EFFICIENT AEROELASTIC SIMULATION IN A PARAMETRIC PROCEDURE FOR FATIGUE ANALYSIS

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Abstract

A parametric procedure is described that provides a way to efficiently calculate flight loads on flexible aircraft subject to variations in configuration, flight conditions and operational usage. The procedure centers around a flight loads database, which has been designed specifically for the generation of loads spectra for fatigue analysis. The parametric procedure is generic in nature; thus, it is in principle applicable to any type of aircraft. For the sake of developing and delivering a proof-of-concept of the procedure, the application has been dedicated to the F-16 combat aircraft. The role of efficient aeroelastic simulation in order to fill the loads database with flight loads for a vast number of conditions is described, taking into account the differences in flight conditions, store configurations, fuel distribution, flap settings, and so on. Validation and application of the developed procedure is presented for selected cases.

1 Introduction

Aircraft are designed for a specific fatigue life. Based on anticipated operational usage, the structural design is dedicated to fulfill the objective of the fatigue life. Monitoring aircraft loads during operational use is highly important for the determination of the actual fatigue life of aircraft [1][3][4][11][12]. Especially for multi-role combat aircraft, the continuously changing operational scenery gives rise to significant variations in the aircraft loads encountered. In most cases, the actual loads spectrum of military aircraft deviates significantly from the design loads spectrum. The deviations can be attributed to changes in the types of missions flown, in the types of stores carried, and in repairs and updates applied to the aircraft. The deviations in the loads spectrum, relative to the design loads spectrum, will impact the remaining fatigue life of the aircraft to a high degree. To keep up with the actual usage of the aircraft, the maintenance plan has to be continuously revised. Input for such a revision is commonly based on measured data, i.e. from post-flight analysis of flown missions, using data obtained at limited locations in the aircraft structure [12][13]. Such data are commonly used to determine relevant inspection intervals and maintenance cycles to service the aircraft.

For the prediction of the impact of changes in operational missions and configurations to the remaining fatigue life, a reliable tool is of paramount importance. Such a tool enables to plan and optimize future changes in missions and configurations prior to implementation, with the objective to limit the reduction in remaining fatigue life of the fleet of aircraft to acceptable values.

In this paper, the results of a study to develop a reliable tool for the analysis of the fatigue life of aircraft are presented. The objective of the study is to develop an analytical procedure, including methods and models, which can be routinely applied to analyze fatigue life variations due to the changing operational role of the aircraft. The study is centered on the capability to generate a fatigue loads sequence at critical locations of the aircraft structure for specified operational usage. Providing a proof-of-concept of the developed parametric procedure is also part of this study. The parametric procedure that has been developed to predict the fatigue life of aircraft is described in section 2. The prominent role of efficient aeroelastic simulations within this procedure, which is the focus of the present paper, is further detailed in section 3. Examples of the application of the aeroelastic simulation tools, showing the proof-of-concept of the approach, are given in section 4.

2 Parametric procedure for fatigue analysis

2.1 Overview of the generic system

The functional diagram of the process to generate load sequences for operational aircraft usage is shown in Fig. 1. Starting point for the whole process is the input describing the aircraft operational usage. This input has to be discretized into missions, segments, statistics of occurrences of maneuvers, etc. In principle, after identification of the required critical load cases from the aircraft operational usage, a fatigue analysis can be based on a full aeroelastic simulation to get the critical loads on the flexible aircraft for each of the load cases. Such an approach is however not practical from a turn-around time perspective.

Instead of computing the loads directly on the flexible aircraft during critical load cases, a parametric procedure is used. The procedure makes use of a central loads database with predefined support points. The loads are parameterized in such a way that interpolation in the database will generate the actual loads with a specified, required accuracy. As shown in Fig. 1, two branches can be identified departing from the operational aircraft usage.

The branch indicated with green arrows in the parametric procedure provides a generic means to calculate flight loads on the flexible aircraft for a sequence of fatigue load cases. The sequence is obtained by the Computer-aided Loads and Stress Sequencing (CLASS) algorithm developed by NLR [5], based on mission input and occurrences per segment. Thus, the critical loads for a vast number of load cases are obtained in a very efficient way, and a fatigue analysis can be performed using these loads. Setting up of the database for this specific purpose is detailed in section 2.2. The usage of the database for fatigue analysis is explained in section 2.3.

The branch indicated with yellow arrows represents the process of building the database using flight mechanics mission analysis to identify critical conditions and aeroelastic tools to determine the critical loads at these conditions. In order to build up the database in an efficient manner, smart combinations of available aeroelastic tools and models have been applied to allow filling a significant part of the database within acceptable turn-around times. In this approach, the computation of external loads, defined here as the aerodynamic and inertial forces and moments, have been separated from the calculation of the internal loads, defined here as the stresses in the aircraft structure. The part indicated by the yellow arrows is described in detail in section 3.

Due to the separation of the computations of internal and external loads, a loads transfer mechanism is required. The loads transfer mechanism in the present study is based on an independent layer called the Neutral Interface (NI). This layer is used to collect external loads data and to redistribute the external loads to the internal loads locations, as will be explained in more detail in section 3.

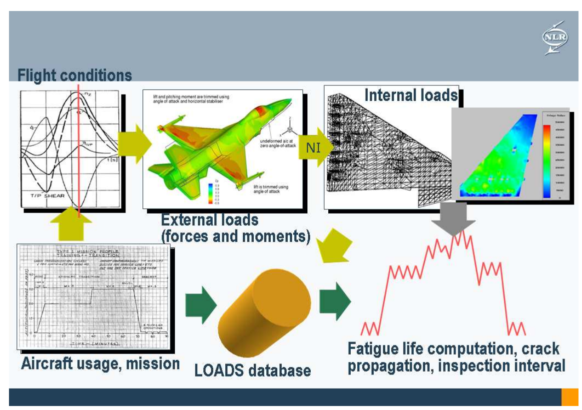

Fig. 1. Functional diagram of the fatigue loads generation system. Green arrows imply operational usage of the procedure for fatigue loads. Yellow arrows indicate build-up of the database.

2.2 Description of the loads database

The loads database is the central part of the generic system for fatigue analysis, containing all the necessary loads for fatigue spectra generation [10]. For adequate storage, the choice has been made to separate the loads into mean and incremental loads for a representative aircraft of the fleet. Both external and internal loads are stored. The database is valid for one specific aircraft type only (e.g. F-16, KDC10, C130, etc.), and for one possible configuration only in terms of weight and balance distribution.

The database has been designed to enable fast and efficient interpolation of loads towards the desired conditions. The stored data points in the database are a smart selection of all possible operational conditions, called points-in-the-sky (PITS). The density and distribution of the PITS are selected in such a way as to span the entire flight and ground envelope with the purpose to be able to obtain sufficiently accurate loads at all interpolated conditions.

In the present study, the major parameters describing the PITS are: Mach number, internal stores weight (i.e. cargo, in case of transport aircraft), external stores weight, fuel weight and altitude. For all the predefined PITS, loads analyses are performed and the results are stored in the database.

The actual magnitude of the loads at a specified flight condition are a function of the aircraft configuration in terms of flap settings, thrust settings, store configuration, landing gear position, type of maneuver, fuel weight and distribution, etc. At one operational PITS, the loads will therefore be different for different flap settings. As a result, for every possible combination of flap settings, landing gear position, and so on, the loads need to be calculated and stored in separate tables in the database. The data tables contain all loads per operational point at various locations in the aircraft. An aircraft location is a geometrical point somewhere in the aircraft structure where the external loads data are available from the loads analysis, i.e. the NI-point. In Fig. 2, a schematic overview of the loads database is given. Obviously, the loads on the aircraft structure depend on the location and therefore the loads database will contain loads per location. At each location, the loads are given as six components, i.e. three force contributions and three moment contributions, one for each of the spatial coordinate directions.

For each of the possible loads sources, i.e. maneuver, gust, limit cycle oscillation (LCO), etc., the data tables contain the mean and incremental loads for each NI-point at each operational grid PITS. Depending on the loads source, different information is stored in the database in order to be able to reproduce the essential features of each loads source. For all loads sources, the aerodynamic and inertial contributions to the total flight loads are stored separately.

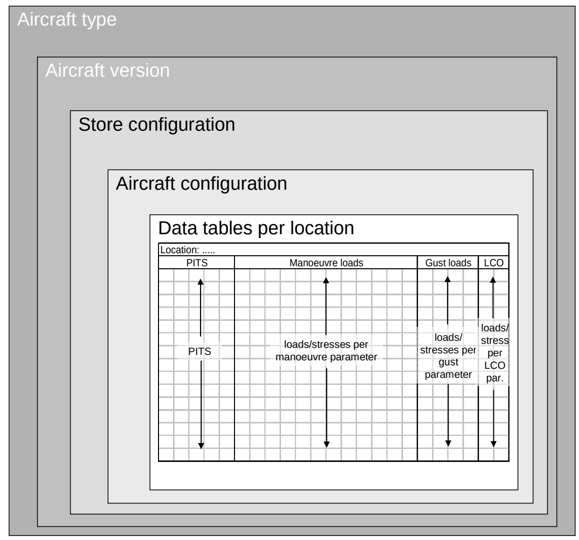

Fig. 2. Schematic set-up of the loads database for fatigue analysis, containing loads per neutral interface point.

The internal loads, or stresses, for specific critical locations in the aircraft are obtained by a stress computation module. This module takes the stress-to-loads ratios (SLR) as well as the mean and incremental external loads as input. The external loads are converted to a set of mean and incremental stress tensors for all the different load sources at various critical locations in the aircraft structure and at PITS grid points. The results are stored in the stress database. The stress database contains the same information as the external loads database, only now expressed in terms of stresses instead of forces and moments. The main difference is that the external loads database contains loads for the complete aircraft while the stress database contains stresses only at critical locations of the aircraft structure. The stress database is used as the starting point for the stress spectra generation program.

2.3 Usage of database for fatigue analysis

This description deals with the path indicated by the green arrows in Fig. 1. Before a fatigue analysis can be performed, time histories or sequences of the loads representing the aircraft usage and the loads environment have to be defined. In addition to requiring knowledge about the magnitude of the loads, a fatigue analysis also requires statistical input on the number of occurrences each load case is applied during the period of the analysis. This information is defined by the combination of the following two items:

- usage, specifying in detail how the aircraft is being used;
- criteria, specifying the rate of occurrence of loads per unit time for a given flight or ground phase.

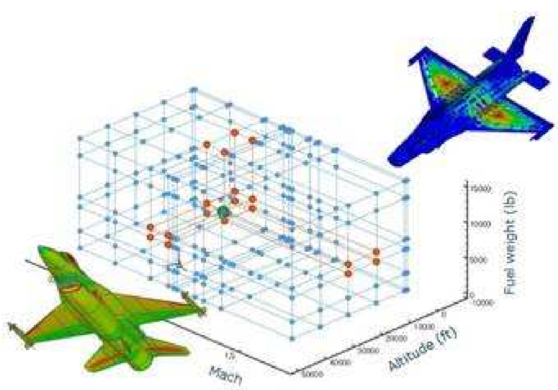

Fig. 3. Aircraft loads database visualized for three parameters Mach, altitude and fuel weight; bullets representing points-in-the-sky (PITS).

For the description of the aircraft usage, mission profiles are typically being used. A mission profile is defined as a collection of mission phases that chronologically define the operational parameters, i.e. the PITS, versus the duration of the complete flight. The set of mission profiles represent how the aircraft is used during a period of time for which the fatigue life analysis is made. The aircraft loads response is a direct function of the operational parameters during the mission profiles.

Most of the time, the operational PITS of a mission profile do not coincide with the PITS stored in the database. This is in part due to the choices made for a limited total number and distribution of the defining PITS in the loads database over the entire flight and ground envelope, and in part due to the vast number of possible mission phases. As a result, loads data have to be interpolated from the surrounding PITS in the database to the specified flight condition, see Fig. 3. The interpolation routine is a core component of the loads database extraction procedure. Due to the large number of variables that make up a PITS (like fuel weight, altitude, airspeed, etc.), a multi-dimensional interpolation is being used. It is assumed that the loads are linear with respect to the chosen PITS parameters. The selection of the PITS grid forming the support points of the database is therefore critical and needs to encompass all specific aircraft features during operational usage. For example, the weight of a combat aircraft is defined by its store configuration which is discrete, whereas the onboard fuel is consumed continuously in a specific manner. Also, for the distribution of the PITS over the entire Mach number range, it is important to take the peculiarities of the transonic region into account.

Using the information from the input on usage and occurrence criteria, loads spectra in the form of exceedance curves can be generated and, using the sequence generator CLASS [5], the loads over the mission segments can be distributed to form a time history sequence.

3 The role of efficient aeroelastic simulation

3.1 Loads database extent

The loads database represents a huge amount of loads information, parameterized in PITS space. Currently, for a selected configuration of a combat aircraft, a PITS is represented by the main variables Mach number, altitude, and fuel weight. Per PITS, a dozen subcases are defined by storing the loads for several load factors ranging between its minimum and maximum values. Furthermore, per load factor, in addition to a symmetric pull-up maneuver, loads due to rolling maneuvers are stored for two different roll rates. Finally, the information per PITS is complemented with loads information for a specified pitch and roll acceleration. Thus, for filling a minor part of the entire database, i.e. representing the flight loads for one selected configuration over the entire flight envelope, 4500 cases are easily required to be filled.

On the basis of all the stored information per PITS, the required loads for a specific flight condition as occurring during an operational mission are easily obtained by applying a multi-dimensional interpolation approach. Note that the stored loads information is distributed over the complete aircraft in the loads points as defined by the neutral interface.

In the following, considerations are given for filling the loads database in case no representative wind tunnel or flight test data are available, leaving simulation of the loads per PITS as the only option.

3.2 Accurate simulation

For determining external loads with sufficient accuracy, the application of Computational Fluid Dynamics (CFD) in static aeroelastic simulations, coupled with a linear structural model, is currently the state-of-the-art [6][7], see also Fig. 4. In this approach, trimming of the aircraft for the required flight condition, including the influence of the flight control system (FCS), is part of the loads generation process resulting in balanced loads. The accuracy of the loads obtained in this way is usually very acceptable, especially if viscous flow solvers are used. However, this approach demands significant effort in terms of turn-around times and preprocessing activities. A relatively high degree of user intervention is usually required. Even for simulations based on the Euler equations, the effort amounts to very many computing and preprocessing hours, as will be shown in section 3.4.

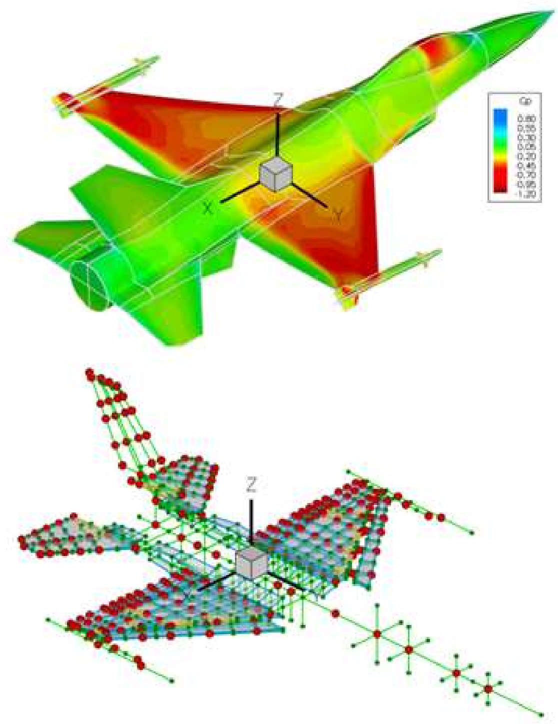

Fig. 4. Examples of an aerodynamic model (top) and a structural dynamic model (bottom) as used during aeroelastic analyses to obtain flight loads

3.3 Fast simulation

By applying mainstream aerodynamic methods, specifically the more traditional linear doublet lattice methods for static aeroelastic simulations coupled with a linear structural model, largely reduced turn-around times are obtained. The price for the improved turn-around times is however a reduced accuracy. For certain areas of the flight envelope, these methods work very well though.

Especially in the transonic regime and at the extremes of the flight envelope, the loads obtained using linear methods are less reliable as shown in previous work [7]. In Fig. 5, the spanwise loads, viz. bending moment and torsion moment, are depicted as obtained with linear as well as CFD-methods. It is found that for this transonic flight condition the wing bending moment is obtained with acceptable accuracy, even when using linear methods. For the wing torsion moment, however, the situation is completely different. The actual pressure distribution on the wing upper surface contains regions of locally supersonic flow and embedded strong shock waves. The transonic pressure distribution generates significantly different chordwise loads compared to those predicted by linear methods. As a result, the torsion moment as obtained with linear methods is an outlier in the set of results. The CFD-approach is better capable of reproducing the net loads as used by the aircraft manufacturer.

For conditions where the reduction in accuracy of the loads obtained with linear methods is too significant to be acceptable, a smart combination of linear and nonlinear aerodynamic methods is a promising alternative to conserve both loads accuracy and acceptable turn-around times.

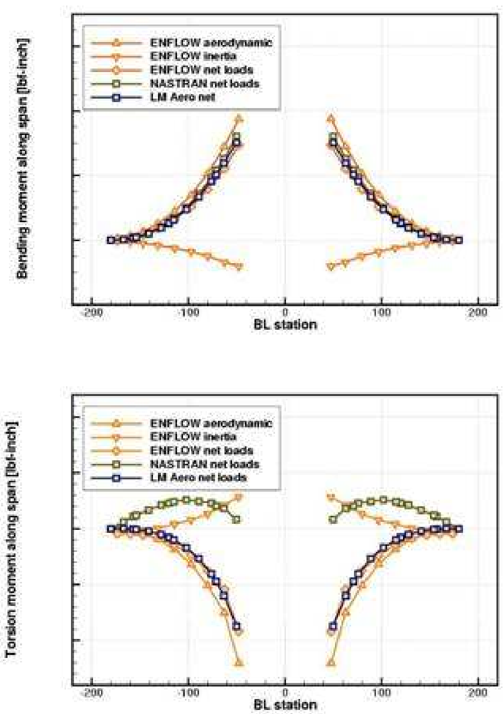

Fig. 5. Comparison of loads analysis results employing linear aerodynamic methods (NASTRAN) and a CFD method (ENFLOW) for an F-16 in a symmetric pull-up maneuver at Mach 0.9, showing the deficiency of linear aerodynamic methods to capture the nose-down moment in transonic flow

3.4 Smart combination of methods

Efficiency in filling the loads database is of major importance to enable studies of various mission scenarios and store configurations. When a loads database needs to be filled for multiple configurations over the entire flight envelope, the total effort can amount to tens of thousands of different load cases. In the current study, an efficient loads computation procedure is obtained through smart combinations of methods.

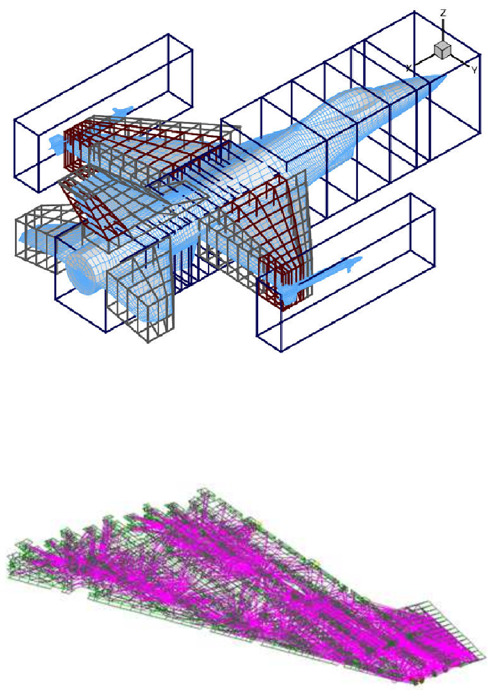

Fig. 6. Definition of neutral interface (top) to extract loads data from various loads models, and mapping of loads from neutral interface to a finite element model for stress analysis employing beam elements (bottom)

To facilitate the combination of methods the following have been applied:

All possible flight conditions have been defined in terms of a limited set of variables. All other details of the maneuvers are discarded. Thus, the loads at a PITS are composed of contributions due to a discrete set of variables comprising Mach number, altitude, fuel weight, load factor, and manoeuvering rates and accelerations.

Standardized usage of the neutral interface for loads transfer to the structural model has been applied. The neutral interface needs to be defined once. All external loads are collected in the points of the neutral interface, and redistributed to the loads points of the structural model in an a priori defined manner, see Fig. 6. A well-defined neutral interface implies a reliable and routinely loads transfer to the structural model.

Calculation of internal loads (stresses) using a sufficiently fine structural finite element model (FEM, see Fig. 7) for each PITS and its subcases requires lengthy evaluations. Usually, the FEM-model is linear, discarding its applicability when plasticity is involved. Under this condition, internal loads for an actual flight condition are a multiple of the internal loads resulting from unit external loads acting on the loads points of the FEM-model. Usage of stress-to-loads ratios (SLRs) representing the stresses due to unit external loads on the load points of the FEM-model allow for a quick and routinely calculation of stresses. The SLRs only need to be evaluated once for a given FEM-model and neutral interface.

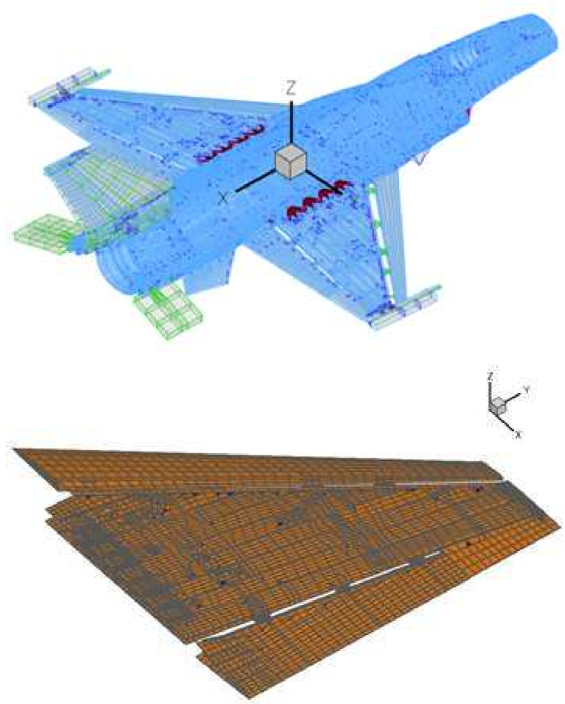

Fig. 7. Finite element model for the computation of internal loads, i.e. stresses, in the aircraft structure, showing full FEM (top) and wing details (bottom)

Combining accurate and mainstream methods to obtain the loads data over the entire flight envelope is possible in the following ways, ordered for increasing computing time and accuracy:

1. Apply the mainstream method for all PITS over the whole flight envelope.
2. Apply the accurate method for a rigid aircraft at each Mach number and apply the mainstream method to complete the loads computation by taking care of the remaining parameters: trim and flexibility effects at level flight, altitude, fuel weight, load factor, roll rate, pitch rate, roll acceleration and pitch acceleration.
3. Apply the accurate method for the level flight condition at each Mach number and apply the mainstream method to complete the loads computation. This approach is almost similar to the second one except that the flexibility and correct trim state are now included in the level flight loads evaluation.
4. Apply the accurate method for various load factors at each Mach number and apply the mainstream method to complete the loads computation, taking care of the remaining parameters: altitude, fuel weight, roll rate, pitch rate, roll acceleration and pitch acceleration.
5. Apply the accurate method for all PITS.

The actual selection of the required approach is made by the man-in-the-loop. In specific regions of the flight envelope, CFD-methods are mandatory to obtain accurate baseline loads, e.g. in the transonic regime. The effects caused by the presence of shockwaves and their inherent influence on chordwise loads distribution and wing twist, needs to be correctly obtained using CFD. Therefore at least approach number 2 has to be applied. Other variations, e.g. due to altitude or fuel weight, can be obtained either by CFD or by linear methods without compromising the accuracy to a too large extent.

The application of the above described approach in the loads generation process has improved the efficiency of the loads generation process significantly. It has been possible to fill 6000 cases in the database within a couple of days. For an overview of turn-around times using different approaches, see Table 1.

Table 1. Estimated turn-around times for a set of 6000 PITS, on the basis of turn-around times for single condition calculations using different flow models

Loads calculation approach (see text)	Estimated turn-around time (hours)
1 (mainstream)	50
2 (Euler)	53
3 (Euler)	54
4 (Euler)	70
5 (Euler)	2400
2 (Navier-Stokes)	80
3 (Navier-Stokes)	90
4 (Navier-Stokes)	250
5 (Navier-Stokes)	24000

4 Proof-of-concept and application example

4.1 Proof-of-concept of parametric procedure

The F-16 was originally designed as a light, highly maneuverable aircraft. Over the years, however, its role has changed to a multi-tasking combat aircraft. Therefore, the F-16 airframe has been subject to its design loads at a higher number of occurrences and at a higher rate than originally predicted. Based on many years of experience in monitoring the airframe including strain registrations for a variety of light and heavy store configurations, there is a large database of in-flight information available for validation purposes.

The proof-of-concept of the above described parametric flight loads procedure has been established on the basis of comparisons with flight test data. Due to the presence of the Fatigue Analysis and Combat Evaluation (FACE) data registration system in the Royal Netherlands Air Force (RNLAF) F-16 aircraft, stress histories at several locations in the structure are recorded, along with the flight condition data. The whole process of flight loads computations, i.e. the parameterization of flight data, the external loads calculations, the internal loads computations and the database interpolation, can be verified against recorded data. One complication arises in this process, however, since the finite element model has been developed to represent global stresses instead of detailed local stresses. Careful inspection of the available data from the FACE registrations and the local modelling at the strain gauge locations has led to the selection of two locations in the aircraft structure for verification purposes, one location at the wing root and one at the horizontal tail. Stresses in the direction corresponding to the alignment of the strain gauges have been obtained using the developed database, and these stresses are compared with FACE registrations, see Fig. 8. The agreement between the reconstructed stress sequence, using the full loads procedure, and the actual flight data is very good. Some part of this flight takes place at supersonic speed. In view of the different behaviour of the flight control system in the subsonic and supersonic regimes, the good comparison at the horizontal tail also implies correct modelling of the flight control system.

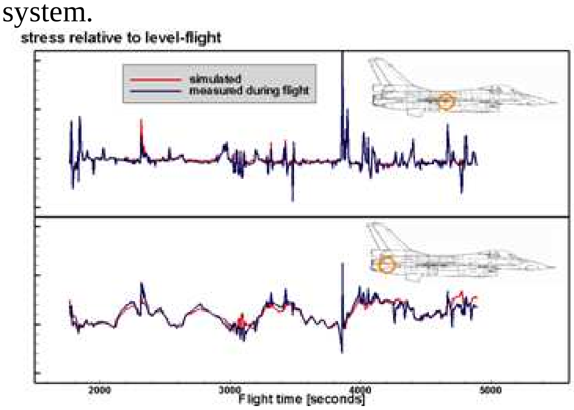

Fig. 8. Comparison of computed and measured stress sequences – relative to level flight – at two locations representing loads at the wing root and at the horizontal tail plane of the F-16 during a mission

4.2 Application example of the procedure

The parametric procedure has been designed specifically for fatigue life analysis. In this section, the application of the procedure to estimate the impact of limit cycle oscillation (LCO) on crack propagation is outlined. The application is hypothetical in nature since it is based on one type of mission only, a “training-transition” flight. A block of flights is subsequently defined as a randomized mix of 17 flights.

The quantitative effect of LCO on the fatigue life of the structure is performed by comparing two F-16 aircraft in heavy store configuration. One of the aircraft endures LCO at a Mach number of 0.9 and altitude of 10,000 feet during its training mission flight, while the other does not. The dominant LCO-frequency for this configuration is 5 Hertz. It is assumed that the F-16 experiences one minute of LCO within a 90 minutes flight. This flight occurs 17 times within a block. The present study is based on the crack propagation at an aircraft structural integrity program (ASIP) point located on the wing. It is assumed that crack initiation has occurred.

For this application, the sequence of stresses at this location is reconstructed based on the external loads experienced by the aircraft during the flight. Pre-computed LCO-loads are superimposed to the nominal loads due to the maneuver. In studies concerning LCO [2][8][9], the severity of the LCO is usually expressed in terms of peak-to-peak amplitude at the forward part of the wing tip missile launcher. The way in which LCO-loads are stored in the database is based on normalization to 1g peak-to-peak accelerations at this location. In the present example, a 5g peak-to-peak LCO at the tip launcher is assumed, which equals a 0.4g vertical acceleration at the center of gravity. This value is quite severe, almost reaching the standard abort criterion.

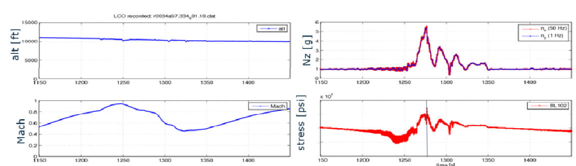

Fig. 9. Example of limit cycle oscillation (LCO) as observed during a flight test, from [13]

The sequences of stresses at the selected ASIP-point for flights with and without LCO are shown in Fig. 10. The differences can also be seen in the exceedance curve, given in Fig. 11. The LCO-loads contribute to the overall loads with the characteristics of repeated loads having a relatively low stress level and a relatively high number of occurrences. From a crack growth analysis, as shown in Fig. 12, the estimated effect of the LCO as assumed in the defined flights is a reduction of about 4 per cent of the fatigue life of the F-16, based on the information at the selected ASIP-point on the wing. It is found that the effect of LCO on the total fatigue life appears to be limited for this specific hypothetical case. More in-depth study is needed, however, to quantify the effect of LCO on fatigue life.

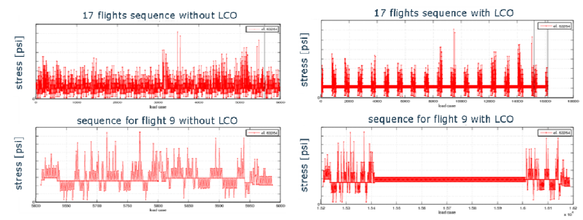

Fig. 10. Comparison of block (top) and flight (bottom) stress sequences with and without LCO

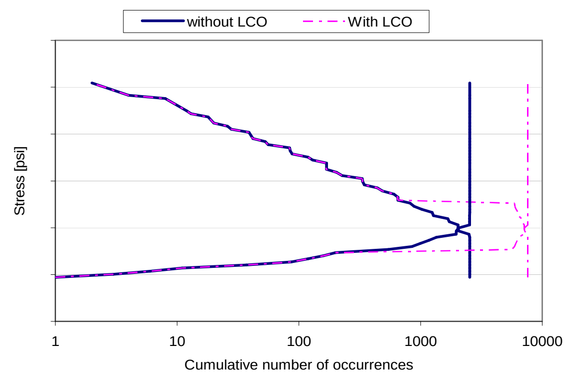

Fig. 11. Exceedance curve of the sequence of 17 flights showing the scale of the LCO cycles

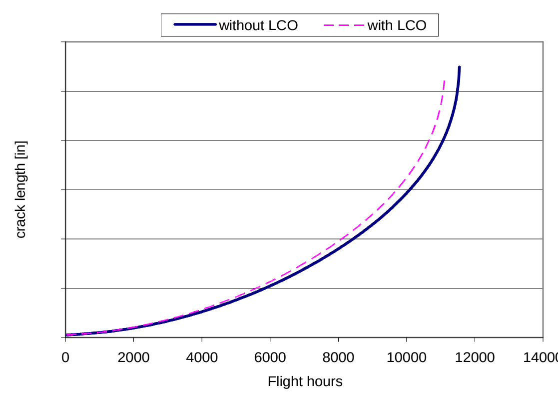

Fig. 12. Impact of LCO on the crack growth curve of an F-16 ASIP location, derived using the reported procedure

5 Conclusions

A generic parametric procedure has been developed to predict the impact of operational usage on the fatigue life of structural parts in aircraft. The parametric procedure allows transforming of loads due to maneuvers and gusts into loads sequences for crack growth analysis. A limited part of the loads database, forming the core of the parametric procedure, has been filled with 6000 load cases for an operational F-16 configuration using a simulation approach based on models for aerodynamics, structural dynamics, and the flight control system. A smart combination of mainstream and CFD-based aerodynamic methods has been applied to keep simulation times for a large number of load cases to a minimum without compromising the accuracy of the resulting loads. Proof-of-concept of the parametric approach has been shown by comparison of computed stress sequences at two locations in the airframe with measured flight data, showing very good agreement. The parametric procedure and its underlying loads generation methodology can be applied on a routinely basis to predict loads for specific configurations and flight conditions, and is suitable for the generation of load sequences for fatigue analyses.

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