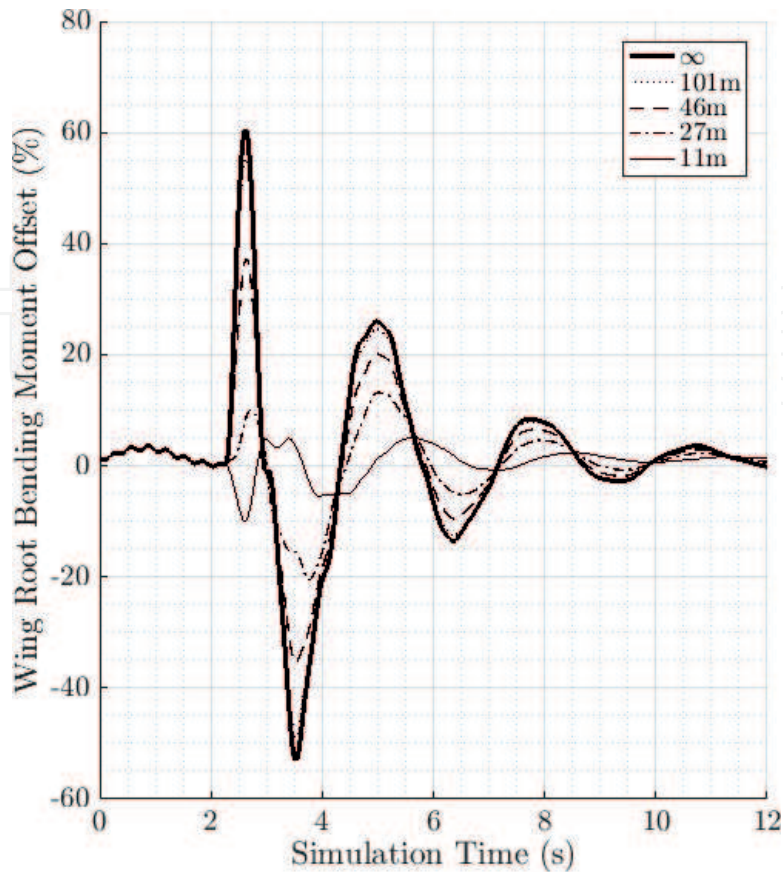
Figure 11. Time histories of wing root bending offset relative to trim for a given H_x and various H_y gradients.

- A 'realistic' model of the gust velocity field compliant with the historical development of the methodology based on vertical load and angle of attack data recordings.

- Higher wing root structural loads due to increased wingtip loading.

5.2. Aileron failure simulations

A control surface failure scenario is one of many failure cases that need to be considered for flight loads evaluation. Here the CA²LM framework is used for simulating a soft aileron failure where the port aileron undergoes an actuation failure and is forced to undergo a 15° amplitude limit cycle oscillation (LCO) whilst starboard aileron remains in the original trim setting. The dynamics of the aileron actuators are modelled through the transfer function:

$$\delta_a(s) = \frac{-1.77s + 399}{s^2 + 48.2s + 399}$$

(26)

The main results obtained from the simulation of the AX-1 model are shown in **Figure 12**. The port aileron moves under a limit cycle oscillation at a constant frequency of 1.16 Hz, which corresponds to the first wing structural bending mode. The amplitude of this oscillation is set to $\pm 15^\circ$.
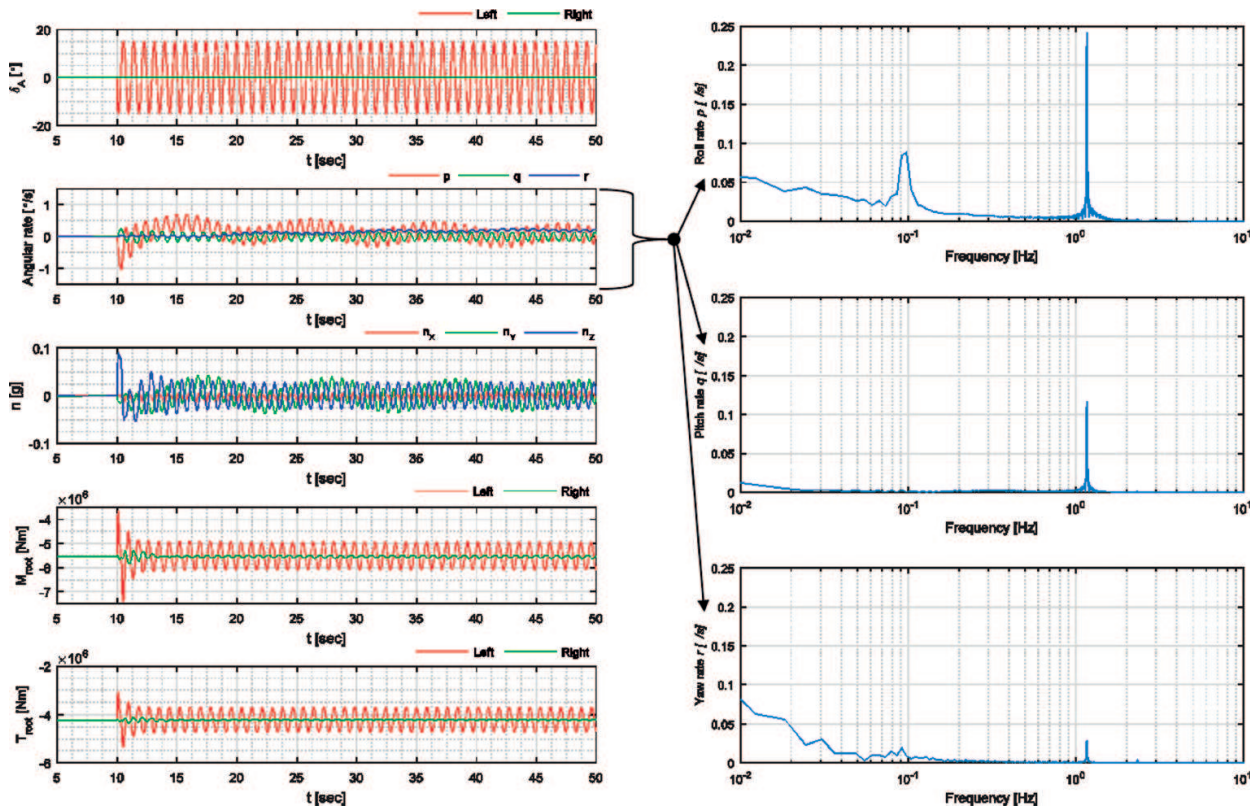
Figure 12. Example of AX-1 aileron cycle oscillation failure simulation results.

The frequency content of the roll rate p and yaw rate r signals show that the failure has excited a low frequency lateral-directional mode corresponding to periods of $T_p = 10.24$ s and $T_r = 10.92$ s in roll and yaw respectively. These correspond to the usual frequencies of the aircraft's Dutch roll mode. The highest peaks, just above 1 Hz, are the direct result of the simulated aileron forcing function. The load factor (n) only exhibits large transients when the aileron failure is initiated.

Figure 13 shows the frequency content of the wing root bending moment M_{root} at different aileron excitation frequencies. At a frequency of 1.245 Hz, slightly higher than the frequency of the first structural mode of the wing (1.1634 Hz), the first aeroelastic mode appears and a resulting resonance is observed. Upon magnification (bottom right subfigure) another two peaks can be observed at 2.5 and 3 Hz. These correspond to aeroelastic modes associated with the 5th and 11th aircraft structural modes. At the frequency of 0.9 Hz, M_{root} is higher than at the frequency of 1.1 Hz, which can be explained by the fact that the forcing function frequency is getting closer to rigid-body frequencies.

Simulations like this provide the insight loads engineers and flight control engineers need for exploring scenarios where a novel solution could be tested and design improvements can be made. Simulation frameworks such as CA^{2}LM provide a rapid simulation capability needed especially at low technology readiness levels, where engineers and designers are interested in the impact of novel technologies such as folding wingtips, possible aircraft-pilot coupling scenarios [40] and flight loads during collision avoidance [6].
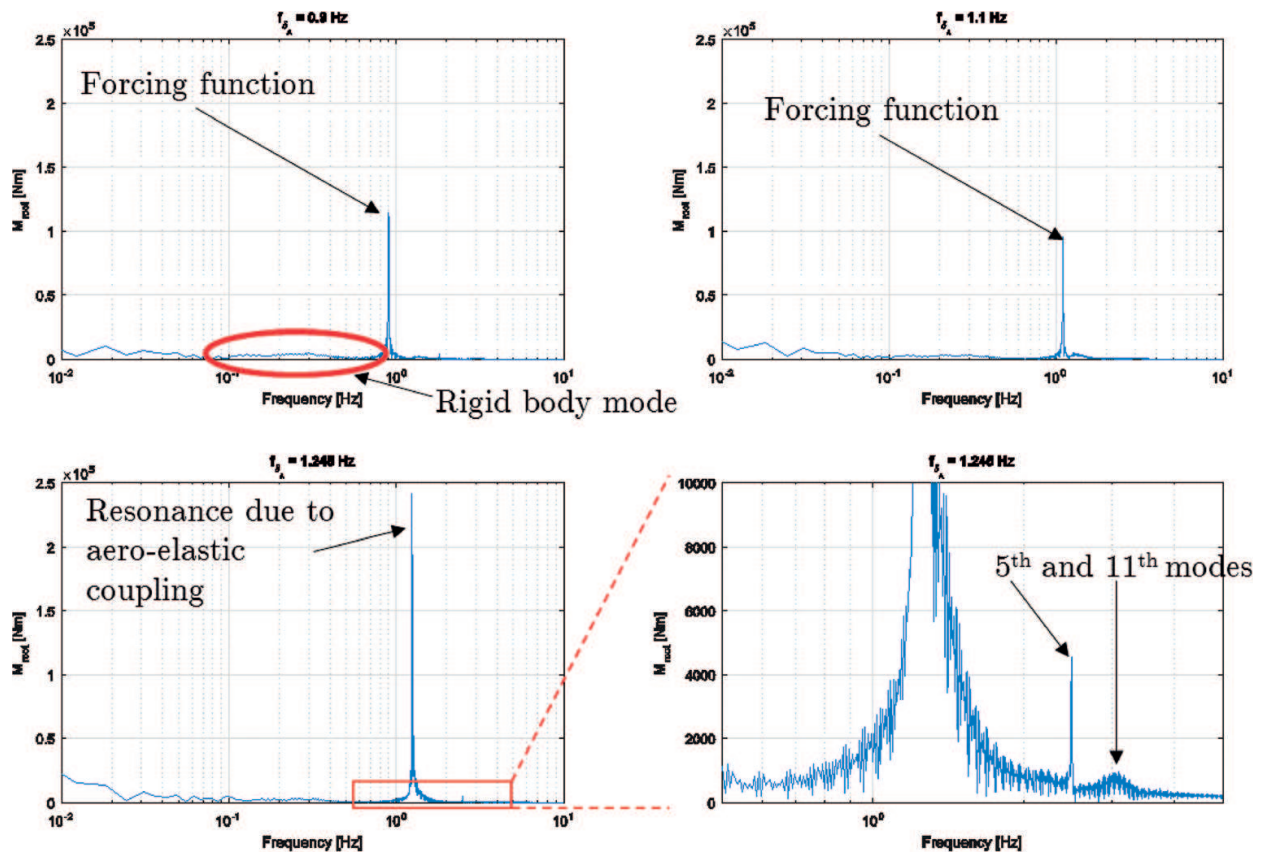
Figure 13. Wing root bending moment frequency spectrum for different aileron excitation.

6. Conclusions

Technologically innovative and highly integrated concepts are being considered in response to increasing aircraft efficiency and reducing the environmental impact of aviation. The development of these concepts has highlighted the need for modular low fidelity aircraft simulation frameworks at the conceptual design stage that are capable of predicting the flight dynamics, flight loads and aeroservoelastic characteristics. This chapter has presented the key aspects of developing such a framework and the need for a modular physics based approach. This approach requires a careful integration of aerodynamic models with models for structural dynamics and then both need to be coupled with the flight dynamic equations of motion. It has been shown that the aerodynamic representation must include a combination of unsteady and steady aerodynamic models implemented through aerodynamic panels. These panels need to then be linked to the aircraft structure which is typically implemented as a series of nodes and beams. The coupled aero-structural model then needs to provide forces and moments to the equations of motion. The details of developing such a simulation framework has been presented in this chapter and the utility of such a tool is illustrated through two test cases. The first case focuses on aircraft response to a gust that has a spanwise varying profile. The second investigates aircraft dynamics during control surface failure scenarios. The Cranfield Accelerated Aeroplane Loads Model (CA^{2}LM) forms the basis of the presented discussion.

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