Studying aeroelastic behavior of aircraft with NeoCASS.
The Danbus configuration

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Abstract

This Master Thesis assesses the performance of NeoCASS software as a tool for structural sizing for deformable aircraft at conceptual design level. NeoCASS is a collection of Matlab modules included in CEASIOM. Some test cases (simple wing model, TCR, Danbus, Agard 445.6 wing) are computed to evaluate the aeroelastic behavior and provide additional validation on the structural models NeoCASS computes. Finally, Danbus, a new conceptual design aircraft, is analyzed with the software.
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Nomenclature

$\alpha$  Angle of attack
$\beta$  Sideslip angle
$\delta$  Control deflection
$\delta_a$  Aileron deflection
$\delta_c$  Canard deflection
$\delta_e$  Elevator deflection
$\delta_r$  Rudder deflection
$\eta$  Modal coordinates of the system
$\hat{A}$  Generalized aerodynamic matrix
$\hat{v}$  Laplace transform of $v$
$\dot{\rho}$  $pb/u$
$\Omega$  $Z^T K Z$
$M$  Pitch moment
$L$  Roll moment
$\omega$  Natural vibration frequencies of the system
$\rho$  Density
$C_{Tv}^r$  Elastic deflection roll moment coefficient matrix
$f_0$  Initial external forces
$K$  The stiffness matrix
$M$  Mass matrix
$Q$  Aerodynamic transfer matrix
\( Q_0 \) \hspace{1em} Initial aerodynamic transfer matrix
\( Q_\delta \) \hspace{1em} Aerodynamic transfer matrix for the control deflection
\( v \) \hspace{1em} Elastic deformation of the structure
\( Z \) \hspace{1em} Reference change matrix from the original to the modal coordinates
\( z \) \hspace{1em} Eigenvectors of the vibration eigenvalue problem
\( a_w \) \hspace{1em} \( a_w = C_L/\alpha \)
\( b \) \hspace{1em} Aerodynamic reference length
\( c \) \hspace{1em} Reference chord
\( C_{L\delta} \) \hspace{1em} Roll moment coefficient for control deflection
\( C_L \) \hspace{1em} Roll moment coefficient
\( C_{L\alpha} \) \hspace{1em} Lift coefficient derivative for the angle of attack
\( C_{L\delta a} \) \hspace{1em} Lift coefficient derivative for the aileron deflection
\( C_{L\delta c} \) \hspace{1em} Lift coefficient derivative for canard deflection
\( C_{Lq} \) \hspace{1em} Lift coefficient derivative for the pitch angular velocity
\( C_{S\delta a} \) \hspace{1em} Lateral force coefficient derivative for aileron deflection
\( C_{S\delta r} \) \hspace{1em} Lateral force coefficient derivative for rudder deflection
\( C_{Sp} \) \hspace{1em} Lateral force coefficient for the yaw angular velocity
\( C_{L\delta a} \) \hspace{1em} Roll moment coefficient derivative for the aileron deflection
\( C_{L\delta p} \) \hspace{1em} Roll moment coefficient for the yaw angular velocity
\( C_{M\alpha} \) \hspace{1em} Pitch moment coefficient derivative for the angle of attack
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\( C_{Np} \) \hspace{1em} Yaw moment coefficient for the yaw angular velocity
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\( f_{a\theta} \) \hspace{1em} Aerodynamic contributions to the external forces
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Chapter 1

Introduction

NeoCASS (Next Generation Conceptual Aero-Structural Sizing Suite) is a collection of Matlab modules focused on initial aircraft structural sizing regarding static, modal and aeroelastic analyses. The suite is a part of CEASIO (Computerised Environment for Aircraft Synthesis and Integrated Optimisation Methods [3]), a simulation environment for conceptual and preliminary design.

The aim of NeoCASS is to enhance the structural sizing at early design phases. The simplified methods used in conceptual design no longer provide sufficient reliability for modern, flexible aircraft. The aircraft structure is not taken into account until the preliminary design stage and the weight is evaluated using statistical-based methods, thus resulting in bad weight predictions for newer designs and ignoring the aeroelastic requirements altogether. Therefore, the main goals of NeoCASS are to provide a more realistic estimation of structural weight and to allow for aeroelastic analysis and optimization at the conceptual design level.
Chapter 2

Structural sizing in conceptual design

Structural sizing in conceptual design can be approached from two different perspectives. On the one hand, an analytical sizing based on the initial definition of the aircraft can be performed to have a rough estimate of the airframe. On the other hand, a more detailed sizing can be achieved by using computer techniques that require more data about the aircraft but provide a more accurate estimation of the final structure. Since this last method cannot be always implemented at first, analytical sizing is usually the first step.

2.1 Analytical sizing

The initial sizing of airframe structural components can be achieved in two ways. Firstly, the classical approach consists of the use of loading data to determine shear force, bending moment and torque diagrams and then, evaluating the initial size of the structural members.

Alternatively, the load distribution can be used directly to obtained a first sizing using computational techniques. These methods are based on an initial definition of the structural disposition and they imply an iterative process that needs to re-calculate the sizes of the components. GUESS, the structural sizing module within NeoCASS follows this type of procedure.

Design procedure is similar in both approaches and requires a knowledge of the following:

- comprehensive load distributions, together with any particular concentrated load inputs
- any relevant airframe life requirements and stiffness criteria
- initial definition of the location of the main structural components.
- initial choice of the main materials of construction. In terms of the initial sizing of the members the main distinction is between metals and composites.
2.1.1 Synthesis technique

This structural analysis method requires more detailed specifications, usually obtained through an expert program or using an arbitrary structural layout. Unless a design resembles a similar aircraft, this is not necessarily an efficient approach. In GUESS, the first approximation to the sizing is based on the AcBuilder input data.

It is often more convenient to use the elementary theory to provide an initial estimation of the main structural items dimensions. This provides a good basis for input to an advanced analytical treatment.

The simple approach enables understanding of the way in which the structure functions, it also provides a validation of the concept and is a reference for checking the output of a more advanced analysis, as shown in [4, chapter 15].

2.2 Structural models

Aircraft structures are very complex, so to get a working model at the first stages of design they need to be simplified. The structure can be considered as the sum of different components, that can be made of either beams, or boxes. The first are simpler and demand more general data, while the latter provide more accurate results but may require initial specifications that are not yet determined.

2.2.1 Beam model

In order to discretize a structure, the traditional mathematical model is developed using a beam like structure, where the aircraft components are modeled as beams.

In this method, the reference axes are usually set at the flexural axis. The flexural axis is the locus of the shear centers along the span (see [8, page 511]). The beams are capable of bending, shear, torsional and axial deformation and are divided into several elements, allowing the use of more refined meshes.

![Figure 2.1: Beam-like representation of a wing](image-url)
The combination of all these basic elements is called a stick or beam model, shown in figure 2.1.

![Figure 2.2: Cross section parameters](image)

The mechanical properties are generally estimated from the member section properties, represented in figure 2.2, using classical structural analysis methods. The shear center, the point in which the applied shear forces will not induce twisting to the wing, will be used as the reference for the structural sizing. The reference point in the figure (c/4) shows the aerodynamic center.

This method is easy to implement and needs few calculations.

### 2.2.2 Box representation

The beam estimation is rather inaccurate; considering the aircraft structure as simple beams might be suitable design stage, where the detailed structure has not yet been defined, where properties from previous aircraft might be employed. The fidelity of the model also decreases when unconventional configurations, such as blended wing-body airframes, are considered.

However, a more detailed and accurate model can be developed using finite element methods that can represent the ‘box-like’ construction of the structure. In this case, the model will simulate bending and twisting of the entire structure, not the equivalent beam.

This approach provides finer results, but it is highly demanding from a computational point of view. For this representation of an aircraft, there is a huge number of nodes so that a condensation has to be performed, the final model will be similar to that represented in figure 2.3. This process allows to reduce a box like representation into a beam model, without compromising accuracy. A more detailed description can be seen in reference [10, page 427]
22.1.5 Stiffness Model – ‘Box-Like’ Condensed to a ‘Beam-Like’ Model

It was explained in the last section how lumped masses could be linked rigidly to the nodes of a beam model. However, for the box-like representation of an aircraft, there are a huge number of nodes and so a condensation has to be performed; this process reduces the stiffness model to correspond with a limited number of structural reference (or nodal) points lying on the structural reference axes (or elsewhere as necessary). The condensation is typically carried out using a method such as Guyan reduction (see Appendix D) where the FE stiffness model of original order $N$ is reduced to a significantly smaller set of master degrees of freedom $N_m$ corresponding to the chosen reference axis (and any other chosen points).

The master responses, once they have been calculated from the reduced model, may be used to obtain the responses at the slave degrees of freedom $N_s$ ($= N - N_m$) that were condensed out. An example of a condensed beam-like model is shown in Figure 22.4 where the lumped masses are shown and where additional condensation points are employed to represent the engine pylon; similar arrangements could be used for the landing gear support points.

The representation of the control surface behaviour depends upon the case being considered. For dynamic loads calculations, the control surface modes may be ignored and control rotation simply treated as imposing forces and moments on the reference axes. However, for aeroelastic calculations such as flutter, the control modes need to be represented and so any reduced model would be extended to include suitable condensation points in the region of the control surface; mass stations would still be linked rigidly to chosen grid points.

Figure 22.4: Condensed beam-like FE model on structural reference axes. Reproduced by permission of Airbus.

Figure 2.3: Condensed beam-like model representation
Chapter 3

Aeroelasticity in structural sizing

Aeroelasticity studies the interaction between airflow and flexible structures. Aerospace light-weight structures can present considerable elastic deformation during operation. Thus, the deformed structure experiences different loads than the original, which can lead to unexpected problems.

The design process of a modern aircraft is an interdisciplinary process in which aeroelasticity is deeply involved. This can be appreciated in Collar’s well-known aeroelastic triangle, shown in figure 3.1.

There are two types of problems in aeroelasticity. The first group, static aeroelasticity, concerns the interactions between the aerodynamic loads and the structure, the distortions loads provoke affecting the geometry, thus modifying the aerodynamic loads. This leads to failure in a too flexible structure or to an equilibrium in
a more stiffen one.

The second kind of problems involve the inertia of the structure as well as the aerodynamic and elastic loads. Under dynamic loads, oscillations of the structural components may cause failure if their natural frequencies are excited. This type of problems are included in dynamic aeroelasticity and are usually solved using a modal analysis. A more detailed classification is available in [8, page 745].

3.1 Static aeroelasticity

Static aeroelasticity studies the deflection of flexible aircraft structures where aerodynamic loads are considered to be independent of time. The interaction between the wing deflections and the aerodynamic loads determines the bending and twist and must be considered to model the static aeroelastic aircraft behavior.

The equilibrium equations can be simplified eliminating the time-dependent elements, therefore omitting the inertial forces, only the elastic terms will be included in the equations.

Also, only steady aerodynamic forces need to be included in the analysis. Therefore, the problem is simplified to finding the steady-state solution of the system:

\[ \mathbf{Kv} = \mathbf{f}_e \]  
\[ \text{(3.1)} \]

where \( \mathbf{K} \) is the stiffness matrix, \( \mathbf{v} \) is the elastic deformation of the wing, \( \mathbf{f}_e \) are the contributions to the external forces: aerodynamic, control deflection and weight, as shown in [1, page 61]. Disclosing these terms, we will have:

\[ \mathbf{Kv} = q\mathbf{Q}_0\mathbf{v} + q\mathbf{Q}_0\mathbf{v}_0 + q\mathbf{f}_0\delta + \mathbf{f}_w \]  
\[ \text{(3.2)} \]

in which \( q \) is the dynamic pressure and, \( \mathbf{Q} \) corresponds to the different terms of the aerodynamic transfer matrix. Rearranging the equation:

\[ [\mathbf{K} - q\mathbf{Q}_0]\mathbf{v} = \mathbf{f}_{a0} + \mathbf{f}_{a\delta} + \mathbf{f}_w = \mathbf{f}_{\text{tot}} \]  
\[ \text{(3.3)} \]

Given that the structure is statically stable, this linear system of equations can be solved for a given dynamic pressure \( q \).

Most important static aeroelastic phenomena include divergence and control reversal.

3.1.1 Aeroelastic divergence and elastic twist

Instabilities due to divergence occur when the aerodynamic forces caused by aeroelastic deformation overcome the elastic restoring forces in the wing. At the divergence pressure we have equilibrium:

\[ \mathbf{Kv} = q\mathbf{Q}_0\mathbf{v} \]  
\[ \text{(3.4)} \]
The tip twist increases with dynamic pressure leading to divergence. When the divergence condition is reached the twist tends to infinity, leading to structural failure.

For a fixed root wing, the dynamic pressure at divergence \( q_w \) is found as:

\[
q_w = \frac{3GJ}{ec^2s^2a_w}
\]  

(3.5)

where \( s \) is the semispan, \( c \) the reference chord, \( a \) is \( a_w = C_L/\alpha \). A more comprehensive explanation is found in reference [10].

The smaller the distance between the aerodynamic centre and the flexural axis, and the greater the torsional rigidity \((GJ)\), the greater the divergence speed becomes. If the flexural axis lies on the axis of aerodynamic centers there is no twist due to aerodynamic loading, so that divergence will not occur. If said axis were actually forward of the aerodynamic centre, the applied aerodynamic moment would become negative preventing divergence.

In general, divergence is not the most requiring phenomenon for sizing the aircraft for it happens at higher speeds than flutter.

Divergence can be computed following a modal approach, used in reference [1]. If there is a flight speed at which the aerodynamic forces due to the elastic deformation overcome the elastic restoring forces, divergence occurs. Equilibrium (Eq. 3.4) is found at the divergence dynamic pressure and can be rearranged as an eigenvalue problem (Eq. 3.6).

\[
[K - qQ_0]v = 0
\]

(3.6)

The dynamic pressure is the eigenvalue and the elastic deflection, the eigenvector. Thus, the smallest real eigenvalue \( q_D \) is the divergence dynamic pressure.

Depending on the airspeed, the structure will present different behaviors:

- If \( q < q_D \), the structure is stable.
- At \( q = q_D \), the structure is neutrally stable.
- When \( q > q_D \), the structure is unstable and \( v \to \infty \) for any perturbation.

### 3.1.2 Control effectiveness and reversal

As the speed increases the effectiveness of the control surfaces compared to the rigid wing decreases until a critical speed when there is no response of the control surface. At speeds greater than the reversal speed, the action of the controls reverses.

Defining the control efficiency in roll as the ratio between the deformable and rigid roll rates, as shown in reference [10, page 144] allows to determine the control reversal speed by setting the efficiency to zero. In such manner, the dynamic reversal pressure, hence the speed is found at:

\[
q_{rev} = \frac{3GJa_C}{c^2s^2a_w(ea_C - b_C)}
\]

(3.7)
On the other hand, an eigenvalue approach will also determine reversal, as stated in [1].

For a wing structure, the rolling moment is calculated by integrating the lift distribution along the span. Using a linear model, the lift will depend on the elastic deformation $v$ and the control surface deflection $\delta$. The rolling moment and the rolling moment coefficient can be expressed as:

$$ L = qSlC_{L_v}^T v + qSlC_{L_\delta} \delta $$

$$ C_L = \frac{L}{qSl} = C_{L_v}^T v + C_{L_\delta} \delta $$

where $C_{L_\delta}$ is the common rolling moment derivative with respect to $\delta$ and $C_{L_v}$ is a vector that contains the derivatives with respect to the elastic displacement. To compute the coefficient we need to take into account the deformation due to the aileron deflection which is obtained by solving the linear system of equations

$$ [K - qQ_0]v = qf_0 \delta $$

The reversal speed can be computed directly by noting that $C_L = 0$ is a condition for reversal. Imposing such condition, a relation between $\delta$ and $v$ is established. Inserting the relation in equation 3.10 we obtain the eigenvalue problem:

$$ [K - q(Q_0 + Q_\delta)]v = 0 $$

where $Q_\delta = -f_0C_{L_v}^T/C_{L_\delta}$ is the aerodynamic transfer matrix that reproduces the aerodynamic forces caused by the control deflection $\delta$, giving the opposite rolling moment of $v$. This problem is solved similarly as the divergence for the smallest real eigenvalue representing the reversal dynamic pressure $q_{rev}$.

In NeoCASS, the elastic contributions to the derivatives are computed directly while evaluating the desired maneuver, so we will monitor the value of the roll moment coefficient due to the aileron deflection.

### 3.2 Dynamic aeroelasticity

The calculations for dynamic aeroelasticity problems are carried out based on a modal analysis. The vibration solver allows to get the natural frequencies and the corresponding modal shapes for the models.

For a wing, the frequency-domain equations of motion can be expressed as follows. Disregarding the external forces that do not depend on the elastic deformation as described in [1, page 67], the equations can be written as:

$$ M\ddot{\mathbf{v}} + K\dot{\mathbf{v}} = q\mathbf{Q}(\hat{p})\dot{\mathbf{v}} $$

(3.12)
3.2. DYNAMIC AEROELASTICITY

where the aerodynamic transfer matrix $Q$ depends on the reduced Laplace variable $\hat{p} = pb/u$, $p$ is the Laplace variable for the transform, $b$ is the aerodynamic reference length and $u$ the speed along the x axis.

To obtain the nonlinear eigenvalue problem, the equation needs to be rearranged:

$$[M\hat{p}^2 + K - qQ(\hat{p})]\hat{v} = 0$$  \hspace{1cm} (3.13)

Making $q = 0$ we obtain the vibration problem, that will provide the natural frequencies and modal shapes.

$$[M\hat{p}^2 + K]\hat{v} = 0$$  \hspace{1cm} (3.14)

The stability of an aeroelastic mode is determined by the real part of the eigenvalue. If all eigenvalues have negative real parts, the airframe is aeroelastically stable. Still, if the real part of any eigenvalue becomes positive, the structure will suffer a flutter instability affecting said mode.

### 3.2.1 Flutter

At some critical speed, flutter speed, the structure sustains oscillations following some initial disturbance. Below this speed, oscillations are damped, while above it, structural behavior changes so unstable oscillations occur leading to failure.

The eigenvalue problem is highly demanding computationally, for a real aircraft. Hence, a more efficient method is introduced by using a set of modal coordinates. To do so, we must solve the vibration eigenvalue problem first, defined in equation 3.15, where $K$ is the stiffness matrix and $M$ the mass matrix.

$$[K - \omega^2M]z = 0$$  \hspace{1cm} (3.15)

With the frequencies ($\omega$) and modal shapes($z$) from the vibration problem we approximate the aeroelastic motion according to equation 3.16 where $\eta$ represents the new modal coordinates.

$$\hat{v} = \eta_1z_1 + \eta_2z_2 + \ldots + \eta_mz_m = Z\eta$$  \hspace{1cm} (3.16)

Normalizing the eigenvectors, the flutter problem can be written in non-dimensional form as:

$$\left[I\hat{p}^2 + \left(\frac{b}{u}\right)^2 \Omega - \frac{\rho b^2}{2} \hat{A}\right]\eta = 0$$  \hspace{1cm} (3.17)

where $\hat{A}$ is the generalized aerodynamic matrix, $I = Z^TMZ$ and $\Omega = Z^TKZ = \text{diag}(\omega^2_j)$

By using this method, we need to solve the $z$ and $\omega^2$ problem, but we can set the size of the flutter problem $\eta$. That allows the computational load to be significantly reduced while keeping accuracy.

The flutter frequency and modal shape are then computed with the $p$-$k$ method, as shown in [1]. This is an iterative method based on that, the structure will perform
undamped vibrations at the flutter boundary, so the forces $Q(k)$ will be a reasonable approximation to the transfer matrix $Q(\hat{p})$ for weakly damped motion.

This implies that a flutter solution in the form $\hat{p} = ik$ to the approximate problem 3.18 will be a solution to the full eigenvalue problem.

$$\left[ \left( \frac{u}{b} \right)^2 M \hat{p}^2 + K - qQ(k) \right] \hat{v} = 0$$

Given the aerodynamic matrix, the eigenvalue problem must then be solved for a given airspeed. Then, the problem is solved iteratively considering $k$ a parameter. For every $k$ value we can solve a linear eigenvalue problem for a set of eigenvalues $\hat{p}_j^2(k)$. If any of the solutions satisfies 3.19, the solution must be an eigenvalue to 3.18.

$$\text{Im}(\hat{p}_j(k)) = k$$

If all eigenvalues have negative real parts, the structure is stable. The flutter speed is defined as the lowest airspeed for which some eigenvalue crosses the imaginary axis.
Chapter 4

Methodology in NeoCASS software

NeoCASS is a CEASIOM module performing the structural and aeroelastic computations for deformable aircraft. It is a collection of Matlab analysis modules for initial aircraft structural sizing, static and modal analysis and aeroelastic analysis.

The program is divided in two main components: GUESS, which carries out the structural sizing for the models, and SMARTCAD, which analyzes static and dynamic aeroelastic behavior of aircraft.

The software uses AcBuilder xml files as its main input, enabling the user to make some additional considerations for the analysis. This input is used to run the GUESS structural analysis that provides the model needed for SMARTCAD analysis.

4.1 AcBuilder

AcBuilder is the CEASIOM module that models the geometry and some other parameters that are stored in the xml file. It also provides an initial estimate of the weights and center of gravity of the aircraft. There are several menus related to the different aspects of the software (Project, View, Geometry, Weights & Balance, Technology and Help).

NeoCASS uses the xml file as an input, for the geometry, material, weights and wingbox information. Close attention should be paid to the following items:

**Geometry:** Especial focus should be taken on checking the present flag for all the control surfaces to be analyzed. Currently, no leading edge surfaces are taken into account in NeoCASS. Winglets and tailbooms can only be analyzed through Guess modify.

**Wingbox:** The wingbox is indirectly defined from the wing fuel tank. The fore and aft spars defined for the tank determine the position for the front and back spars that will be used in NeoCASS. The elastic axis is defined in a similar fashion, calculated from the tank data. Figure 4.2
Figure 4.1: General structure of the NeoCASS code
Technology: The technology menu (figure 4.3) in AcBuilder allows the user to specify a variety of parameters that will be stored in the xml file for other parts of the CEASIOM suite to use. The main fields related to NeoCASS are:

- VLM calculations: To enable the vortex lattice method solver for the stability and control derivatives the corresponding checkboxes should be flagged. This solver provides a more reliable calculation than the standard NeoCASS does.

- Optimization, enables the optimization module within NeoCASS.

- Number of panels: There are several fields related to the aerodynamic panels used in NeoCASS. There are separate entries for each lifting surface and its corresponding control surfaces.

![Figure 4.2: AcBuilder fuel specification screenshot](image-url)
NeoCASS is divided into two different modules, GUESS and SMARTCAD. GUESS is used as the model generator, pre-processing the xml data from the general input file, as shown in figure 4.1.

GUESS (Generic Unknowns Estimator in Structural Sizing) computes the structural, and aerodynamic models from the xml input file that will be used for the aeroelastic analysis in SMARTCAD. GUESS has two variations: standard and modify. Figure 4.9

For both modes, the xml file from AcBuilder is the essential input. The difference lies in the sizing maneuvers, GUESS standard uses a default set of flight conditions whereas GUESS modify requires some flight condition input from the user.

The Standard mode uses the predefined sizing maneuvers and simplified aerodynamics whilst the Modify mode gets user-defined maneuvers and computes rigid aircraft trimming using vortex lattice method (VLM) based aerodynamics. As a consequence, Standard mode calculations are faster while Modify mode will achieve more reliable results.

At the end of the sizing calculations, a structural stick model is provided for SMARTCAD, this model is obtained through a semi-monocoque method. The
4.2. GUESS

Aerodynamic and aeroelastic models are automatically generated as well. Additional files regarding the different mass configurations will be provided, if necessary.

This aeroelastic model can be processed by SMARTCAD to compute static and dynamic aeroelasticity calculations and optimization of the GUESS-generated structural model according to the aeroelastic response.

4.2.1 GUESS structural model

The beam model used in GUESS is computed in several stages.

Firstly, the aircraft components are simplified to be represented as a collection of beam elements, representing the inertial properties (figure 4.4). Basically, a correct analysis allow the complex structures to be idealized in simple models (figures 4.4, 4.5). To perform such reductions the appropriate constraints have been implemented to consider the behavior of the structure regarding possible failure modes, such as local and global buckling. The cross-section of the wing is modelled as a simpler semi-monocoque wingbox, and the different beam elements are sized accordingly to the corresponding inertia properties.

![Beam Element Idealization](image)

Figure 4.4: Beam element idealization

Then, a geometric stick model is computed, using analytical simplified methods. The data obtained from this initial sizing will then be used to interpolate a more detailed mesh.

In the final model, the grid will present cruciform nodes, formed by beam elements, the center element being the one from the initial model and the ending points the so-called ‘stress-recovery points’ where the stresses are calculated with the interpolated data.
4.8. Engines

Engine thrust and mass are evident for the design task. Prototypes of engines inclusive defined load introduction points, gears and gear spars are modelled equivalently. Gears and gear spars are modelled as additional objects. The location of the engine reference point is defined by relative span and offset to the leading edge. At-
tachments to the wing are foreseen at one or two load introduction points. For this, the points of the surface mesh have to be transferred to cylinder coordinates $R$, $\phi$, $Z$. Un-
fuselage sections, too. For this, the points of the surface mesh have to be flagged as stringers which require modified gen-
eration rules.

4.9. Fuselage

The fuselage flexible structure definition. Today the fuselage is carried out:

1. Fuselage is the first component to be assembled, defined by the minimum number of points required to describe its geometry.

2. The wing is the second component to be connected. One point, coincident with the first point on the wing root is added to the fuselage in order to assemble both components.

3. Vertical tail is now added to the model. The connection is performed as follows: if the first point for V-tail does not lay on the reference fuselage line, an extra sector is added to the tail, if it does not link to any existing point, it is also added to the fuselage. These extra nodes will only be added if necessary.

4. The horizontal tail is the last component assembled since it can be connected to either the fuselage or the V-tail.

This enables a closed model to be created by detecting the mutual intersections of the different components. In some cases, additional sectors are added to allow the connectivity, as shown in figure 4.6.

**Geometric stick model**

The aircraft is made of single independent components that need to be assembled together by sharing some points in common. Each component is composed of different sectors that allow defining different properties along the axis of the component. The created geometric stick model is converted into the classic structural stick model; each line becomes one or more beams with associated structural and inertial properties.

To accomplish the generation of the geometric stick model, the following process is carried out:

1. Fuselage is the first component to be assembled, defined by the minimum number of points required to describe its geometry.

2. The wing is the second component to be connected. One point, coincident with the first point on the wing root is added to the fuselage in order to assemble both components.

3. Vertical tail is now added to the model. The connection is performed as follows: if the first point for V-tail does not lay on the reference fuselage line, an extra sector is added to the tail, if it does not link to any existing point, it is also added to the fuselage. These extra nodes will only be added if necessary.

4. The horizontal tail is the last component assembled since it can be connected to either the fuselage or the V-tail.

This enables a closed model to be created by detecting the mutual intersections of the different components. In some cases, additional sectors are added to allow the connectivity, as shown in figure 4.6.
Analytical sizing

Once the discretized model is available, ultimate loads are estimated to carry out the sizing task. Aerodynamic, inertial and propulsive loads are considered.

Inertial loads are considered from the connection with the WB module. All the non-structural masses are incorporated as lumped masses or mass distributions. Preliminary weights are estimated by statistical formulas and then refined in an iterative process. Propulsive forces are considered since the engine position is already specified so contributions to internal forces and moments can be easily calculated.

A sizing performed on bending loads is carried out for different maneuvers and when total shear force and bending moment along the axes have been computed, net stress resultants are calculated on a station-by-station basis. Then, the minimum amount of material required to avoid failure is determined.

Finally, section stiffness is determined: while axial and bending stiffness can be calculated by analytical formulas, shear and torsional stiffness are determined through a semi-monocoque solver. Structural, primary and total weights are computed by applying regression techniques to the ideal weights obtained with the above method. The new estimations are then used to improve the overall airframe weight.

Finite-element aeroelastic model

Once the airframe sizing is finished, the internal aero-structural mesh is exported in a SMARTCAD formatted file (similar to NASTRAN format).

Stiffness, inertial and aerodynamic properties need to be interpolated from the analytical model to the final mesh. Grid nodes are laid to form a cruciform shape and keep the connection between lifting surfaces and fuselage to allow the load transfer.

Mesh nodes are connected through 3-point beam elements, considering a central point and the two ending points of the beam. The node is positioned in the center, the ending points being the connections between the adjoining nodes. This sums up as a beam element.

Figure 4.6: Geometric and analytical models
Then, the stress recovery points are located in every segment midpoint, interpolating the stresses to correspond to actual points in the wing-box and fuselage. The elements can be appreciated in figure 4.8.

Apart from the mechanical properties and the basic stick model, some other data are included:

- Stress recovery points: Correspond to actual points of wing-box and fuselage.
- Extra nodes for coupling with the aerodynamic model: These nodes are perpendicular to the beam axis and connected to beam nodes by rigid elements.
• Non-structural masses: Either as lumped masses on mesh nodes or as distributions of non-structural densities along the beams.

Airfoils are defined at different spanwise control sections. This allows to determine the dimensions of the wing-box and the allowable volume for fuel. Also, the camber mean line can be used to correctly apply no-penetration boundary conditions during the aerodynamic calculations.

Finally, considering the early design phase the framework is intended for, control surfaces are represented by their aerodynamic contribution only, neglecting their inertia, dynamics and actuation systems.

Currently the structural model is represented by a three-node linear/non linear finite-volume beam element. As mentioned, classic lifting aerodynamic surfaces are used. A comprehensive explanation is described in [2].

**4.2.2 GUESS code structure**

Even though the full GUESS code is quite complex, the workflow can be simplified thoroughly without losing track of the process.

First of all, depending on the selected mode, either guess.m or guess_mod.m will be called from the GUI. Both of these functions serve as a main program, that will be calling all the required subroutines. They will first call the stick_model.m to compute the stick model, afterwards, the program will call a range of writing functions to export the output and finally, the mechanical properties are interpolated and exported by the setup functions. The main difference between the solvers is that whereas guess.m has the sizing maneuvers implemented, in guess_mod.m sizing maneuvers are an input so the trim solver (solve_free_lin_trim.m) is needed to perform the analysis. This solver is a part of SMARTCAD and requires some other function handling to run.
4.3 SMARTCAD

SMARTCAD (Simplified Model for Aeroelasticity in Conceptual Aircraft Design) is the module dedicated to aeroelastic analysis and optimization. The input for this calculations are both the GUESS output model and the SMARTCAD card, a file that specifies the analysis parameters.

4.3.1 Analysis setup, the SMARTCAD card

The analysis options are provided through a plain text file. Solver and relevant parameters are specified in the SMARTCAD card. An example card is provided in figure 4.11.
By means of the user interface the cards are prepared for the different analysis and solvers, then they will be stored in a .inc file. Firstly, the reference values (reference chord, wing area and wingspan) should be typed in the pop-up window (figure 4.12). Afterwards, the settings panel should be filled in according to the
desired analysis. Last, pressing the generate button on the main GUI will provide a complete card file for the solver.

Finally, pressing the assembly button, we create a .dat file that will specify which analysis cards and GUESS models we would like to analyze. Within this file, the supplementary files that GUESS provides for the different mass configurations can also be added.

The SMARTCAD card can also be generated and edited by a plain text editor, for options not available within the GUI. All the different parameters and the file structure are shown in the NeoCASS manual [7].

4.3.2 Static aeroelasticity and trim analysis

Static aeroelasticity solvers implemented in NeoCASS allow to perform trim analysis and establish a comparison between the rigid and deformable aircraft. SMARTCAD establishes a comparison between the deformable and the rigid aircraft. To do so, the VLM/Rigid solver (solve_vlm_rigid) is run in all cases.

The default solver for this analysis is solve_free_lin_trim. It will compute the trim solution for free flying aircraft. On the other hand, the solver solve_lin_aerostatic performs analyses for other boundary conditions.

Analysis parameters, including aircraft characteristics and flight conditions, are provided through the trim cards. To create a trim card through the GUI, the following directions should be considered:

- Control surfaces must be selected, according to the model geometry.
- Data disclosed in figures 4.13 and 4.14 must be included in all cases, with special attention to the ‘Support’ node number, that can be found in the .inc output file from the GUESS model.
- Remaining fields in the flight maneuver panel just will need to fit the model constraints.
• Each of the flight conditions analyzed will generate a different trim card, with its own trim ID within the same SMARTCAD .inc file.

This SMARTCAD module is easily used to carry out static aeroelastic calculations such as divergence speed and control reversal, by running multiple cases and monitoring the corresponding variables.

The trim solver provides the dynamic pressure of divergence for any trim analysis if a special field is enabled. This is set by adding to the trim card an optional parameter as shown in figure 4.15.

Control reversal speed is obtained by computing the deformable aircraft stability derivatives until the monitored field becomes zero.

Stability and control derivatives are computed during the trim analysis. The values are calculated considering both rigid and deformable aircraft. This data are stored in the beam_model.Res.Aero.RStab and beam_model.Res.Aero.DStab fields.

**4.3.3 Modal and flutter analysis**

The vibration modes of the structure are computed by the solve_eig function. The output for this solver comprises the natural frequencies and the modeshapes.
In order to run this analysis, only the reference values and the desired number of modes need to be specified.

The flutter calculations are carried out by `solve_linflutt`. This solver tracks each of the vibration modes and provides a V-g plot where the changes in damping are shown. Therefore, the input parameters include those of the vibration solver. Moreover, the data shown in figure 4.16 must be specified.
4.3. SMARTCAD

The beam_model structure, as disclosed in Appendix 8, is a Matlab structure created by NeoCASS to store all the data from aeroelastic analysis, including the structural model and the weight and balance input from GUESS. A limited amount of these data can be accessed by the GUI via the results panel, figure 4.17. Thus, access to this structure is essential to have a comprehensive output from the program.

By using the command global beam_model after an analysis has been run we gain access to all the data. The main fields involved in storing the results are under beam_model.Res, the most important being:

- **Bar**: Stores the structural results, including stresses and strains in the beam model.
- **Aero**: Stability and control derivatives, trim solution and divergence dynamic pressure, among other aerodynamic data.
- **WB**: Weight and balance results from GUESS.

From the GUI, we have easy access to plots regarding modal shapes and deformed models corresponding to modal or trim solutions. Besides, a V-g flutter diagram is plotted automatically when carrying out such analysis.

Figure 4.16: Flutter analysis settings
Figure 4.17: Results panel
Chapter 5

Testing and verification

5.1 KTH Compendium

As a first assessment of the NeoCASS suite, the computer tasks in the KTH compendium for the Course *Aeroelasticity of Slender Wing Structures in Low-Speed Airflow* will be recalculated (Reference [1]). These tasks perform several analyses to a simple rectangular half wing that is also analyzed in a wind tunnel. We will compare the results obtained with NeoCASS to those reached with Matlab or wind-tunnel analysis.

5.1.1 Model description

The model, featuring a plate with constant thickness \( t \) and density \( \rho_m \) is made of a glass fiber and epoxy composite. Shown in figure 5.1 below, it has a simple rectangular planform. Model’s properties are shown in table 5.1

![Figure 5.1: Wing model](image)

The wing is modeled as a beam for both the Matlab code and NeoCASS analyses, more accurate results would have been achieved using a plate model. However, the beam offers a good global approximation.
Table 5.1: Model properties

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Description</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>l</td>
<td>semispan</td>
<td>1200 mm</td>
</tr>
<tr>
<td>2b</td>
<td>chord</td>
<td>300 mm</td>
</tr>
<tr>
<td>2ba</td>
<td>aileron chord</td>
<td>60 mm</td>
</tr>
<tr>
<td>E</td>
<td>Young’s modulus</td>
<td>26.6GPa</td>
</tr>
<tr>
<td>G</td>
<td>shear modulus</td>
<td>5.7GPa</td>
</tr>
</tbody>
</table>

NeoCASS model uses this number of elements for the beam and this number of panels for the aerodynamical mesh, with an extra 3 panels in the aileron surface. The full length aileron is modeled by linking together the flaps and ailerons. Moreover, the input file from AcBuilder will feature a complete aircraft and will be edited after running GUESS to present just the half wing model.

5.1.2 Modal analysis, Vibration testing

Once the files have been edited to have a half wing model, the calculations will be started computing the natural frequencies of the structure, as shown in Table 5.2. For these cases, the modal shapes are also plotted in figures 5.2.

Table 5.2: Natural frequencies

<table>
<thead>
<tr>
<th>Mode</th>
<th>NeoCASS</th>
<th>Matlab</th>
<th>wind tunnel</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>1.2313</td>
<td>1.059</td>
<td>1.20</td>
</tr>
<tr>
<td>2</td>
<td>6.8259</td>
<td>6.053</td>
<td>6.70</td>
</tr>
<tr>
<td>3</td>
<td>7.7154</td>
<td>6.732</td>
<td>6.97</td>
</tr>
<tr>
<td>4</td>
<td>20.4633</td>
<td>16.779</td>
<td>17.97</td>
</tr>
<tr>
<td>5</td>
<td>21.593</td>
<td>19.737</td>
<td>20.24</td>
</tr>
</tbody>
</table>
5.1.3 Static aeroelasticity. Trim condition

In this analysis, a 25-degree deflection will be applied to the control surface. Running the trim condition at $M = 0.1$ the plots shown below are obtained. To reproduce the clamp in the half wing, a complete wing model has been used, featuring a
simple support in the root. Figure 5.3 shows the deformed structural and aerodynamic models, with a scale factor of 10. Figure 5.4 shows the bending moment, and the shear force distribution along the span, which show the adequate conditions to simulate the clamping. The stability and control derivatives for this analysis are present in tables 5.3 and 5.4, respectively.

Figure 5.3: Deformed models. Scale factor: 10

Figure 5.4: Bending moment and shear forces distributions along the span
Rigid

<table>
<thead>
<tr>
<th>$C_{L\alpha}$</th>
<th>$C_{M\alpha}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>9.6104</td>
<td>2.5809</td>
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</tbody>
</table>

Deformable

<table>
<thead>
<tr>
<th>$C_{L\alpha}$</th>
<th>$C_{M\alpha}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>9.556</td>
<td>2.5344</td>
</tr>
</tbody>
</table>

D/R

<table>
<thead>
<tr>
<th>$C_{L\alpha}$</th>
<th>$C_{M\alpha}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.99434</td>
<td>0.98198</td>
</tr>
</tbody>
</table>

(a) $\alpha$ derivatives

Rigid

<table>
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<th>$C_{S_p}$</th>
<th>$C_{L_p}$</th>
<th>$C_{N_p}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>-3.3741e-26</td>
<td>-4.36</td>
<td>3.1163e-08</td>
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Deformable

<table>
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<th>$C_{N_p}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>-3.5653e-05</td>
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<td>1.3978e-05</td>
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D/R

<table>
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<th>$C_{N_p}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>//</td>
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<td>448.5445</td>
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</table>

(b) $p$ derivatives

Rigid

<table>
<thead>
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<th>$C_{Mq}$</th>
</tr>
</thead>
<tbody>
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</tbody>
</table>

Deformable

<table>
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<th>$C_{Mq}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>5.1297</td>
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</tr>
</tbody>
</table>

D/R

<table>
<thead>
<tr>
<th>$C_{Lq}$</th>
<th>$C_{Mq}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.99321</td>
<td>1.0803</td>
</tr>
</tbody>
</table>

(c) $q$ derivatives

Table 5.3: Stability derivatives

Table 5.4: Control derivatives for $\delta_a$ deflection

### Aileron reversal speed

Control reversal speed is now studied. As in previous analyses, a 25-degree aileron deflection is used. To confirm that the aileron is inverted, the deformable to rigid ratio for the roll moment coefficient should be negative.

Table 5.5 shows the control derivatives, both rigid and deformable. The reversal is found at $M_R = 0.13$; $u_R = 44.2m/s$ with NeoCASS, whereas the Matlab code this speed is much lower ($u_R = 18.08m/s$).

<table>
<thead>
<tr>
<th>$C_{L_{\delta a}}$</th>
<th>$C_{L_{\delta a}}$</th>
<th>$C_{L_{\delta a}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.029497</td>
<td>0.024241</td>
<td>0.82183</td>
</tr>
<tr>
<td>1.9533</td>
<td>0.061534</td>
<td>0.81366</td>
</tr>
</tbody>
</table>

Table 5.5: $\delta_a$ derivatives
In NeoCASS, the control derivative regarding the aileron deflection is monitored while in the Matlab code used in the booklet, the eigenvalue problem is computed as stated in equation 3.10. The results achieved by NeoCASS may not be as accurate as expected because of the extensive modifications that there were made in order to get a working wing model from AcBuilder.

**Divergence speed**

The last static aeroelastic analysis carried out is divergence. For this kind of analysis NeoCASS will provide the divergence dynamic pressure that will be converted so that the divergence speed can be compared to the speed that is obtained with the KTH Matlab code.

Following this procedure, using NeoCASS, a divergence dynamic pressure of $\Delta q = 236.21 \, \text{Pa}$ is computed. That translates into a speed of $u_D = 19.8 \, \text{m/s}$. This result is in good agreement with the KTH code that provides $u_D = 20.15 \, \text{m/s}$ as a divergence speed.

**5.1.4 Flutter**

Running the corresponding analysis in NeoCASS, the results shown in Figure 5.5 are obtained; the flutter speed is $u_F = 17.7 \, \text{m/s}$. Flutter occurs at the second vibration mode ($6.83 \, \text{Hz}$), being the first torsional mode of the structure as can be observed in figure 5.2. This result is consistent with the Matlab code, where $u_F = 14.5 \, \text{m/s}$ for the second mode, as well.

![Figure 5.5: V-g plot for the half wing model](image-url)
5.2 Agard 445.6

In this section, a flutter analysis of the Agard wing is going to be carried out. This wing is a very common case in the literature, and its results are widely available. Finally, a comparison between NeoCASS and Nastran results will be established.

Flutter tests were carried out at the NASA Langley Transonic Dynamic Tunnel, were published in 1963 and re-published in 1987 [11]. Various wing models were tested (and broken) in air and Freon-12 for Mach number between 0.338 and 1.141.

The case most often used in the literature and that will be used is the ‘weakened 3’ model at zero angle of attack in air. This test model has been applied to validate many aeroelastic simulation techniques. For this analysis, we will perform flutter analysis at $M = 0.678$.

5.2.1 Model description

The AGARD 445.6 ‘weakened 3’ model has a symmetric airfoil (NACA 65A004) with a 4% thickness. The wing has a 45 degree quarter chord sweep angle, a semi-span of 0.762 m and a taper ratio of 0.66. The AGARD 445.6 is made of laminated mahogany and its stiffness is reduced by the holes drilled along the span. The planform is shown in figure 5.6.

The NeoCASS model is generated with data from the Ground Vibration Tests (GVT). This model is the only possible approach in this case since the models for the wing created through AcBuilder do not replicate the geometry and the structural model generated by GUESS is thus unreliable. However, the GVT model only allows to perform modal and flutter analysis.

![Figure 5.6: NeoCASS model for Agard 445.6](image-url)
5.2.2 Vibration modes and frequencies

Experimental data and extracted frequencies by NeoCass and Nastran are shown in table 5.6. The frequencies fit the experimental data well, as expected, since the GVT were used to obtain the models in both cases. The modeshapes computed with NeoCASS (Figure 5.7) correspond to those obtained from the experimental results.

<table>
<thead>
<tr>
<th>Mode no.</th>
<th>Experimental</th>
<th>NeoCASS</th>
<th>Nastran</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>9.6</td>
<td>9.60</td>
<td>9.60</td>
</tr>
<tr>
<td>2</td>
<td>38.1</td>
<td>38.16</td>
<td>38.16</td>
</tr>
<tr>
<td>3</td>
<td>50.7</td>
<td>48.35</td>
<td>48.34</td>
</tr>
<tr>
<td>4</td>
<td>98.5</td>
<td>91.54</td>
<td>91.54</td>
</tr>
</tbody>
</table>

Table 5.6: Normal frequencies for the Agard wing (Hz)

5.2.3 Flutter analysis

NeoCASS and Nastran provide V-g plots for the Agard 445.6 wing. The analysis has been performed for $M = 0.678$. For such conditions, the experimental data show a flutter speed of 231 m/s. NeoCASS calculates a flutter speed of 238 m/s, similar to that from Nastran. Such agreement of the results can be seen in figures 5.8, 5.9, flutter speed is defined where the damping of the first mode (9.6 Hz) becomes positive.

In this case, NeoCASS and Nastran show almost identical V-g plots. Both programs also provide coherent modal behavior, for the evolution of the tracked modes in the $p$-$k$ method is similar. These analyses provide close approximations to the experimental results.
Figure 5.7: First four vibration modes
Figure 5.8: NeoCASS flutter analysis for Agard 445.6
5.2. AGARD 445.6

Figure 5.9: Nastran results for Agard 445.6
Chapter 6

Confirmation of TCR analysis

TCR Transonic Cruiser is a case study for most packages in the SimSAC project. The TCR shows the difficulties in using standard methodology when designing aircraft operating in the transonic speed region. The goal is to compare the results gained by means of classic methodologies and the new tools such as NeoCASS. The design specifications and geometry are shown in table 6.1 and figure 6.1.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Specification</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cruise Mach</td>
<td>0.97 at 37000 ft</td>
</tr>
<tr>
<td>Range</td>
<td>5500 NM</td>
</tr>
<tr>
<td></td>
<td>+ 250 NM to alternate airport</td>
</tr>
<tr>
<td></td>
<td>+ 0.5 hour loiter at 1500 ft</td>
</tr>
<tr>
<td>Maximum payload</td>
<td>22000 kg</td>
</tr>
<tr>
<td>Passengers</td>
<td>200</td>
</tr>
<tr>
<td>Crew</td>
<td>2 pilots, 6 cabin attendants</td>
</tr>
<tr>
<td>Take-off distance</td>
<td>2700 m at MTOW</td>
</tr>
<tr>
<td>Landing distance</td>
<td>2000 m at MLW</td>
</tr>
<tr>
<td>Powerplant</td>
<td>2 turbofans</td>
</tr>
<tr>
<td>Maneuvering load factors</td>
<td>2.5; -1</td>
</tr>
<tr>
<td>Maximum load factors</td>
<td>3.1; -1.7</td>
</tr>
</tbody>
</table>

Table 6.1: Design specifications for TCR
### 6.1 Structural and aeroelastic model

After importing the AcBuilder model (Figure 6.2), the structural model is computed through GUESS, as shown in figure 6.3. The input model shows the aerodynamic mesh that will be linked to the structural model and the initial weight and balance data.

For the TCR model, several weight and balance parameters are computed to update the initial estimations. These are shown in tables 6.2, 6.3 and 6.4.

<table>
<thead>
<tr>
<th>Solver</th>
<th>X</th>
<th>Y</th>
<th>Z</th>
</tr>
</thead>
<tbody>
<tr>
<td>Standard</td>
<td>33.6401</td>
<td>0</td>
<td>-0.6588</td>
</tr>
<tr>
<td>Modify</td>
<td>35.3701</td>
<td>0</td>
<td>-0.4463</td>
</tr>
</tbody>
</table>

Table 6.2: CoG position
6.1. STRUCTURAL AND AEROELASTIC MODEL

Figure 6.2: Guess input model from xml

Figure 6.3: Guess structural model

<table>
<thead>
<tr>
<th>Solver</th>
<th>Weight (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Standard</td>
<td>51964</td>
</tr>
<tr>
<td>Modify</td>
<td>55835</td>
</tr>
</tbody>
</table>

Table 6.3: Estimated structural weights
6.2 Aerostatic analysis

We will perform an assessment of the TCR static behavior by analyzing some frozen maneuvers and determining the divergence speed for the aircraft.

6.2.1 Pull-up maneuver

Firstly, we will compute a pull up maneuver, with a load factor \( n_z = 3.1 \) at \( M = 0.6 \), sea level flight. With those trimming parameters, we determine the remaining variables, both rigid and deformable cases, shown in Table 6.5. We can appreciate an important change in both angle of attack and deflection of the canard between the deformable and rigid calculations.

<table>
<thead>
<tr>
<th>Variable</th>
<th>Rigid</th>
<th>Deformable</th>
</tr>
</thead>
<tbody>
<tr>
<td>X acceleration</td>
<td>(-0.209m/s^2)</td>
<td>(-0.828m/s^2)</td>
</tr>
<tr>
<td>Y acceleration</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>Z acceleration</td>
<td>(30.4m/s^2)</td>
<td>(30.4m/s^2)</td>
</tr>
<tr>
<td>(\alpha)</td>
<td>3.60°</td>
<td>6.86°</td>
</tr>
<tr>
<td>(\delta_c)</td>
<td>27.96°</td>
<td>(-1.62073°)</td>
</tr>
</tbody>
</table>

Table 6.5: Trim parameters \( n_z = 3.1 \) at \( M = 0.6 \)

6.2.2 Snap roll maneuver

This maneuver consists of a simultaneous deflection of both the canard and the rudder. Table 6.6 and Figure 6.5. The deformable case presents lower Y and Z accelerations and a slightly higher longitudinal acceleration.
6.2. AEROSTATIC ANALYSIS

Figure 6.4: Deformed model for the pull-up ($n_z = 3.1, M = 0.6$)

<table>
<thead>
<tr>
<th>Variable</th>
<th>Rigid</th>
<th>Deformable</th>
</tr>
</thead>
<tbody>
<tr>
<td>X acceleration</td>
<td>$0.332m/s^2$</td>
<td>$0.427m/s^2$</td>
</tr>
<tr>
<td>Y acceleration</td>
<td>$-4.39m/s^2$</td>
<td>$-3.67m/s^2$</td>
</tr>
<tr>
<td>Z acceleration</td>
<td>$13.65m/s^2$</td>
<td>$7.64m/s^2$</td>
</tr>
<tr>
<td>$\delta_c$</td>
<td>$30^\circ$</td>
<td>$30^\circ$</td>
</tr>
<tr>
<td>$\delta_r$</td>
<td>$30^\circ$</td>
<td>$30^\circ$</td>
</tr>
</tbody>
</table>

Table 6.6: Trim parameters $\delta_c = 30^\circ$ and $\delta_r = 30^\circ$

Figure 6.5: Deformed model for the snap roll maneuver ($\delta_c = 30^\circ, \delta_r = 30^\circ$)
6.2.3 Divergence speed

A cruise maneuver is analyzed, enabling the divergence solver. This solver provides the dynamic pressure of divergence for the given flight conditions. We will monitor $q$ while increasing the flying speed until the speed calculated through the dynamic pressure is the same as the divergence speed.

Divergence calculations from $M = 0.6$ are performed up to $M = 0.85$. At this speed, the divergence speed is lower than the airspeed, so $M = 0.8$ ($u_D = 272m/s$; $q_D = 1.89e + 05Pa$) is the maximum flying speed. These calculations are carried out for $n_z = 1$, sea level flight; the most critical altitude.

<table>
<thead>
<tr>
<th>Mach</th>
<th>Dynamic Pressure</th>
<th>Divergence Speed</th>
<th>Divergence Mach</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.60</td>
<td>2.52e + 05</td>
<td>475</td>
<td>1.40</td>
</tr>
<tr>
<td>0.65</td>
<td>2.39e + 05</td>
<td>449</td>
<td>1.32</td>
</tr>
<tr>
<td>0.70</td>
<td>2.25e + 05</td>
<td>418</td>
<td>1.23</td>
</tr>
<tr>
<td>0.75</td>
<td>2.08e + 05</td>
<td>379</td>
<td>1.11</td>
</tr>
<tr>
<td>0.80</td>
<td>1.89e + 05</td>
<td>325</td>
<td>0.96</td>
</tr>
<tr>
<td>0.85</td>
<td>1.66e + 05</td>
<td>243</td>
<td>0.71</td>
</tr>
</tbody>
</table>

Table 6.7: Divergence speed calculations. TCR h=0

6.3 Modal and flutter analysis

The vibration modes that will be used for the flutter analysis of the TCR are shown in table 6.8 and in figures 6.6 and 6.7.

<table>
<thead>
<tr>
<th>Mode</th>
<th>Frequency</th>
</tr>
</thead>
<tbody>
<tr>
<td>7</td>
<td>1.391</td>
</tr>
<tr>
<td>8</td>
<td>1.872</td>
</tr>
<tr>
<td>9</td>
<td>2.028</td>
</tr>
<tr>
<td>10</td>
<td>2.140</td>
</tr>
<tr>
<td>11</td>
<td>2.609</td>
</tr>
<tr>
<td>12</td>
<td>3.289</td>
</tr>
<tr>
<td>13</td>
<td>3.664</td>
</tr>
<tr>
<td>14</td>
<td>4.335</td>
</tr>
<tr>
<td>15</td>
<td>5.081</td>
</tr>
</tbody>
</table>

Table 6.8: TCR Vibration modes

The mode tracking calculation solver for flutter analysis did not show any instabilities for this aircraft. The analysis was set at a flying speed of $M = 0.9$ and
the tracking ranged from 0 up to 350 m/s. This is in good agreement with the data shown in references [6] and [5].

6.4 Stability derivatives

The stability derivatives computed for TCR at $M = 0.6$ are shown in tables 6.9, 6.10.

<table>
<thead>
<tr>
<th>Rigid</th>
<th>Deformable</th>
<th>D/R</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_{L_\alpha}$</td>
<td>3.7877</td>
<td>$C_{L_\alpha}$</td>
</tr>
<tr>
<td>$C_{M_\alpha}$</td>
<td>-4.269</td>
<td>$C_{M_\alpha}$</td>
</tr>
</tbody>
</table>

Table 6.9: $\alpha$ derivatives

<table>
<thead>
<tr>
<th>Rigid</th>
<th>Deformable</th>
<th>D/R</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_{S_\beta}$</td>
<td>0.2872</td>
<td>$C_{S_\beta}$</td>
</tr>
<tr>
<td>$C_{N_\beta}$</td>
<td>0.17389</td>
<td>$C_{N_\beta}$</td>
</tr>
</tbody>
</table>

Table 6.10: $\beta$ derivatives

Control derivatives for the aileron (shown in table 6.11), rudder (table 6.12) and canard (table 6.13) deflections. The results are similar to those in report [5]. The variations are caused by the use of different models for the GUESS sizing, since the maneuvers are the same.

There are considerable differences between the the rigid and deformable coefficients that illustrate the importance of the elastic terms to be considered to calculate the derivatives.

<table>
<thead>
<tr>
<th>Rigid</th>
<th>Deformable</th>
<th>D/R</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_{S_{\delta_a}}$</td>
<td>-0.0074672</td>
<td>$C_{S_{\delta_a}}$</td>
</tr>
<tr>
<td>$C_{L_{\delta_a}}$</td>
<td>0.07813</td>
<td>$C_{L_{\delta_a}}$</td>
</tr>
</tbody>
</table>

Table 6.11: $\delta_a$ derivatives

<table>
<thead>
<tr>
<th>Rigid</th>
<th>Deformable</th>
<th>D/R</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_{S_{\delta_r}}$</td>
<td>-0.12992</td>
<td>$C_{S_{\delta_r}}$</td>
</tr>
<tr>
<td>$C_{N_{\delta_r}}$</td>
<td>-0.092021</td>
<td>$C_{N_{\delta_r}}$</td>
</tr>
</tbody>
</table>

Table 6.12: $\delta_r$ derivatives
Table 6.13: $\delta_c$ derivatives

<table>
<thead>
<tr>
<th></th>
<th>Rigid</th>
<th>Deformable</th>
<th>D/R</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_{L_{\delta}}$</td>
<td>0.037246</td>
<td>-0.020834</td>
<td>-0.55937</td>
</tr>
<tr>
<td>$C_{M_{\delta}}$</td>
<td>0.21227</td>
<td>0.29536</td>
<td>1.3914</td>
</tr>
</tbody>
</table>

Figure 6.6: TCR vibration modes 1 of 2
Figure 6.7: TCR vibration modes 2 of 2
Figure 6.8: V-g plot for TCR
Chapter 7

Application to new conceptual design - Danbus

The Danbus is a medium range aircraft similar to Airbus A320. The AcBuilder model (Figure 7.1) that will be used in this analysis has the following dimensions:

<p>| | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Length</td>
<td>44.51 m</td>
</tr>
<tr>
<td>Wingspan</td>
<td>39.44 m</td>
</tr>
<tr>
<td>Wing surface</td>
<td>155.33 $m^2$</td>
</tr>
<tr>
<td>Reference chord</td>
<td>3.94 m</td>
</tr>
</tbody>
</table>

Table 7.1: Danbus dimensions

Figure 7.1: Danbus AcBuilder model
7.1 Structural weight

GUESS provides the stick model for the aircraft and the estimated structural weight. Different mass configurations are implemented in this case, to ensure the most requiring condition for the aircraft in each sizing maneuver. These configurations are taken into account in the GUESS modify sizing, that consists of the maneuvers shown in table 7.2 with a full loaded fuel tank and an empty one.

<table>
<thead>
<tr>
<th>ID</th>
<th>Maneuver</th>
<th>Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Cruise / Climb</td>
<td>(M=0.5, h=5000, n_z = 2.5)</td>
</tr>
<tr>
<td>2</td>
<td>Sideslip levelled flight</td>
<td>(M=0.5, h=5000, n_z = 1, \beta = 20)</td>
</tr>
<tr>
<td>3</td>
<td>Aileron abrupt input</td>
<td>(M=0.5, h=7000, n_z = 1, \delta_a = 25)</td>
</tr>
<tr>
<td>4</td>
<td>Snap roll</td>
<td>(M=0.5, h=5000, n_z = 1 \delta_r = 25, \delta_e = 25)</td>
</tr>
</tbody>
</table>

Table 7.2: Sizing maneuvers description

Thus, the standard solver calculations compute only the weight associated with the maximum fuel load whereas the modify also solver takes into account the empty fuel tank in the wing. The solvers provide the data in table 7.3. GUESS data are compared to RDS weight estimates from Rarymer, reference [9, page 90] and AcBuilder.

<table>
<thead>
<tr>
<th>Solver</th>
<th>Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Standard</td>
<td>25574 kg</td>
</tr>
<tr>
<td>Modify</td>
<td>22006 kg</td>
</tr>
<tr>
<td>AcBuilder</td>
<td>27897 kg</td>
</tr>
<tr>
<td>Raymer</td>
<td>26641 kg</td>
</tr>
</tbody>
</table>

Table 7.3: Estimated structural weight

To compute the structural weight, GUESS computes the stick model that will represent the structure, as shown in figure 7.2.
7.1. STRUCTURAL WEIGHT

Figure 7.2: GUESS stick model

The mass for the different parts of the aircraft are also computed by GUESS (Table 7.4). Once all the weights are obtained, GUESS also recalculates the CoG for the configurations provided (Table 7.5).

<table>
<thead>
<tr>
<th>Solver</th>
<th>Standard</th>
<th>Modify</th>
<th>AcBuilder</th>
<th>Raymer</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fuselage</td>
<td>13466</td>
<td>12766</td>
<td>8144</td>
<td>9957</td>
</tr>
<tr>
<td>Wing</td>
<td>8656</td>
<td>7423</td>
<td>7436</td>
<td>9761</td>
</tr>
<tr>
<td>Horizontal tail</td>
<td>2898</td>
<td>520</td>
<td>502</td>
<td>344</td>
</tr>
<tr>
<td>Vertical tail</td>
<td>553</td>
<td>1297</td>
<td>438</td>
<td>1285</td>
</tr>
</tbody>
</table>

Table 7.4: Individual weights, in kg

<table>
<thead>
<tr>
<th>Solver</th>
<th>X</th>
<th>Y</th>
<th>Z</th>
</tr>
</thead>
<tbody>
<tr>
<td>Standard</td>
<td>22.2395</td>
<td>0</td>
<td>-0.0742</td>
</tr>
<tr>
<td>Modify</td>
<td>21.5881</td>
<td>0</td>
<td>-0.0878</td>
</tr>
<tr>
<td>AcBuilder</td>
<td>20.69</td>
<td>0</td>
<td>-0.42</td>
</tr>
</tbody>
</table>

Table 7.5: CoG position in m, body axis reference defined in AcBuilder.

As mentioned before, the results for GUESS modify are more accurate, so the matrix of inertia (Table 7.6) and the complete weights provided are those calculated by this solver.
Table 7.6: Moments of inertia (kgm²)

<table>
<thead>
<tr>
<th>Mode</th>
<th>Frequency (Hz)</th>
<th>Mode</th>
<th>Frequency (Hz)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ixx</td>
<td>5.50 · 10⁵</td>
<td>Iyy</td>
<td>2.624 · 10⁵</td>
</tr>
<tr>
<td>Izz</td>
<td>5.50 · 10⁵</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Table 7.7: Danbus modes, first mass configuration

<table>
<thead>
<tr>
<th>Mode</th>
<th>Frequency (Hz)</th>
<th>Mode</th>
<th>Frequency (Hz)</th>
</tr>
</thead>
<tbody>
<tr>
<td>7</td>
<td>2.42</td>
<td>14</td>
<td>7.91</td>
</tr>
<tr>
<td>8</td>
<td>3.25</td>
<td>15</td>
<td>8.5</td>
</tr>
<tr>
<td>9</td>
<td>5.08</td>
<td>16</td>
<td>8.56</td>
</tr>
<tr>
<td>10</td>
<td>5.28</td>
<td>17</td>
<td>9.33</td>
</tr>
<tr>
<td>11</td>
<td>6.38</td>
<td>18</td>
<td>11.3</td>
</tr>
<tr>
<td>12</td>
<td>6.44</td>
<td>19</td>
<td>12.8</td>
</tr>
<tr>
<td>13</td>
<td>7.58</td>
<td>20</td>
<td>13.1</td>
</tr>
</tbody>
</table>

7.2 Modal and flutter analysis

The flutter analysis performed in NeoCASS provides the V-g plot representing the frequency and the damping of the structural modes along the velocity. With this data we can analyze the unstable modes in which flutter may appear. The analysis is set for M=0.9.

The V-g plot represents the evolution of frequency and damping with the increasing flight speed. There will be flutter if the damping of the system is positive for any of the modes, since it implies that the oscillations will increase and the system will become unstable. The first point will then be the flutter speed, as indicated in the plots.

7.2.1 First mass configuration

For the first mass configuration, that includes all of the fuel and payload mass, the following modes (table 7.7) and flutter diagram (figure 7.3) are obtained. The first six modes are ignored because they represent the rigid body modes.

Mode 12, represented in figure 7.4, is unstable at high speed. The flutter speed is found is 316.78 m/s, as appreciated in the V-g diagram. This unstable mode correspond to in-plane bending, that does not generate aerodynamic forces, limiting the flutter concerns.
### 7.2. MODAL AND FLUTTER ANALYSIS

![V-g plot](image1)

**Figure 7.3: V-g plot**

![Unstable mode](image2)

**Figure 7.4: Unstable mode**
7.2.2 Second mass configuration

The results regarding the second mass configuration, that is, full payload and central fuel tank but with the wing tank empty, are shown in table 7.8 and figure 7.5.

<table>
<thead>
<tr>
<th>Mode</th>
<th>Frequency (Hz)</th>
<th>Mode</th>
<th>Frequency (Hz)</th>
</tr>
</thead>
<tbody>
<tr>
<td>7</td>
<td>2.42</td>
<td>14</td>
<td>7.91</td>
</tr>
<tr>
<td>8</td>
<td>3.25</td>
<td>15</td>
<td>8.5</td>
</tr>
<tr>
<td>9</td>
<td>5.08</td>
<td>16</td>
<td>8.56</td>
</tr>
<tr>
<td>10</td>
<td>5.28</td>
<td>17</td>
<td>9.33</td>
</tr>
<tr>
<td>11</td>
<td>6.38</td>
<td>18</td>
<td>11.3</td>
</tr>
<tr>
<td>12</td>
<td>6.44</td>
<td>19</td>
<td>12.8</td>
</tr>
<tr>
<td>13</td>
<td>7.58</td>
<td>20</td>
<td>13.1</td>
</tr>
</tbody>
</table>

Table 7.8: Danbus modes, second mass configuration

![Figure 7.5: V-g plot for the second case](image)

The obtained modes are the same in both mass configurations, and the flutter speed that is computed is similar (313.57 m/s) to that in the first mass configuration.
In this case, mode 11 has unstable behavior. The corresponding modal shape is represented in figure 7.6, where we can see that, as in the previous analysis, this is an in-plane bending mode.

![Vibration mode 11 – Freq: 6.3767 Hz](image)

Figure 7.6: Unstable mode

### 7.2.3 Nastran flutter analysis

Guess model files have been converted to Nastran format in order to perform a comparison with the results NeoCASS provided.

<table>
<thead>
<tr>
<th>Mode</th>
<th>Frequency (Hz)</th>
<th>Mode</th>
<th>Frequency (Hz)</th>
</tr>
</thead>
<tbody>
<tr>
<td>7</td>
<td>2.42</td>
<td>14</td>
<td>8.11</td>
</tr>
<tr>
<td>8</td>
<td>4.15</td>
<td>15</td>
<td>8.62</td>
</tr>
<tr>
<td>9</td>
<td>5.45</td>
<td>16</td>
<td>9.29</td>
</tr>
<tr>
<td>10</td>
<td>5.46</td>
<td>17</td>
<td>11.1</td>
</tr>
<tr>
<td>11</td>
<td>6.45</td>
<td>18</td>
<td>13.0</td>
</tr>
<tr>
<td>12</td>
<td>6.68</td>
<td>19</td>
<td>13.8</td>
</tr>
<tr>
<td>13</td>
<td>7.64</td>
<td>20</td>
<td>14.8</td>
</tr>
</tbody>
</table>

Table 7.9: Danbus modes, Nastran
The modes extracted by Nastran (table 7.9) have a similar range of frequencies to those of NeoCASS. However, flutter analysis differs, since the Nastran calculations do not show any flutter instability in either configuration.

Figure 7.7: Nastran V-g plots

Considering the unstable modes in NeoCASS, they are close to zero at all speeds.
7.3 Aileron reversal speed

Analyzing the ratio between rigid and dynamic roll moment coefficients for different Mach numbers, the aileron reversal speed will be the velocity that makes the roll moment derivative equal to zero, being the ratio D/R zero as well. We will consider 5, 10 and 15-degree deflections for the aileron, so that the variation of the reversal speed with the deflection can be seen.

7.3.1 5 degree deflection

For the lowest deflection we are analyzing, the speed should be the highest of the three. When the aircraft is fully loaded, the reversal speed is found at $M=0.67$. The control derivatives regarding the aileron deflection are shown in table 7.11. The first row presents the lateral force coefficients, while the second row evaluates the roll moment. The roll coefficient is the one that determines aileron reversal.

<table>
<thead>
<tr>
<th>Configuration</th>
<th>Mode</th>
<th>Maximum</th>
<th>Minimum</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>12</td>
<td>-4.61e-04</td>
<td>-2.89e-03</td>
</tr>
<tr>
<td>2</td>
<td>11</td>
<td>-3.03e-04</td>
<td>-1.29e-03</td>
</tr>
</tbody>
</table>

Table 7.10: NeoCASS unstable modes, Nastran values

This same behavior can be observed in the Nastran V-g plots 7.7.

Nastran aerodynamic module disregards the airfoil camber considering a flat plate instead. However, NeoCASS considers the camber, so some aerodynamic contributions will be obtained in the in-plane bending modes, obtaining positive roots for the analyzed model. Nevertheless, the contributions to the aerodynamic forces provided by in-plane modes are not important, as can be appreciated in table 7.10 where their maximum and minimum values are disclosed.

Overall, disregarding the in-plane bending modes since flutter usually concerns out of plane bending and torsion modes, NeoCASS and Nastran provide a similar evolution of the tracked modes.
7.3.2 10 degree deflection

For the first mass configuration, the reversal speed is now found at $M=0.66$. The derivatives are disclosed in table 7.13.

<table>
<thead>
<tr>
<th>Rigid</th>
<th>Deformable</th>
<th>D/R</th>
</tr>
</thead>
<tbody>
<tr>
<td>$CS_{\delta a}$ -0.037463</td>
<td>$CS_{\delta a}$ -0.010521</td>
<td>$CS_{\delta a}$ 0.28084</td>
</tr>
<tr>
<td>$CL_{\delta a}$ 0.18612</td>
<td>$CL_{\delta a}$ -0.0028557</td>
<td>$CL_{\delta a}$ -0.015343</td>
</tr>
</tbody>
</table>

Table 7.13: $\delta_a$ derivatives, 10-degree deflection

The results for the second mass configuration at $M = 0.66$ are shown in table 7.14. The aileron reversal speed is the same in both mass configurations.

<table>
<thead>
<tr>
<th>Rigid</th>
<th>Deformable</th>
<th>D/R</th>
</tr>
</thead>
<tbody>
<tr>
<td>$CS_{\delta a}$ -0.037463</td>
<td>$CS_{\delta a}$ -0.014436</td>
<td>$CS_{\delta a}$ 0.38534</td>
</tr>
<tr>
<td>$CL_{\delta a}$ 0.18612</td>
<td>$CL_{\delta a}$ -0.003091</td>
<td>$CL_{\delta a}$ -0.016607</td>
</tr>
</tbody>
</table>

Table 7.14: $\delta_a$ derivatives, 10-degree deflection

7.3.3 15 degree deflection

For the first mass configuration, the aileron reversal speed is achieved at $M = 0.65$. The control derivatives for the aileron are shown in table 7.15.

For the second mass configuration the aileron reversal speed is $M=0.55$. In this configuration, the computed control derivatives are found below (table 7.16).

<table>
<thead>
<tr>
<th>Rigid</th>
<th>Deformable</th>
<th>D/R</th>
</tr>
</thead>
<tbody>
<tr>
<td>$CS_{\delta a}$ -0.037036</td>
<td>$CS_{\delta a}$ -0.016793</td>
<td>$CS_{\delta a}$ 0.28084</td>
</tr>
<tr>
<td>$CL_{\delta a}$ 0.18612</td>
<td>$CL_{\delta a}$ 0.0029151</td>
<td>$CL_{\delta a}$ -0.015343</td>
</tr>
</tbody>
</table>

Table 7.15: $\delta_a$ derivatives, 15-degree deflection
### 7.4 Divergence Speed

Divergence is found at $M = 0.91$ (Table 7.17), at sea level flight for the second mass configuration, the most restrictive condition. At $M = 0.91$, the divergence speed will be $309.4\text{m/s}$, above the flutter speed, that will be the most restrictive limit for the aircraft.

<table>
<thead>
<tr>
<th>Mach</th>
<th>Dynamic Pressure Pa</th>
<th>Divergence Speed m/s</th>
<th>Divergence Mach</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.60</td>
<td>408318</td>
<td>707.96</td>
<td>2.08</td>
</tr>
<tr>
<td>0.65</td>
<td>387868</td>
<td>683.98</td>
<td>2.01</td>
</tr>
<tr>
<td>0.70</td>
<td>364497</td>
<td>655.49</td>
<td>1.93</td>
</tr>
<tr>
<td>0.75</td>
<td>337596</td>
<td>621.09</td>
<td>1.83</td>
</tr>
<tr>
<td>0.80</td>
<td>306238</td>
<td>578.40</td>
<td>1.70</td>
</tr>
<tr>
<td>0.85</td>
<td>268869</td>
<td>523.01</td>
<td>1.53</td>
</tr>
<tr>
<td>0.90</td>
<td>222477</td>
<td>444.75</td>
<td>1.31</td>
</tr>
<tr>
<td>0.95</td>
<td>159371</td>
<td>307.85</td>
<td>0.91</td>
</tr>
</tbody>
</table>

Table 7.17: Divergence speed calculations. Danbus; $h=0$
Chapter 8

Analysis of results. Conclusions

In general, the results computed by NeoCASS are similar to those achieved by other calculation methods. This agreement can be seen in the first two models where NeoCASS is compared with a simple Matlab code and Nastran, a more complex finite element method. The differences in the wing model control reversal analysis are likely to have been caused by the modifications that had to be implemented in the GUESS output file.

Then, regarding the TCR analysis, all the data are consistent with the tests carried out in Milano both in NeoCASS and Nastran, including the control deformable derivatives for the aircraft.

Finally, the Danbus analysis is consistent with the Nastran flutter results. Therefore, taking into consideration the other analyses, the Danbus model is expected to provide a good first approximation of the final design.

To sum up, NeoCASS is a suitable tool for structural and aeroelastic analysis in preliminary design, as long as reliable structural models are used. Otherwise, the limitations of the models will be reflected upon the aeroelastic analysis. Static and dynamic aeroelastic analyses have been proved to be reliable.
Functions and subroutines in NeoCASS

Related to guess

- guess.m: runs guess standard model
- guess_mod.m: runs guess modify mode, main difference: calls solve_free_lin_trim_guess to compute the trim maneuvers used for the sizing

Related to smartcad: main solvers

- solve_eig: vibration modes (MSOL=103)
- solve_free_lin_trim: trim configuration (MSOL=144)
- solve_linflut: flutter solver (MSOL=145)
- solve_lin_aerostatic: Trim analysis, not launched by the gui (MSOL=144)
- solve_vlm_rigid: vlm/rigid solver (MSOL=700)

All of the above but the lin_aerostatic can be called from the GUI in the panel RUN, the enabled buttons corresponding to the solver code read in the input file (MSOL=103) / beam_model structure (beam_model.Param.MSOL=103) Both aerostatic and trim have the same code, the difference is the analysis conditions: free lin trim is used for free flying aircraft, aerostatic for some clamped examples

Internal solvers: not called by the user interface

- solve_buckling (MSOL=105)
- solve_defo_mean_axes
- solve_dynder (MSOL=701)
- solve_lin_static (MSOL=101)
- solve_nlin_aerostatic (MSOL = 644)
- `solve_nlin_static (MSOL = 600)`
- `solve_free_lin_trim_guess`: Called internally when running guess modify

### Output and plotting functions

- `plot_beam_defo(nfig, scale, set)`: plots structural and aerodynamic models showing the deformed aircraft (trim config)
- `plot_beam_model (nfig)`: plots the model
- `plot_beam_forces(nfig, scale, set)`
- `plot_control_aelink(nfig)`: plots the link between the aerodynamic control surfaces
- `AI_plot(ifig,IND)`: plots the stresses on the beam.
  - IND=1 local $T_x$
  - IND=2 local $T_y$
  - IND=3 local $T_z$
  - IND=4 local $M_x$
  - IND=5 local $M_y$
  - IND=6 local $M_z$
- `plot_cp_surf(scale, def) plots $C_p$`

### Other

- `load_nastran_model('filename')`: loads a smartcad file and creates the beam_model structure from the input file.
- `set_neocass_path`: required to run the program, sets the path.
- `NeoCASS`: launches the GUI
- `Report_Stab_der`: produces a tex file with the desired fields, particularly stability and control derivatives with the following command:

  ```
  Report_Stab_der(['filename'], ...
  ```

- `Convert('input_filename.dat','output_filename.dat')`: Converts a GUESS model to Nastran format.
Data structure

**Beam_model**

**Info** Card Counters, General info and parameters

**Param** Model parameters

**Coord** Reference frames data

**Node** Node data

**Mat** Material properties

**Bar** Bar data

**PBar** Bar property data

**Beam** Beam data

**PBeam** Beam property data

**F** Applied forces data

**M** Applied moments data

**F_FLW** Applied follower forces data

**ConM** Concentrated Mass data

**WB** Model Weight and balance data

**SPC** Model constraints

**Aero** Aerodynamic state

**Optim** Optimization parameters

**Celas**

**RBE2**

**Gust**
SET
Surfdef
Res Results

**Beam_model.Res**
SOL Solver used
FM [1x1 structure]
CS
State
Bar Bar results
  - CForces
  - CStrains
  - CStresses
  - CSM:
    - Tmax_Norm
    - Tmin_Norm
    - SM_Norm
    - SM_Buck
    - Tmax_Shear
    - Tmin_Shear
    - SM_Shear
    - SM_PBuck
  - R
  - Colloc

Beam
NDisp
NRd
Aero Aerodynamic results
  - RStab_Der: [1x1 structure] Rigid Stability and Control derivatives
  - RIntercept: [1x1 structure]
  - DStab_Der: [1x1 structure] Deformable Stability and Control derivatives
- DIntercept: [1x1 structure]
- RTrim_sol: [1x1 structure] Rigid Trim solution
- DTrim_sol: [1x1 structure] Deformable trim solution
- DIVERG_Q: Divergence dynamic pressure
- Qaa: [1022x1022 double] Aerodynamic transfer matrix
- Ka: [1022x10 double]
- Fa0: [1044x1 double]

**WB** Weight & Balance data

- CG: CG position
- MCG [6x6 double] Inertia matrix
- MRP Reference change matrix
- MCG_pa Inertia matrix, diagonal
- R_pa Matrix used for the reference change

**CPaero**

**Gamma**
Bibliography


