

Aeroservoelastic Design and Verification of a combat Aircraft

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Abstract

The design strategy of the advanced flight control system development is important through the integrated design optimization process, which includes besides the modeling of the coupled system of the flight dynamics, also the structural dynamics, the actuators of the control surfaces and the sensors as well as the effects of the digital control systems.

Results from structural mode coupling investigations are presented which describe the design, test verification and clearance procedure of an advanced digital flight control system. The advantages of an integrated interdisciplinary flight control system design, on the basis of the coupled dynamic model which is derived from structural dynamics model and the flight dynamics model of the aircraft, is presented.

Different examples demonstrate the advantages of the integrated, interdisciplinary design. Analytical and experimental methods to avoid structural mode and flight interaction are described. Especially the design of filters to minimize interaction is outlined, using a mathematical model of the elastic aircraft which describes the coupled flight dynamics, flight control dynamics and the structural dynamics behavior on ground and in-flight structural mode coupling tests. The paper explains design procedures, design and clearance requirements, test procedures and the correlation between mathematical model predictions and structural coupling tests as well as the aeroservoelastic model update on ground and in flight.

Introduction

The development of advanced digital Flight Control Systems (FCS) for a modern military aircraft is strongly influenced by aeroservoelastic effects. The flexible aircraft behaviour especially for artificial unstable aircraft configurations with outer wing missiles, tip pods and heavy under wing stores and tanks has significant effects on the flight control system. The signals of the Aircraft Inertia Measuring Unit (IMU) - the gyro platform - contain besides the necessary information of rigid aircraft rates and accelerations also flexible aircraft rates and accelerations in the frequencies of the aircraft elastic modes. The 'flexible' aircraft rates and accelerations measured by the IMU are passed through the flight control system control paths, they are multiplied by the FCS gains and FCS filters and inserted in the control surface actuator input which then drives the controls in the frequencies of the elastic modes of the

aircraft. The flexible aircraft is excited by the high frequency control deflections and might also experience aeroservoelastic instabilities i.e. flutter or limit cycle oscillations may occur and dynamic load and fatigue load problems can arise. The FCS design therefore has to minimize all structural coupling effects through the available means like optimum sensor positioning, notch filtering and additional active control. This paper describes the major aspects, problem areas to be considered in the FCS design with respect to aeroservoelastic effects, it outlines an integrated design of FCS gains and phase advance filters together with notch filters. The integrated design was followed, since independent design of notch filters or FCS has created many problems for stabilization of rigid aircraft or elastic modes. Many of the design and clearance aspects described here have been addressed in previous publications, Ref.'s ⁽¹⁾ to ⁽¹⁰⁾.

Integrated Design for Advanced Flight Control Systems, Design Philosophy

Integrated design shall include the derivation of FCS gains, phase advance filters and notch filters to minimize structural coupling in one combined optimization process. The FCS shall be designed to cover the full rigid, flexible aircraft frequency range with respect to aircraft rigid mode and structural mode coupling stability requirements for each control system individual loop for on ground and in flight. The structural coupling influences shall be minimized by FCS notch filters. The FCS shall be designed to be as robust as possible with respect to all possible aircraft configurations and configuration changes, (missiles on, off, tanks on and off etc.). That includes that all structural coupling changes with configuration should be covered by a constant set of notch filters to avoid system complexity due to configuration switches for different sets of notch filters. In addition any scheduling of notch filters with flight conditions should be avoided in a wide range of the flight envelope but not excluded for critical structural coupling areas. In order to avoid problems in the notch filter design due to non-linear elastic mode and unsteady control surface aerodynamics and non-linear actuator dynamics the elastic mode stability requirements should mainly be based on gain stabilization of the flexible modes. Phase stabilization shall only be applied to low frequency elastic modes in order not to create too complex design and clearance procedures. Phase stabilization of low frequency elastic modes might not be avoided; it is used as tool to meet handling requirements.

The notch filter design can be based upon an analytical model of the aircraft structure including a linear FCS model. The analytical model must however be verified through ground test results both from ground resonance and structural coupling testing and from in flight flutter and structural coupling testing. The model should be updated by the test results for different configurations. Due to restrictions in the accuracy of the analytical model predictions on ground and in flight mainly at high frequency elastic modes where the prediction becomes more and more unrealistic the analytical model data with respect to inertia shall be replaced by on ground measured data. In order to cover all possible sets of aircraft store configurations a selection of critical configuration has to be established by analytical model investigation in advance.

The most critical selected configurations have to be introduced into the design of the structural filters.

The integrated FCS gain, phase advance filter and notch filter design shall cover the full range of stores and fuel states for the absolute worst case of FCS gain for trimmed aircraft conditions and shall also take into account worst gain situations in out of trim conditions.

Design Requirements

The design requirements are primarily stability requirements for all flight rigid and flexible aircraft modes. The stability of the structure will be achieved by the introduction of structural filters (Notch filters). Different criteria's were be used for prototype and production aircraft. In principal the low frequencies of the flexible modes are phase stabilized whereas the higher frequency flexible modes are gain stabilized. The Military Specification MIL-F-9440D for FCS requirements shall be met.

In addition to the stability requirements for the structural coupling unacceptable high vibration levels must be avoided by including vibroacoustic and noise levels. It is known from other aircraft that vibration levels induced by structural coupling might create high fatigue loads on actuators and the aircraft structure itself. Notch Filters with noise filters have to be designed to meet the vibration requirements.

Beside this the FCS design for elastic structural mode coupling reduction with structural filters has to fulfil the flutter requirement of the aircraft including FCS. The additional 15% flutter safety margin as well as the minimum elastic mode damping requirements of 3% g structural damping as described in MIL-A-8870C shall be met.

The integrated FCS design for the flexible aircraft is possible with the assumption that the aircraft characteristics are predictable to the necessary accuracy to optimize notch filters which meet the requirements. The characteristics of the controlled flexible aircraft shall be described in the form of open loop frequency transfer functions of the FCS control path feedback loops to a sufficient high frequency, see block diagram in Fig. 1. In detail for the longitudinal control system the pitch rate, the normal acceleration and the flow sensor α open loop signal at the control opening point has to be known. For the lateral control the roll rate -, yaw rate-, lateral acceleration - and flow sensor signal β open loop signal has to be described. The open loop signal consists of the transfer function of the aircraft due to control surface input sensed at the inertia measuring unit (rates and accelerations) and at the flow sensors, and the transfer function of the FCS from the sensor to the opening point and from the opening point to the actuators.

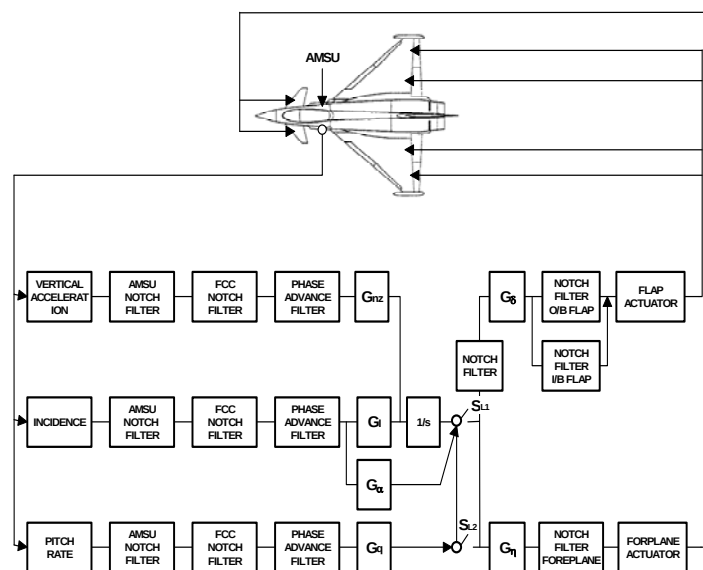


Fig. 1: Block Diagram of Longitudinal Control Stabilization path

The individual transfer function can be derived from two different methods the first using the analytical dynamic model calculation, the second using on ground measured sensor to actuator input transfer functions from the structural coupling test superimposed with calculated magnitudes of unsteady aerodynamic transfer functions. The applicability of the analytical dynamic model calculation depends on the accuracy of the modelling and its verification. Both methods depend on the accuracy of the unsteady aerodynamic transfer functions which are in both methods derived from theoretical predictions of unsteady aerodynamics for elastic modes and control surface deflection.

Analytical Model of the Flexible Aircraft with Flight Control System

The analytical model of the flexible aircraft with FCS consists of the linear dynamic description of the flight mechanic equations of motion, the description of the flexible aircraft through modal description using generalized coordinates, generalized masses, stiffness and model structural damping and generalized aerodynamic forces of the flexible modes and generalized control surface inertia and unsteady aerodynamic terms, the FCS is described through linear differential equations. In addition hardware and software, i.e. all sensors, actuators, computer characteristics are described by differential equations. The flexible aircraft with FCS can be demonstrated in a matrix form.

The equations of motion for the forced dynamic response of an aero elastic system can be written in matrix differential equation form:

$$m_r b_r^2 \begin{bmatrix} M_{qq} & M_{q\delta} \\ M_{\dot{q}} & M_{\delta\delta} \end{bmatrix} \begin{Bmatrix} \ddot{q} \\ \ddot{\delta} \end{Bmatrix} + \frac{s_R}{kV} \left\{ \omega_r^2 m_r b_r^2 \begin{bmatrix} gK_{qq} & 0 \\ 0 & K'_{\delta\delta} \end{bmatrix} + \frac{\rho}{2} V^2 F s_R \frac{b_r}{s_R} \begin{bmatrix} C''_{qq} & C''_{q\delta} \\ C''_{\dot{q}} & C''_{\delta\delta} \end{bmatrix} \right\} \begin{Bmatrix} \dot{q} \\ \dot{\delta} \end{Bmatrix} + \left\{ \omega_r^2 m_r b_r^2 \begin{bmatrix} K_{qq} & 0 \\ 0 & K'_{\delta\delta} \end{bmatrix} + \frac{\rho}{2} V^2 F s_R \frac{b_r}{s_R} \begin{bmatrix} C_{qq} & C_{q\delta} \\ C_{\dot{q}} & C_{\delta\delta} \end{bmatrix} \right\} \begin{Bmatrix} q \\ \delta \end{Bmatrix} = \{Q(t)\}$$

where m_r , b_r and ω_r are the reference mass, length and frequency and M , K and C are referred to as the generalized mass, stiffness and aerodynamic matrices which are non-dimensional. The generalized mass and stiffness matrices are calculated using a Finite Element Method (FEM) of the total aircraft. For dynamic response calculation the FEM is reduced to representative generalized dynamic DoF's. The true airspeed V and semi-span s_R of the reference plane are used to form the reduced frequency $k = (\omega s_R)/V$. F is the area of reference plane and g is the structural damping of the elastic modes. The generalized forces $Q(t)$ are equal to zero for the conventional flutter problem. The generalized coordinate q describes the amplitude of the elastic airplane modes including elastic control surface modes for a system with actuators whereas δ_0 denotes the rotation of the rigid control surface according to the complex actuator stiffness represented by the impedance function

$$K_{\delta_0 \delta_0} = K'_{\delta_0 \delta_0} + iK''_{\delta_0 \delta_0}$$

or the controlled aircraft the servo-induced control deflection $\Delta\delta$ has to be introduced as an additional degree of freedom for each control surface. The generalized forces generated by the servo induced control deflections $\Delta\delta$ can be described as the right-hand term by

$$\{Q(t)\} = -m_b b_r^2 \begin{Bmatrix} M_{q\Delta\delta} \\ M_{\delta_0\Delta\delta} \end{Bmatrix} \Delta\ddot{\delta} - \frac{\rho}{2} V^2 F s_R \frac{b_r^2}{s_R^2} \frac{s_R}{k \cdot V} \begin{Bmatrix} C''_{q\Delta\delta} \\ C''_{\delta_0\Delta\delta} \end{Bmatrix} \Delta\dot{\delta} - \frac{\rho}{2} V^2 F s_R \frac{b_r^2}{s_R^2} \begin{Bmatrix} C'_{q\Delta\delta} \\ C'_{\delta_0\Delta\delta} \end{Bmatrix} \Delta\delta$$

Assuming normalized rigid control surface modes δ_0 and $\Delta\delta$, the rotation of each control surface can be superimposed by

$$\delta = \delta_0 + \Delta\delta$$

δ is used here as abbreviation of foreplane, inboard and outboard flap or for rudder, and differential inboard and outboard flap. The state-space-description of the basic equation is as follows:

$$\{\dot{X}\} = [A]\{x\} + [B]\{u_i\}$$

The matrix in the equation describing the flexible aircraft with FCS is enlarged by linearized rigid flight mechanic equations. For example the state vector for longitudinal control includes then rigid aircraft state variables

$$X = \left[\Delta V / V; \Delta\alpha; \Delta\omega; \Delta\theta; \dot{q}; \dot{\delta}_0; \Delta\dot{\delta}; q; \delta_0; \Delta\delta \right]$$

The flight mechanic equations may in a first approximation contain elastified aerodynamic derivatives as function of incidence, Mach number and they are for low frequency assumed to be decoupled from the flexible aircraft equations. In another approximation the flight mechanic equations are fully rigid and theoretical inertia and unsteady aerodynamic coefficients are introduced. The flight mechanic equations for longitudinal control are also part of the equation system, but not described in this paper.

The assumptions to be made concerning the dynamic modelling of the structure have to be made in a conservative sense, because the design of the structural filters should be very robust to work proper in case of failures and amendments or updates of the structure.

FCS Optimisation and design updates

Different procedures are available to minimize structural coupling effects in the Flight Control System. The practical tools are to minimize structural coupling are:

- Optimum sensor locations
- Stiffening of the Inertia Measuring Unit platform
- Actuator transfer function
- Minimum weight/inertia of control surface

Fig. 1 demonstrates schematically the feed-back paths for the longitudinal stabilization. After the measurement of the vertical, lateral accelerations, the pitch rate, the roll rate and yaw rate, the flow sensor signals α and β , these signals are filtered first in the IMU by notch filters which minimize the signals at high frequencies, then the signals are filtered by the flight control computer notch - and phase advance filters. After multiplication with the Flight Control Computer (FCC) gains the signals are passed to the different control surface actuators. In front of the actuator input the signals are filtered by flap, canard and rudder notch filters. A

certain optimization is achieved by putting a number of notch filters in the feedback paths and to use also in the paths to the control additional filters. With this concept a better minimization of phase shifts at low frequency, which is necessary to meet the handling criteria, is possible.

Phase advance filters used in the FCS shall be configured that low frequency phase shifts due to notch filtering and other delays are put to zero at low frequency. In addition to minimize elastic mode coupling the well known dB increase of advance filters at higher frequencies shall be minimized. The optimization of phase advance filter should be combined with the notch filter optimization. This might be performed in an iterative manner or in a combined optimization with the frequency response functions including rigid aircraft response.

The notch filter optimization is the major tool for decoupling the aircraft control from aeroservoelastic influences. Since the coupling is of severe impact on the FCS on delta wing unstable aircraft a mathematical filter optimization had to be developed in order to achieve flight dynamic requirements. The optimization is described below.

In order to optimize the filters it is necessary to establish the open loop frequency response functions at the opened summation points of the longitudinal and lateral control S_{L1} , S_{L2} and S_{A1} , S_{A2} .

For example the open loop frequency response function at the longitudinal open loop point S_{L1} can be formulated using the separate transfer functions of the loop response without notch filters due to flap and foreplane excitation (S_{L2} closed) and by putting for each separate flap or foreplane excitation the gains of the loops in a sequence to zero.

An update phase of the FCS gains, phase advance filters and of notch filters is necessary and depends on the criticality of mismatch between design assumptions and test.

The structural coupling test on ground shows different frequencies of the elastic modes than assumed in the design (for instance a special configuration was not included in the design etc.).

The in-flight structural coupling test shows different dB and different phase of the low frequency elastic modes than assumed during design the on aircraft actuator characteristics are different from design assumptions. Redesign of structural parts local weight changes, change of pylon stiffness etc. during development phase. It is the task of the structural coupling/notch filter design and clearance procedure to treat this situation.

In Fig. 2 the advantages of integrated FCS design is demonstrated in a Nichols diagram of open loop frequency response.

- Case A shows the pure 'rigid' A/C design (flight mechanics design)
- Case B presents the result on open loop frequency response function using separate independent notch filter design (elastic design) applying a full gain stabilization concept for elastic modes, which leads to high phase shift at the low frequencies and conflicts the requirements.
- Case C demonstrates the advantages of phase stabilization on the 1st wing bending (WB1) and rigid A/C motion.
- Case D points out the profits of integrated design; both the rigid A/C and elastic modes have sufficient phase and gain margins.

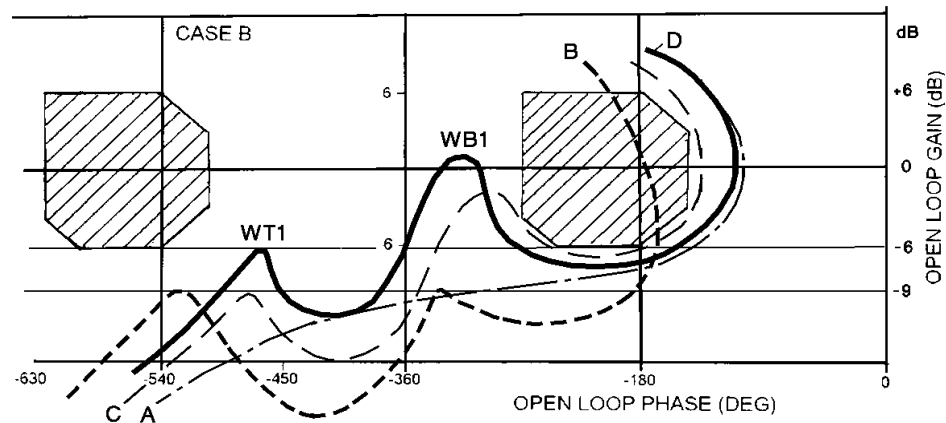


Fig. 2: Integrated versus classical concept

The design and clearance procedure which is based upon a series of on aircraft tests and on rig tests for identification and clearance and consequently a series of dynamic model updates from testing and from updated FCS definitions. Reiteration of the full process of ground tests, prediction model updates and notch filter optimization is necessary for new configuration etc.. Fig. 3 explains the design and clearance procedure which is based upon a series of on aircraft tests and on rig tests for identification and clearance and consequently a series of dynamic model updates from testing and from updated FCS definitions. Reiteration of the full process of ground tests, prediction model updates and notch filter optimization is necessary for new configuration etc..

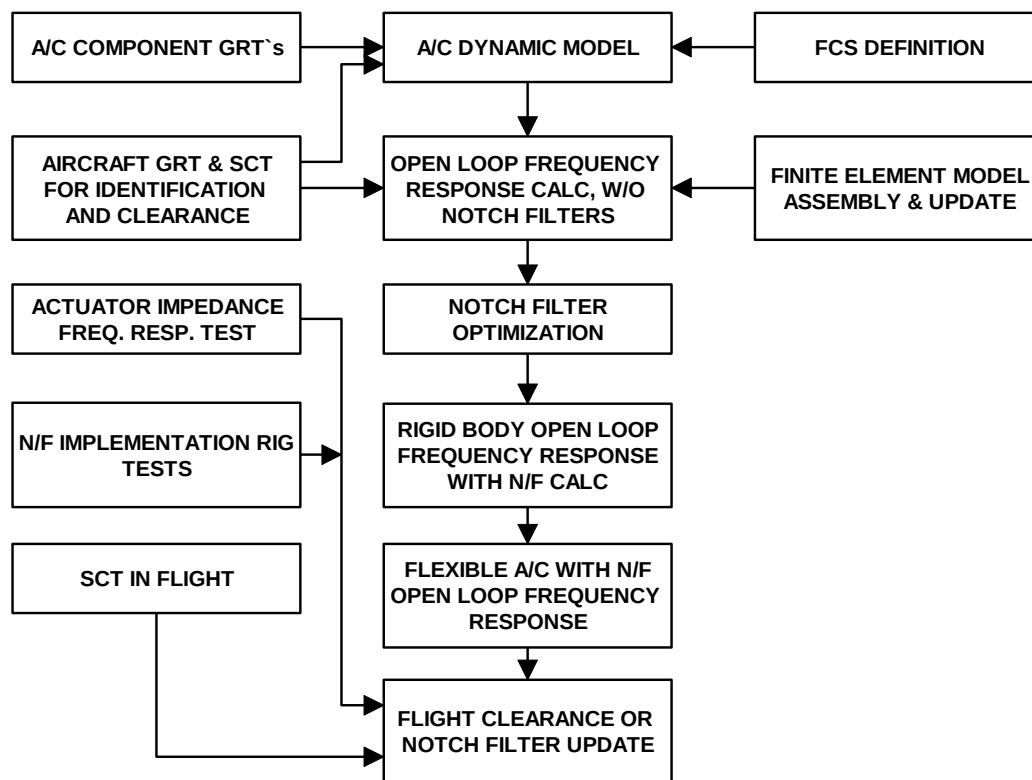


Fig. 3: Design and Clearance Procedure

Structural Coupling Test (SCT) on Ground

The main objectives of the ground SCT are:

- Model validation
- Not modelled aspects
- Provide data for the frequency range where the model alone is not considered to be an adequate basis for production of filter design information.

The amount of analysis to be carried out using the aeroservoelastic model is really huge. Calculations are in fact required for Notch Filter (NF) design and optimisation, and subsequently for flight clearance and qualification purposes. Considering the wide possibility of combinations of external stores for a military multirole aircraft, which strongly influence the dynamic response characteristics of the airframe, it is evident that only a limited set of external store configurations can be tested on ground, the rest being studied only through calculations. The consequence is the need of an adequate mathematical model for SC analysis and methods to augment the model predictions using a limited set of test data.

The main objective of the ground SCT is to get all information needed to evaluate how the model simulates the Structural Coupling (SC) characteristics of the aircraft in absence of aerodynamics, and then to update the model and the preliminary NF, if necessary. The other important objective is to collect enough experimental data in the high frequency range. It is well-known, in fact, that the quality of the model predictions above a certain frequency is rather poor. Since the NF are to be implemented in a digital system the frequency range that must be covered in their design depends on the sampling rate and, as a consequence, the analysis is to be extended beyond the model capabilities. During the Identification Test the measurement of transfer functions is performed also covering the frequency range where the model is not satisfactory. The relevant experimental data will feed a procedure developed to augment model predictions, based on the direct usage of experimental data combined with model predictions.

Additionally, the test is very important to assess the influence on the aircraft response of structure non-linearity, hydraulic failures, control surface trim position, actuator hinge backlash, undercarriage support, etc., not implemented in the linear model.

Essentially the test consists in measuring the IMU signals in response to the excitation of the aircraft obtained by means of sinusoidal rotation of control surfaces about their hinge axes. This gives rise to inertial forces due to the surface Center of Gravity (cg) offset with respect to the hinge axis, and makes the structure respond at the same frequency of the surface oscillation. The relevant vibration levels are picked up by the IMU sensors, measured and then used to calculate the transfer functions corresponding to those already employed for the preliminary NF design. The test is carried out in open loop, avoiding that IMU signals are sent to the FCC and therefore to the control surface actuators. This stage of testing is defined *Identification Test*, because it serves to identify the SC aircraft characteristics and it must be performed quite early with respect to the flight date, depending on the time required for the updating of notch filters.

Another SCT stage is usually foreseen in the route to clearance just before the first flight, called *Confirmatory test*, the aim of which is to verify that the updated NF satisfy the requirement for the aircraft in the pre-flight standard. This test is necessary when significant

structural changes are introduced, above all in the mass distribution, between pre-flight and identification test aircraft standard.

The following paragraphs will be devoted to describe more in detail all the aspects which are typical of ground SCT.

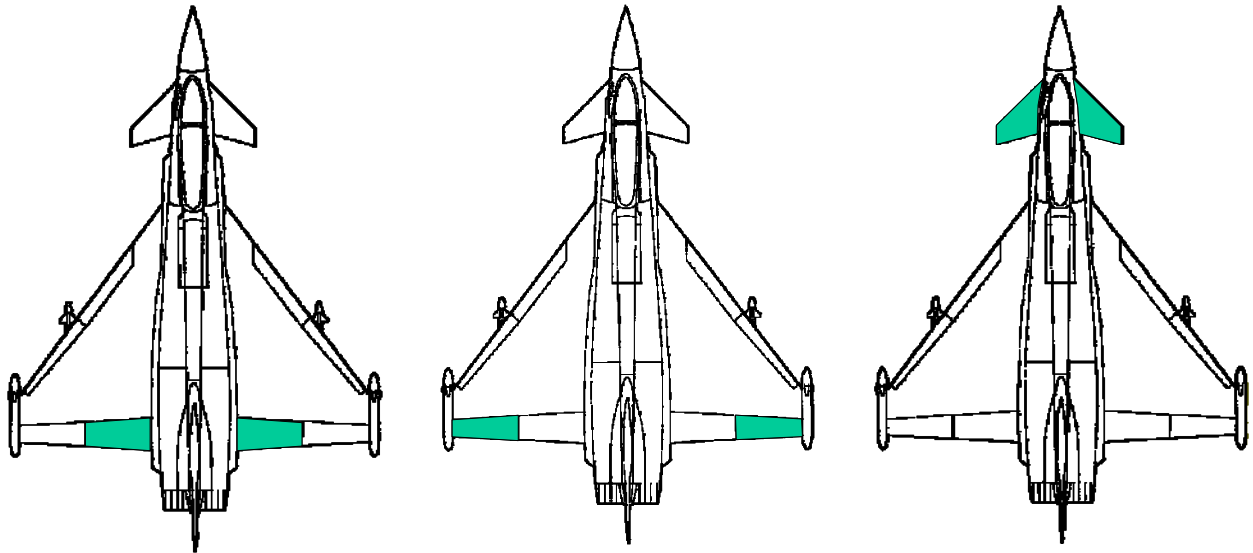


Fig. 4: SCT Symmetric excitation system

Excitation Procedures

SCT is peculiar for the way adopted to shake the airframe in order to get the dynamic response. The inertial forces which excite the aircraft are generated making the control surfaces oscillate about their hinge axis. To do this a sinusoidal signal is generated by the test equipment and then sent to the control surface actuators through an appropriate setting of FCC. The control surfaces are not moved all at the same time, but they operate in couple or alone, depending whether they are symmetrically located on both sides of the aeroplane or not (rudder). Two different types of excitation can be considered: symmetric, sending the same signal to the two surfaces of the couple; anti-symmetric, sending signals with same amplitude but shifted in phase of 180 degree. With this approach it is possible to excite separately the symmetric and anti-symmetric modes of the airframe. Fig. 4 shows the different combinations of control surfaces and the relevant IMU signals measured to calculate the Open Loop Frequency Response Function (OLFRF). Fig. 5 shows two possible excitation profiles, sine step and logarithmic step excitation.

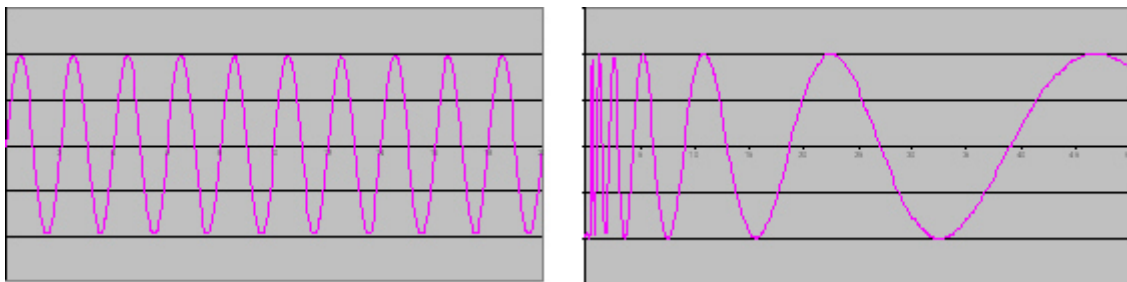


Fig. 5: Sine Step and logarithmic frequency sweep excitation system

The aim of the test is to identify how the principal modes of the aircraft respond to this kind of excitation to fulfil this task a sine step sweep procedure has been adopted, changing the frequency of the signal with discrete steps and maintaining the same signal for a certain number of cycles, during which signals from IMU are measured. The Sine Step method has shown to be more appropriate than a sine continuous sweep with logarithmic frequency variation, because it allows gathering data for more cycles at each frequency and consequently a better average of the aircraft response.

The amplitude of the oscillation must be chosen wide enough to obtain the level of forces needed for a proper response of the aircraft. It must change with frequency: it is wide at low frequency, and it diminishes as frequency increases, in order to respect the Leuven measuring system (LMS) constraints. The approach normally followed to obtain the best response of the structure is to maintain the level of the excitation amplitude as high as allowed by the LMS constraints. Some preliminary runs are dedicated to optimise the amplitude of the sine sweep: starting from a value at low frequency, the run is repeated increasing every time the level of amplitude, until the LMS cuts off the input signals. On the basis of this level an appropriate amplitude profile versus frequency can be defined, following two opposite necessities: to keep the amplitude as high as possible and to avoid a continues interruption of the test by the LMS. Fig. 6 shows an example of how the amplitude profile, of the signal sent to the control surface actuators, can change with frequency. It can be seen that, due to LMS constraints it might be necessary to reduce the amplitude also at some resonance.

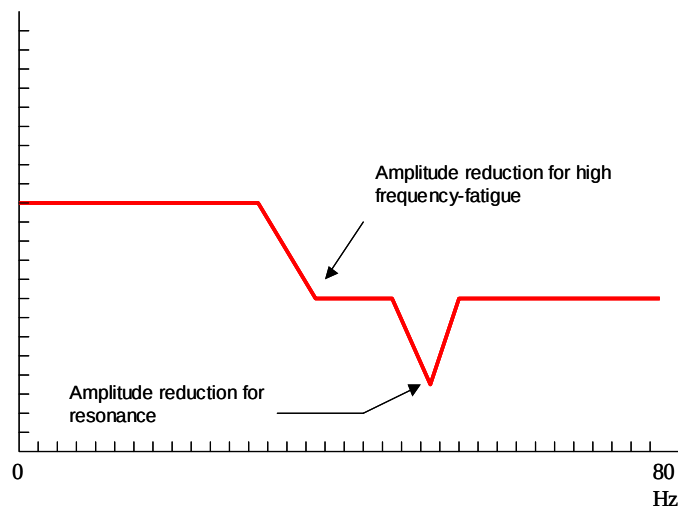


Fig. 6: Amplitude Profile used for excitation

Identification Test

The OLFREs to be measured are defined by the procedure for the NF design, and can be deduced from the sketches reported in Figure 4. They must be measured in a frequency range extended up to the sampling rate which characterises FCC digital signals. This is necessary to take into account the folding-back effect of the high frequency range due to digitalisation.

To carry out the SCT it is necessary to exchange data with the FCC and this function is performed by the ATE (Automatic Test Equipment), a device designed for pre-flight FCC checks and able to perform the following operations during SCT:

- set up and read/write FCC parameters
- injection of the excitation signal in the FCC
- reading of IMU sensor signals from FCC facilities
- real time presentation of FCC signals.

The excitation signal is generated by a TFA, incorporated in the ATE and interfaced with an external PC. The same TFA performs the calculation of the OLFRRs and sends the relevant data to the PC for storing and subsequent analysis. Figure 7 illustrates the layout of the test, showing the links among the test items and the exchanged data. During the test measured OLFRRs are compared with theoretical predictions, in order to check whether unexpected or unwanted effects are influencing the test.

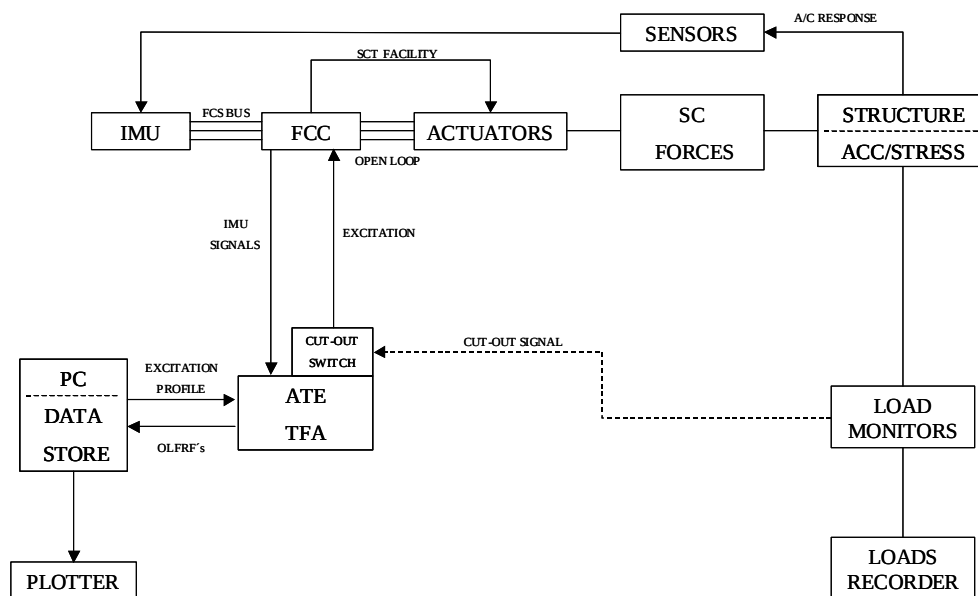


Fig. 7: Structural Coupling Test layout

It is very important to verify the degree of non linearity of the aircraft response during the test, looking at the shape of IMU and LMS sensor time histories traced in real time by a brush recorder. More detailed information is obtained repeating some runs, usually for the most important normal modes, at different amplitude levels. The lesson learnt from the SCT is that the highest amplitude levels compatible with LMS constraints should be used, to keep non linearity effects to a minimum level. Figure 8 shows the same OLFRR measured at different amplitude levels in the first wing bending frequency range, highlighting that the main effect of non linearity is on the amplitude of the peak, with small influence on the frequency. Besides the influence of amplitude other test runs are to be carried out, in order to investigate the influence of failures of one or two of the four redundant hydraulic systems and FCCs. These checks are needed since in this case the actuator performances can present significant changes, influencing the OLFRRs and thus the NF design.

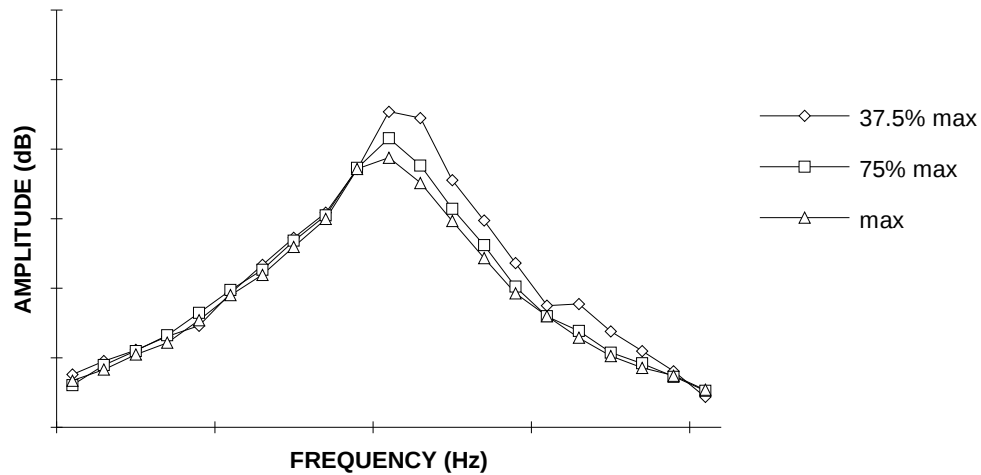


Fig. 8: Influence of excitation amplitude on OPFRF

Confirmatory Test

The *identification test* covers all the aspects necessary to identify the SC characteristics of the aircraft required for the NF design. It is very detailed and carried out for different aircraft configurations, regarding both external stores and internal fuel. This is done to verify the theoretical predictions relevant to the influence of mass characteristics on the aircraft response.

On the contrary, the *confirmatory test* is intended to be a very short test, with the aim of verifying that the aircraft in the ready-to-fly standard does not present significant changes in the response with respect to the *identification test*. The test is therefore to be carried out when the aircraft is in the flight configuration. The verification is normally limited to the modes that are very sensible to mass distribution changes and that play a leading role in the NF design. It consists in a short *identification test*, limited to few modes, selected as most critical from the NF design point of view.

The confirmatory test is the last step in the NF qualification route before flying the configuration investigated. It is needed to issue the SC flight clearance: from the analysis of the test results it will come out whether the NFs, based on data from the *identification test*, can be confirmed for flight or not, and a reassessment for worst flight conditions can be necessary. In the worst case flight limitations might result for some regions of the flight envelope.

Updating of the Aeroservoelastic Model

To accomplish the NF design procedure the OLFRFs relevant to external store configurations are required. Considering the number of configurations and the possible sub-configurations deriving from store release, it is essential the development of a reliable aeroservoelastic model to perform the amount of calculations required for the NF design.

Among the components of the aeroservoelastic model there is the aircraft structural dynamic model, the updating of which is discussed in this paper. The basis of this model is the Nastran Super element Technique, which allows to design simpler models and then to assemble the final model with a linking procedure. In the case of the canard-delta the airframe has been divided in the following super elements:

- wing including flaperons and slats
- fuselage
- foreplane
- fin and rudder

Each super element consists in a mass and stiffness matrix, calculated using the relevant FEM and applying a reduction to a set of DOFs. The dynamic reduction of the model is a very important stage, since it allows reducing drastically the number of DOFs, leading to a simplified model. The DOFs selection must be performed following the guideline that the reduced model has to simulate adequately the structural dynamic characteristics of the component in a certain frequency range. Some trials might be required before a satisfactory result can be achieved.

Model Updating on the Basis of GRT Results

The GRT results represent the basis for the updating of the dynamic model. All the remarks that follow about the model updating, are based on the activity performed after the GRT and SCT campaigns carried out on the first U/W stores configurations to be cleared.

Considering that component GRT for wing, fin, foreplane and pylons, had already been carried out and the relevant super elements updated, the GRT on the assembled aircraft was required to gather data for the updating mainly of the fuselage super element and of all the elastic elements used to simulate, in the assembled model, the links among the super elements. Figure 9 is a sketch of the super element model updating activity performed before the GRT on the aircraft. At this stage a preliminary updated model was available and it was used to predict the response of the aircraft during GRT and SCT. It was also employed to carry out all calculations required for the preliminary NF design. Figure 10 illustrates the next step, carried out after the GRT and concerning with the delivery of the final updated model, including all the effects not covered during previous super element GRTs.

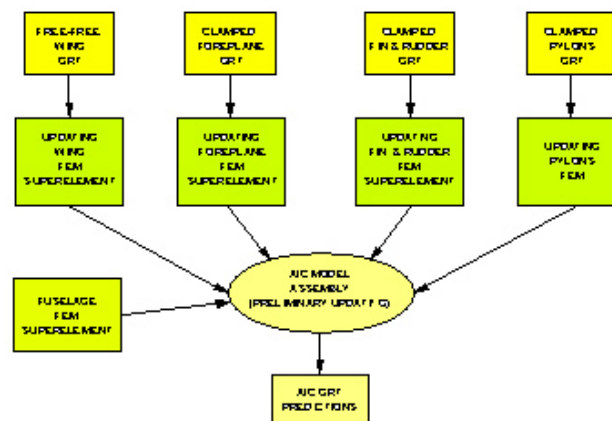


Figure 9: Model preliminary update procedure (Components)

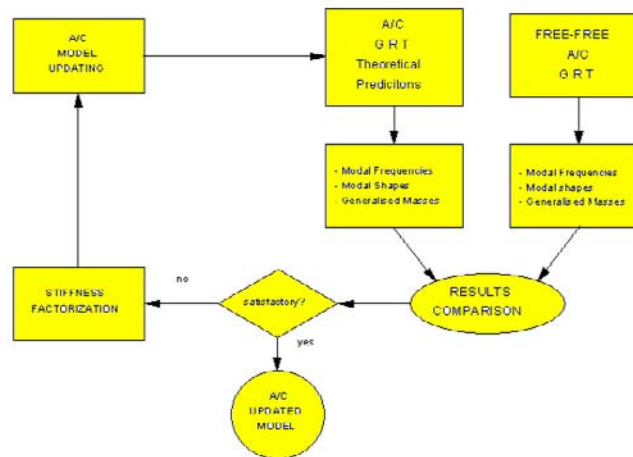


Figure 10: Aircraft dynamic model update based on GRT results

From a first rough look at the aircraft GRT results it came out that the model had the general trend to predict lower modal frequencies. The differences between test results and predictions indicated that a model adjustment was necessary. The correction was obtained applying factors to the superelement stiffness matrices and updating the mass distribution of the model, the latter based on the assessment of the aircraft mass distribution carried out before starting the test. Several trials were needed to find a set of factors for the superelement stiffness matrices, but eventually this approach demonstrated to be adequate to obtain satisfactory results. The factors were all greater than one, the greatest being applied to the fuselage, and the updated stiffness matrices were obtained multiplying all their elements for the relevant factor.

Before starting with the updating procedure it was necessary to manipulate the experimental data, transforming the GRT modal shapes in perfectly symmetric and anti-symmetric modes. This step was needed since the aircraft model is a representation of half aircraft. The main problems with asymmetry in modal shapes came from modes characterised by external stores and control surfaces wide motion. For these cases the approach was to consider data coming only from the accelerometers located on the side of the aircraft which showed a better *phase index*.

The correction procedure was iterative, starting with an initial set of factors. The new model was assembled using the factored superelement matrices and modal characteristics compared with those ones measured during the GRT. From this comparison a new set of factors would be defined and the process repeated until a satisfactory comparison could be found. The modal characteristics monitored during the iterative procedure to establish when the process could be stopped were the modal frequencies, the generalised masses and the modal shapes.

Regarding the frequency, the comparison was based on the percentage difference between test and model data. Figure 11 shows the situation for some modes at the end of the procedure, pointing out to the improvement obtained for this parameter with respect to the GRT predictions.

For the comparison of the generalised masses and the modal shapes it was necessary to renormalize the theoretical modes, in order to make them homogeneous with the measured ones. In general the location of the accelerometer with the highest response level was chosen as reference point. This step was repeated for different points, depending on the modal shape and the accelerometer phase index measured during the acquisition of the mode. The aim of this repetition was to understand how the selection of the reference point could influence the calculated generalised mass. To perform these checks without problems the GRT

accelerometer map was designed making the accelerometer locations coincide with model grids whenever it was possible. This approach could be easily followed for components like wings, foreplane and fin, but for the fuselage an interpolation of sensor data was necessary. For the comparison of modal shapes the following index was calculated:

$$\frac{(\Phi_{theory} \bullet \Phi_{GRT})^2}{|\Phi_{theory}|^2 \times |\Phi_{GRT}|^2}$$

where Φ_{theory} and Φ_{GRT} are the two eigenvectors to be compared.

In the updating procedure a special attention was dedicated to the most significant modes, namely those ones that in the previous analyses had shown to have a considerable influence on flutter, SC and dynamic loads. This approach allowed obtaining a model that can be considered adequate for general dynamic analyses, the modes represented by the model with less precision being not essential for the study of aeroservoelastic criticality.

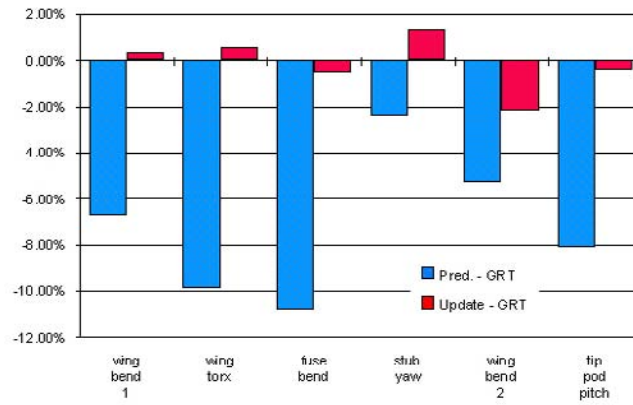


Figure 11: Differences of modal frequencies

Since the issue of Ref. 8 several GRT and SCT campaigns have been performed, each one devoted to investigate a set of critical store configurations, followed by further updates. As expected, the corrections were necessary only to those items, like pylons and launchers, not tested before and there was no need to touch the baseline aircraft model updated after the first GRT campaign.

Model Updating on the Basis of SCT Results

Progressing with the development of the aircraft the necessity to cover more stores configurations with the same set of filters became the most challenging problem to solve. This is also the final target: a unique set of NF able to guarantee the required gain and phase margins for all configurations. It was immediately evident how difficult was this achievement and a less conservative approach was necessary, starting from a better correlation of the Structural Coupling model with SCT results. Moreover, the introduction in the design procedure of the Phase Stabilisation concept (Reference 1) required also validating the phase predicted by the model for the low frequency modes, extending the comparison also to the phase of the OLFRRFs.

The SCT is in general performed in parallel to the GRT, but not necessarily on the same configurations. The aim of the GRT is in fact to collect enough data for the modal identification of the aircraft, and in particular for specific items like pylons and launchers. In this case single store configurations are acceptable. For the SCT, since the high frequency measurements are used directly in the notch filter design, the configurations must be representative of the most critical ones, previously identified by the model.

Immediately after the release of the model updated on the basis of GRT, the next step is the simulation of the SCT runs carried out on ground, using initially the modal damping values measured during the GRT. At this stage a further improvement is introduced in the aeroservoelastic model, replacing the actuator transfer functions with the frequency functions measured during the preliminary phase of the SCT. These frequency functions are thus compared with the test data.

In general the correction is needed only to match the amplitude of the main modes responses, the frequency being already corrected during the GRT updating. No attempt is made to correct the phase, but simply a monitoring of main modes to confirm the applicability of the Phase Stabilisation concept.

The first step in the correction of the SC model is to identify the possible source of errors in the model and next to find a procedure simple enough to obtain a satisfactory result succeeding to manage the several configurations to be covered. The source of errors considered for the model correction is the following:

- Fuselage model and IMU location.
- Modal shapes.
- Non linearity effects.

The first point is very important, but difficult to manage. The superelement representing the fuselage is in fact reduced to a limited number of grids along the longitudinal axis representing the main structural stations, and other grids located in correspondence of main equipment items to simulate their mass and inertia characteristics. Among the latter there is the IMU grid, but trying to match the SCT results changing the elements of the stiffness and mass matrices was not considered practical. Another approach was tried, applying appropriate factors to the modal deformations of the IMU grid, but a satisfactory solution was not found, above all because the same factor had effect on all OLFRFs and instead the level of error in the same mode is different for each control surface excitation. This result confirms that the correction of only the fuselage modal shape is not enough and, since the inertial excitation induced by the oscillating surface depends on the modal response of the aircraft, a more general correction is needed. However, in order to generate a reliable model, the location of sensors and the fuselage model must be considered with great care. Concerning with the modal shapes, the matching of the main modal frequencies was considered a success, since the optimisation of the model with the modes as constraints at the current state of the art is feasible only using very simple dynamic models.

Another aspect to be considered in the model updating is the effect of non linearity on the test results. As explained in the description of the SCT procedure, the transfer functions are measured changing the level of the surface deflection as function of frequency. This means that, if the effect of the non linearity of the system is significant, the comparison with the model is affected by an error distribution that depends on the frequency. For the most important modes this effect is assessed repeating the surface excitation at different input levels. In general the effect on the frequency of the mode is small, but on the amplitude it is sensible and should be taken into account. The general approach is to consider the amplitude

associated to the highest level of excitation and to change the original GRT modal damping values according to SCT data.

On the basis of previous remarks it has been decided to adopt a data base of corrective frequency functions to be applied to the OLFRRFs calculated by the model. The data base is generated according to the following procedure:

- The data base contains several sets of corrective functions, one set for each store configuration tested on ground during the SCT.
- Each set contains one corrective function for each OLFRRFs, calculated comparing the measured and the related analytical OLFRRFs.
- The data base contains also a set where each corrective function is the envelope of the corresponding functions calculated for the tested configurations. This set will be used for configurations not tested during the SCT.

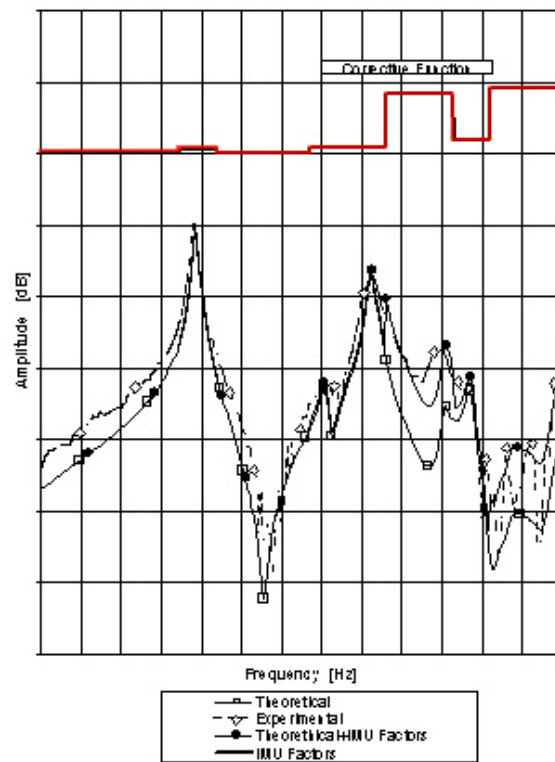


Figure 12: Example of a corrective function

The correction process consists in performing the product of each OLFRRF for the associated corrective function before the filter optimisation phase. This approach allows the correction of the structural uncertainties of the model. Figure 12 is an example of how this method is applied. The picture shows the typical situation encountered during the correction procedure: a very good matching of the model for the first modes and the necessity to introduce a correction for the modes close to the frequency limits of application of the model.

A further correction can be implemented after the structural coupling flight trials have been completed. This correction is much simpler, being associated to the efficiency of control surfaces, generally overestimated by the model. A factor can be identified for each significant mode and applied for all configurations, since the effect of stores on these factors can be

accepted as negligible. The main difference between the structural and the aerodynamic correction is that the first is represented, generally speaking, by an amplification factor, the second by an alleviation factor. Figure 13 illustrates this characteristic from the comparison of model predictions with flight test data. It gives an idea of the degree of conservatism of the model, from which the aerodynamic alleviation factors can be derived.

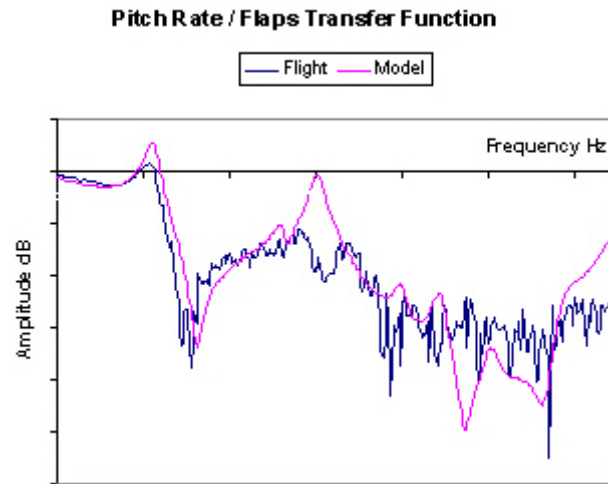


Figure 13a: Comparison of SCF flight test results with model prediction – Pitch rate

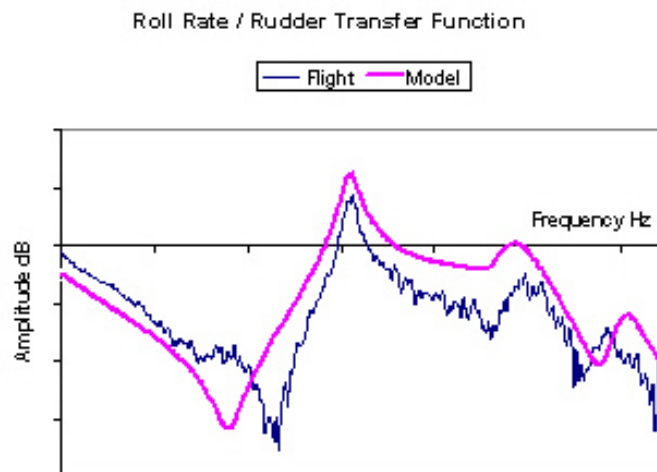


Figure 13b: Comparison of SCF flight test results with model prediction – Roll rate

The procedure described, even though complex, is very practical and easy to be implemented in a global automatic procedure for the generation of the OLFRFs needed for the NF design. Its weak side is represented by the management of a data base with data associated to several stores configurations and the necessity to identify the most critical configurations to be tested on ground. The number of critical configurations can be significant and the dependency on ground testing is a heavy burden in the qualification of a multirole aircraft. For future aircraft an improvement in the structural and aerodynamic modelling techniques is necessary, in order to reduce the cost and the risks inherent in the design and qualification of notch filters.

Flight Test Results - Update of Unsteady Control Surface Aerodynamics

Flight test results from structural coupling/flutter tests are needed to verify or update the predicted results of open loop frequency response functions by the update of unsteady

aerodynamic forces used in the dynamic model. This can be achieved through the comparison of predicted open loop frequency response functions and flight test measured closed loop converted into open loop frequency response functions.

The flight test results are derived through frequency sweep excitation of the control surfaces which should be possible through special software in the FCC's.

Fig. 3 demonstrates a typical result for the comparison of predicted and measured IMU open loop response due to control surface input, showing that unsteady aerodynamic coupling modelling has to be updated with in flight measured results both for low angle and high of attack α . From the flight test results it is concluded that the theoretical control surface unsteady aerodynamic coupling terms used in the total dynamic model have to be tuned to test results for low up to high incidences.

Fig. 6 demonstrates via flight test results the alleviation effect resulting from phase stabilisation concept of first wing bending mode compared to gain stabilisation concept in closed loop. Alleviation of IMU pitch rate is found to be at least 3dB.

Conclusions

From the results of structural coupling investigations performed for an advanced fighter aircraft the following lessons have been learned:

Dynamic modelling of the fuselage response and the essential sensor response due to control surface inputs is limited to a certain low frequency range due to the total aircraft finite element model representation, being extremely complex and due to unsteady control surface aerodynamic representation.

Therefore extensive structural coupling tests on ground are necessary to update dynamic fuselage modelling for the inertia coupling terms and structural coupling in flight tests are necessary to update the unsteady aerodynamic control surface coupling terms.

The integrated design of FCS gains, phase advance filters and of structural coupling/notch filter design and clearance procedure has shown advantages in comparison to the classical separate FCS and notch filter design. The profits of integrated design are mainly found to be:

- less degradation effects on rigid aircraft stability margins
- improvements of elastic mode stability
- improvement of phase advance filters combined with lower high frequency end to end gains
- a robust FCS design has to be adopted which includes the description of all worst case assumptions for the structure, the FCS gains and FCS minimum and maximum phase at low elastic mode frequencies, and worst case assumption for actuator and sensor hardware

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