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ORIGINAL PAPER

A Survey on Structural Coupling Design and Testing of the Flexible Military Aircraft

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Abstract

Structural coupling (i.e., aeroservoelasticity) between the flight control system with multiple motion sensor feedbacks and the aeroelastic modes of the flexible airframe can be a serious problem in high-performance aircraft. In a military specification document, MIL-STD-9490D, adequate stability margin requirement is required to be complied with for all frequencies at or beyond the first aeroelastic mode in all flight conditions. To minimize the structural coupling effect to satisfy the stability margin requirement in flying qualities aspects, various design methods are adopted during the aircraft development stage as follows: to reinforce aircraft's structure on which sensor is mounted, to minimize weight/inertia of control surfaces, to select the optimal sensor position, to optimize control gains, to design notch filters for minimizing phase lag in low-frequency band, to design phase advance filters, to apply additional active control technique, and so on. In this paper, we identify major design considerations which have been applied to the production fighter aircraft to minimize the structural coupling. The reviews regarding the prospects on design technologies would be most helpful to the structure and flight control law engineers.

Keywords Flexible military aircraft · Airframe interaction · Structural coupling · Structural coupling filter (SCF) · Structural coupling test (SCT) · Aircraft motion sensor unit (AMSU)

Abbreviations

ARI	Aileron rudder interconnection
ASA	Accelerometer sensor assembly
AEOL	Aircraft engine operation limitation
AMSU	Aircraft motion sensor unit
ATE	Automatic test equipment
AOSS	Active oscillation suppression system
BDFT	Biodynamic feedthrough
CAS	Control augmentation system
CG	Center of gravity
CP	Center of percussion
EMD	Engineering and manufacturing development
EAP	Experimental aircraft programme
F8	Variable gain
FL	Flutter
FLCC	Flight control computer
FM	Flight mechanics
FCS	Flight control system
FTA	Flight test aid
GM	Gain margin
GVT	Ground vibration test
GRT	Ground resonance test
HQ	Handling qualities
HT	Horizontal tails
IMU	Inertia measurement unit
JSF	Joint strike fighter
KEAS	Knots equivalent airspeed
LEF	Leading edge flaps
LCA	Light combat aircraft
LCO	Limit cycle oscillation
LDGP	Low drag general purpose
LOES	Low-order equivalent system technique
M.G.	Manual gain
MGR	Manual gain roll

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MGY	Manual gain yaw
NF	Notch filter
OLFRF	Open-loop frequency response function
PM	Phase margin
PIO	Pilot-induced oscillation
PSD	Power spectrum density
RSA	Rate sensor assembly
RSS	Relaxed static stability
SC	Structural coupling
S&C	Stability and control
SCT	Structural coupling test
SM	Structural mechanics
TEF	Trailing edge flaps
TFA	Transfer function analyzer
ZOH	Zero order hold
ASA	Accelerometer sensor assembly
ASE	Aeroservoelastic
HM	Hinge moment

1 Introduction

In aircraft development phase, structural engineers try to reduce the weight of wings, fuselage, and empennage for increasing structural efficiency. As there is the trend of using new materials such as composites for these main structures [1–8] in recent years, the aircraft becomes more flexible [9]. Meanwhile, aerodynamic designers have applied the configuration design of the concept of relaxed static stability (RSS) to aerodynamic design to improve aircraft maneuverability, so that the static stability of the aircraft has become inherently unstable [10]. Also, the control law designer has developed a full-authority flight control system to assure stability and achieve the good handling qualities and ride qualities in unstable aircraft [11–15]. Since, unfortunately, each engineer's effort to improve structures, aerodynamics, and flight control was individually conducted, the overall performance degradation caused by each interaction was not recognized.

After World War II, aeroelastic characteristics were beginning to be significantly considered at the stage of aircraft development. Moreover, as the flight control system was widely applied recently, structural coupling (i.e., aeroservoelasticity) which is the interaction among the structural flexibilities, aerodynamics, and flight control systems has been an important design consideration issue [16]. Historically, the importance of structural coupling design has emerged as the aircraft's structural efficiency increases, when minimum structural weight is considered to meet strength requirement, and the usage of high control gain has increased to improve handling qualities. In fact, most military fighter jets requiring newly developed variety external armaments are strongly influenced by structural coupling effects [17–19]. However, these interactions must be less at certain structural mode frequencies to meet stability requirements [20].

For stability, MIL-STD-9490D [20] presents the requirements for structural mode margins. It is specified in the standard document that at least a ± 6 dB gain margin must be satisfied for both aerodynamic and non-aerodynamic closed loops at zero airspeed condition, though the damping of structural modes generally increases as speed increases. And in the operational airspeed regime, at least ± 8 dB gain margin and ± 60 degree phase margin are required in the first aeroelastic mode or higher frequencies. To satisfy this, many considerations must be considered during aircraft development by the development engineer, such as the adequate location selection of the aircraft motion sensor unit (AMSU), which is also called inertia motion sensor (IMU), the optimization of control gain and structural stiffness of AMSU mounting structure, the minimization of control surface weight, and the design of a notch filter in the control surface command or control feedback path [18, 21–25]. Unfortunately, the commonly notch filter for damping structure mode causes an undesirable phase lag at low frequencies [26]. It is evident that the phase lag that occurs at low frequencies is a significant factor in aircraft design, since it appears as a time-delay and can degrade handling qualities [26]. Therefore, when a control law engineer designs a notch filter in a control path, the minimum phase lag in the low-frequency band must be a design constraint, so that the stability margin and equivalent time-delay requirements can be satisfied at the same time [27].

In this paper, representative structural coupling cases experienced in the aircraft development process are analyzed, and major design considerations that developers must be aware of to minimize structural coupling characteristics and eliminate instability caused by these characteristics are summarized. In addition, efficient and appropriate test procedures and methods are reviewed to obtain more accurate and reliable data in structural coupling test (SCT), which is the final stage of verifying the structural coupling compensation design for flight test clearance. The results of this study are expected to contribute to the successful completion of aircraft development by reducing the developer's trial and error as a guideline for effectively resolving instability due to structural coupling that may occur during the aircraft development process. And for students entering aeronautical engineering, it will be very helpful in understanding structural coupling characteristics and the physical relationship between aircraft subsystems.

The rest of this article is organized as follows: Section 2 introduces the airframe interaction. Section 3 describes the historical background of structural coupling in military aircraft development process. Section 4 presents the related

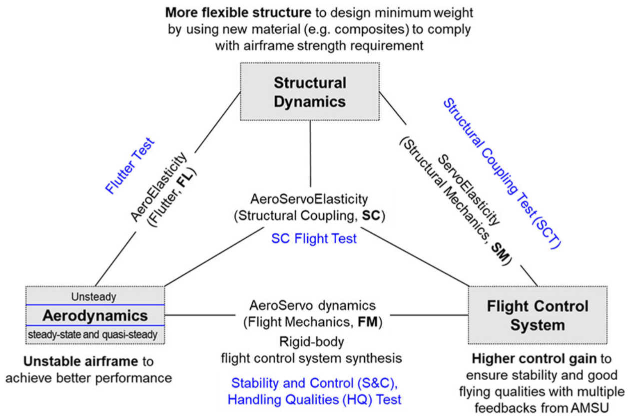


Fig. 1 Airframe interaction [16]

requirements for the structural coupling. Section 5 presents design and testing consideration to minimize the structural coupling characteristics. Section 6 presents model update based on ground test data and describes flight clearance process to guarantee the flight safety. Finally, Section 7 presents conclusions and future works.

2 Airframe Interaction

As shown in Fig. 1, phenomena that occur due to the interaction between structural dynamics, aerodynamics, and flight control system of an aircraft can be divided into flutter, flight mechanics, structural mechanics, and structural coupling [16].

The left leg of the triangle is a classical flutter, which is instability resulting from the interaction of the aircraft's flexible structural dynamics due to the elastic structure and unsteady aerodynamics. It has a significant impact on the evolution of the aircraft since the earliest days of flight.

The lower leg of the triangle is the flight mechanics, which is the instability resulting from the interaction of the steady-state and quasi-steady rigid aerodynamics at a frequency lower than the first elastic mode and the flight control system designed with high control loop gain. These aircraft characteristics are verified through stability margin analysis in the low-frequency band, using the flight control system designed based on rigid-body airframe dynamics. In general, the performance of the control law is proportional to the high control loop gain, but the high control gain results in system instability in conjunction with electrical and mechanical nonlinearities.

The right leg of the triangle is the instability, which is called "Ground Resonance," caused by the interaction between the aircraft's flexible structural dynamics and the flight control system when there is no aerodynamic force on the ground. In the process of aircraft development, the satisfaction of the gain margin is verified in all frequency bands including the structural flexible frequency by SCT or ground resonance test (GRT) to confirm flight clearance for the flight. In addition, the high-frequency flexible structural dynamic data obtained in the SCT process are used for model update to improve the reliability of structural coupling analysis results.

Finally, structural coupling, the subject of this study, is the instability caused by the interaction among structural dynamics, aerodynamics, and flight control system of an aircraft. This instability is a phenomenon caused by the introduction of a closed-loop control system to the flexible aircraft frame that feeds back the aircraft status measured from the sensors. In general, control laws engineers are interested in designing to ensure the stability of an aircraft's rigid-body motion and to improve flying qualities. However, a motion sensor such as an AMSU for measuring the state variables of an aircraft detects not only the rigid-body motion of the aircraft but also the higher frequency oscillation superimposed by the resonance or flexible mode of the structure. Because of this, the higher frequency oscillation that is fed back, as well as the rigid-body rates and accelerations, is amplified through the flight control system to generate control surface commands. In other words, the control surface commands generated by flight control system can excite resonance and form a closed loop with the aircraft's aeroelastic mode, further destabilizing the aircraft with attendant potential for instability.

3 Historical Background

As some recently developed aircraft increase structural flexibility due to weight reduction for structural efficiency improvement, statically unstable configuration design for maneuverability improvement, and full-authority flight control system design for stability enhancement, the aircraft experienced instability due to structural coupling characteristics, so that efforts have been made to improve it. Even advanced aircraft companies specify, in the aircraft development process, that they consider design and verification to improve it.

As shown in Fig. 2, this chapter introduces a list of representative cases of design improved after experiencing instability due to structural coupling during aircraft development [16, 17]. The list presented is B-36 [9], B-52 [28, 29], YF-16 [30–32], YF-17 [33], F/A-18 [34, 35], F-22 [36], and X-35B [37], which is by no means complete as only a few cases have been documented using published data, but it is still useful for identifying design considerations for reducing structural coupling.

Aircraft	B-36 [9]	B-52 [30][31]	YF-16 [32][34]	YF-17 [36]	F/A-18 [37][38]	F-22 [39]	X-35B [40]
							
National	United States	United States	United States	United States	United States	United States	United States
Manufacturer	Convair/Boeing	Boeing	Lockheed Martin	Northrop	Boeing	Lockheed Martin	Lockheed Martin
First Flight	1946	1952	1974	1974	1978	1997	2000
Problems	ASE instability was induced by the autopilot due to sensor package affected by body bending motion	Servoelastic oscillations on ground due to local structural vibration of bulkheads or support beams	ASE instabilities (anti-symmetric oscillation) at frequency of 6.5Hz during 8^{th} and 10^{th} flight test	ASE instabilities at frequency of 6.7 Hz (outer wing antisymmetric torsion mode) and 8.3 Hz (fuselage lateral bending), which were predicted during the analysis stage	Unacceptable 5.0~6.0 Hz oscillation and pilot coupled oscillation at low altitude and high speed in some external store configurations.	"Pilot-in-the-loop" structural coupling and ASE instabilities in high AoA	"Pilot-in-the-loop" structural coupling in ground test
Solutions	Move sensor package from tail gunner's compartment to a position of relatively small body bending motion.	Change accelerometer mount	Reduce roll command gain slightly up to 75% and update notch filter (reducing magnitude 0.4 and adding phase angle 70°)	Update notch filters	Update Flight control system (FCS)	Update Structural filter design in pitch stick command path and lateral-direction control structure	Design scheduled structural filter in pilot's command path

Fig. 2 Historical review for airframe interaction of the military fixed-wing aircraft

3.1 B-36 Peacemaker and B-52 Stratofortress

In 1948, Convair's B-36 Peacemaker [9] experienced structural coupling instability due to structural coupling between autopilot and body bending motion. The cause of this problem was that a significant body bending motion signal measured at a sensor package located in the tail gunner's compartment was fed back to the autopilot. To solve this, the sensor package was moved to a position of relatively small body bending motion.

Ironically, about 25 years later, Boeing's B-52 Stratofortress [28, 29] experienced structural coupling instability combined with the automatic flight control system and the local structural vibration at sensor mounting locations, in flight. This was caused by the influx of local vibration into the flight control, due to the flexibility of bulkheads or sensor support beams on which acceleration sensors were mounted. Meanwhile, these characteristics were not found by analysis, because the detailed model of the local structure was not included in the dynamic mathematical model. The structural-coupling oscillation causing the instability was eliminated by changing the structure on which the accelerometer was mounted.

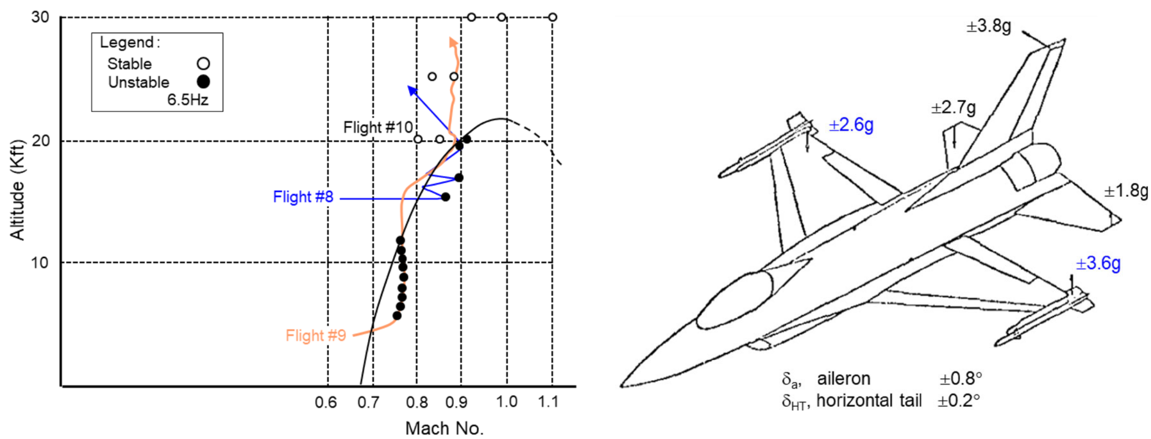
These development cases show that the location and stiffness of the surrounding structure on which a motion measurement sensor is mounted significantly affect the instability of the structural coupling.

3.2 YF-16 Fighting Falcon

During initial flight testing, General Dynamics' YF-16 Fighting Falcon [30, 32] experienced structural coupling instability associated with the flight control system and aeroelastic mode that was not verified by analysis and ground testing. As shown in Fig. 3a, an instability that occurred during the eighth-to-tenth flight tests was caused by anti-symmetric oscillations of constant amplitude sinusoidal motion at a frequency of 6.5 Hz.

The roll pulse excitation input was the most effective in exciting the oscillation at subcritical points, but the aircraft was self-excited and became unstable; the unstable boundary was penetrated. Figure 3b shows the measurements of the accelerometer mounted on the aircraft when the instability occurred. Similar to the measured acceleration characteristics, the pilot and the chaser's pilot described the form of instability as pitching motion of a missile mounted at the wingtip and the oscillations were of sufficient magnitude to be readily visible.

As a way to solve this problem, several control law design methods were considered. The first method was to reduce the manual gain roll (MGR), the control gain of the roll channel within the control law, compared to the reference value, and when it was reduced to 50%, the instability disappeared. Second, a notch filter was designed in the roll angular rate feedback loop for attenuation in the 6.5 Hz frequency band. Based on the results of the trade-off study, the final applied



(a) Instability for YF-16 flight testing (b) Instrumentation reading during instability

Fig. 3 YF-16 flight test result [30, 32]

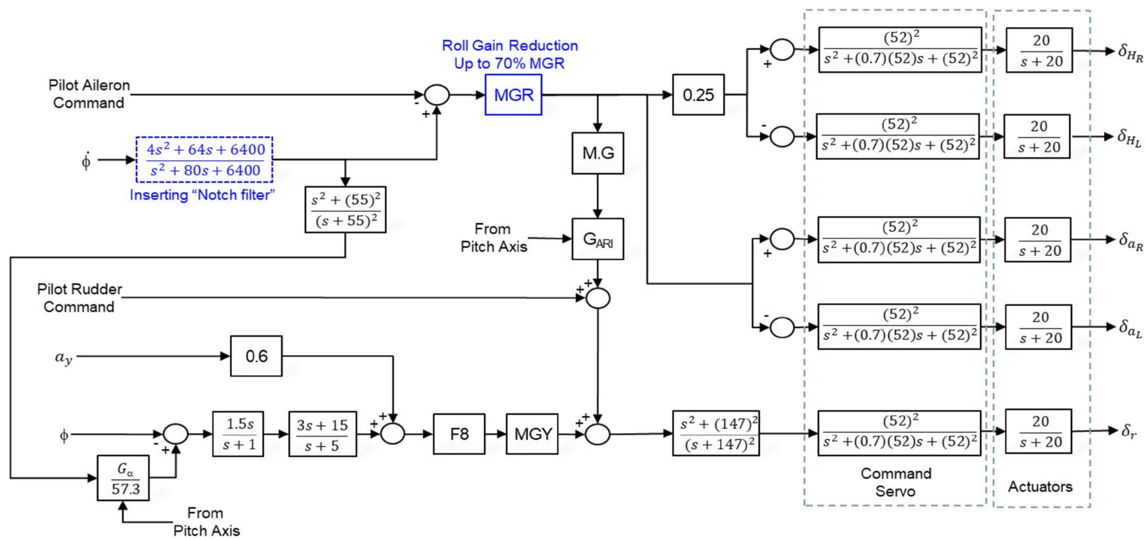


Fig. 4 Update longitudinal control law for Refs. [30, 32, 38]

improvement design is shown in Fig. 4 as follows: reduce the MGR control gain of the roll channel to 75% compared to the reference value, and apply a notch filter to the roll angular rate feedback loop. When the frequency characteristic varies from 0 to 6.5 Hz, the magnitude of the notch filter varies from 0.0 to 0.4, and the phase angle varies from 0 to 70°. The net effect on the unstable loop on the Nyquist plot is to reduce the amplitude and add approximately 70° phase shift. This medication did not adversely affect any other natural mode.

3.3 YF-17 Cobra

During development, Northrop's YF-17 Cobra [33], air combat fighter, experienced three instabilities caused by the interaction between airframe dynamics and the control augmentation system (CAS). The first type of instability was caused by the combination of CAS with rigid-body structural dynamics and the second type was caused by elastic structural dynamics. The third type was ground instability, also known as ground resonance.

In ground tests, the coupling between CAS and rigid aerodynamics caused instability in the roll and yaw axes, while not an issue in the pitch axis. The marginal damping in which yaw rate feedback destabilized fuselage lateral bending at 8.3 Hz frequency was eliminated by designing a notch filter into the control feedback at 8 Hz frequency. However, the analysis was not completed to design a suitable notch filter to address roll axis instability before the first flight, due to program schedule constraints. Considering this, the first flight was performed by reducing the roll control gain of the control system to 15%.

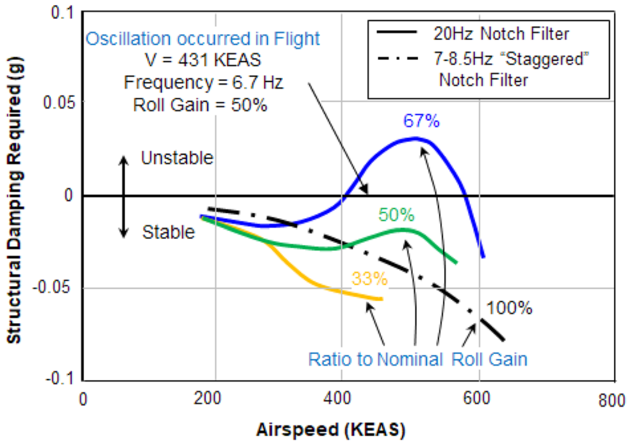


Fig. 5 Predicted damping in wing torsion mode for roll gain levels [33]

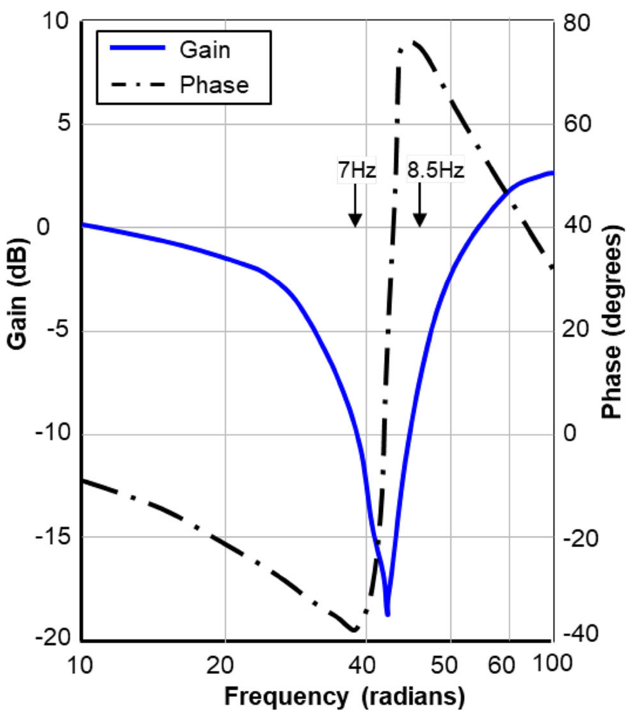


Fig. 6 Gain and phase plot of notch filter in roll CAS channel [33]

When the roll control gain was increased to 50% in the flutter flight test, a neutrally stable oscillation with a frequency of 6.7 Hz occurred at an altitude of 15,000 ft and a speed of 430 KEAS. The flight test results were very similar to the analysis results, as shown in Fig. 5.

A notch filter was designed as a method to decouple the roll CAS from airframe dynamics and increase roll control gain. At this time, the notch filter offered an acceptable compromise between the limit cycle oscillation (LCO) and elastic coupling due to phase lag at low frequency and elastic coupling constraints. The notch filter was finally selected as a "staggered" type. As shown in Fig. 6, the "staggered" type notch filter has the advantage of reducing the phase lag at low frequencies by independently adjusting the width and depth of the notch. In Fig. 5, it is found that with this type of notch filter application, the predicted damping in wing torsion mode for a 100% roll control gain (dashline) is virtually identical to the unaugmented system.

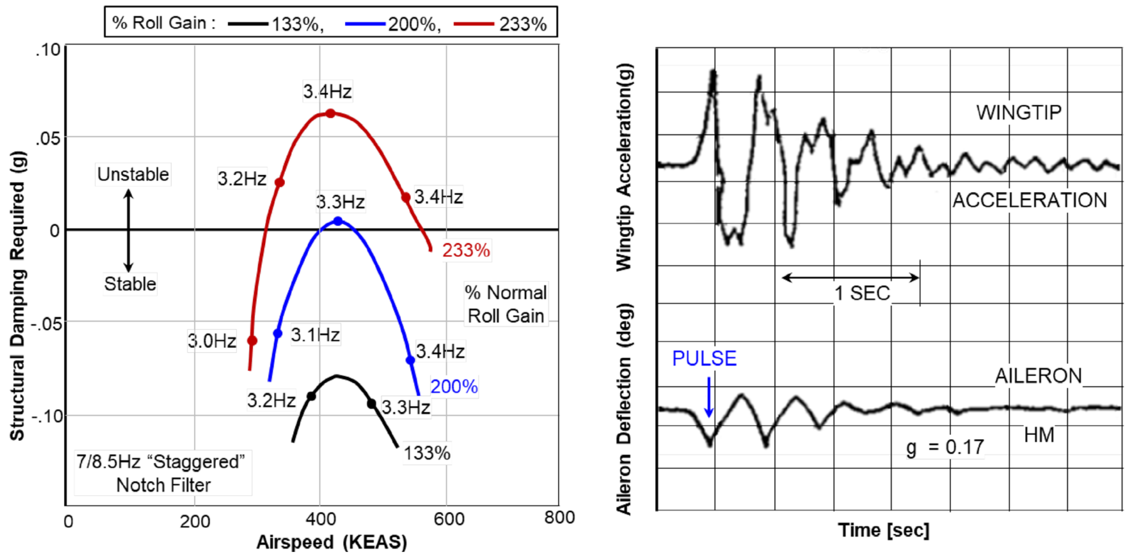
Figure 7a shows the damping tendency and frequency of the flight limit cycle divergence mode when the roll control gain changes based on the designed notch filter, which satisfies the gain margin of 6 dB. And Fig. 7b shows the response to the wingtip acceleration and aileron hinge moment following lateral control stick impulses with the roll control gain increased to 150% at Mach number 0.85 and altitude 15,000 ft. It can be seen that the maximum selectable roll control gain is not high enough to provide a positive form of the predicted rigid-body instability, but it takes about four cycles for the damping in the mode. The mode's damping increased as speed increased to a limit speed, and, as a result of analysis, instability occurred in certain elevated flight conditions where the roll angular rate control gain is more than doubled.

3.4 F/A-18 Super Hornet

Boeing's F/A-18 Super Hornet [34, 35] is equipped with MK-84 Low Drag General Purpose (LDGP) Stores on outboard wing pylons and AIM-9 Sidewinder missiles on wingtips. It experienced unacceptable vibration at 5.0–6.0 Hz due to structural aerodynamic interactions in the altitude and speed flight envelope. The oscillations typically were in the form of a limit cycle and occurred only at altitudes less than 12,000 feet and at speeds greater than Mach number 0.80. Due to these vibrations, the level of lateral accelerations at the pilot station was less than 22 Hz exceeded levels allowed by paragraph 3.1.3 of MIL-A-8870B which specifies a maximum of 0.1 g.

As summarized in Fig. 8, Boeing experienced vibrations for structural coupling with the flight control system while attempting various design changes during Phase I and II developments to solve these problems. In Phase I, the initial approach was to apply a method of symmetrically biasing the control surfaces of leading edge flaps (LEF), ailerons, and trailing edge flaps (TEF) through FCS scheduling. While the symmetrical upward deflection of the LEF and aileron adequately damped vibration, the load on the LEF increased. After that, additionally applied TEF scheduling changes for wing load alleviation further exacerbated the oscillation. The aeroelastic oscillation coupled with the lateral control stick position transducer greatly increased when the pilot inputs control commands to the stick. The frequency of these oscillations was in the 5 Hz frequency band.

Figure 9 shows the analysis result for structural coupling due to lateral control stick for F/A-18 Super Hornet. As shown in Fig. 9b, a high-frequency pilot model and a stick



(a) Predicted flight limit cycle damping for various roll gains after staggered notch filter implemented

(b) Damping recorded following lateral control stick impulses with 150% roll gains after staggered notch filter implemented

Fig. 7 Predicted damping characteristics for staggered notch filter various roll gain setting [33]

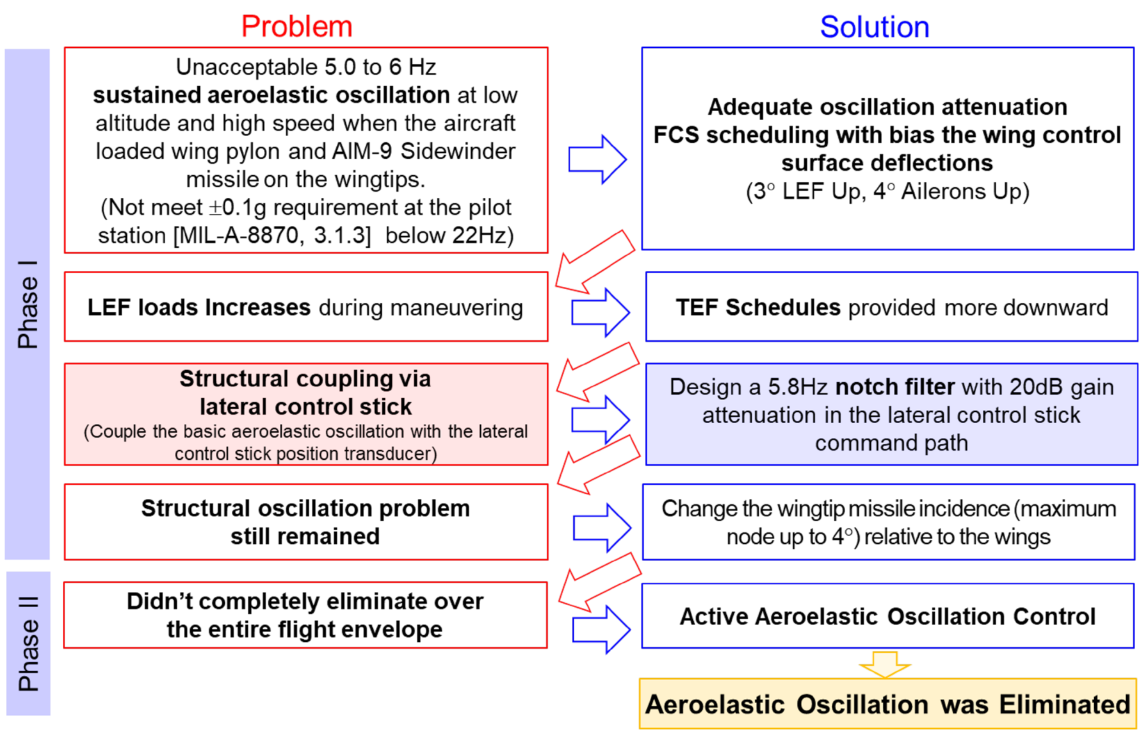
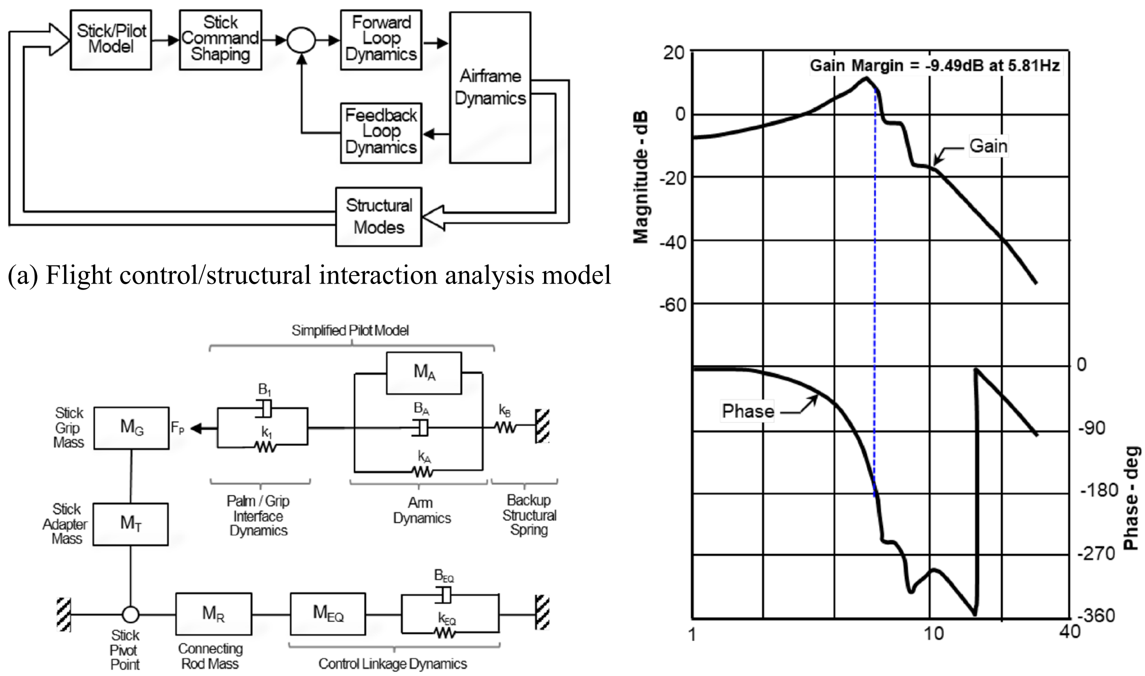


Fig. 8 Design activities to improve the instability for F/A-18 Super Hornet [35]



(a) Flight control/structural interaction analysis model

(b) Lateral stick and pilot model

(c) Lateral stick open-loop frequency response M0.85 and altitude 5,000 ft

Fig. 9 Structural coupling via lateral control stick for F/A-18 Super Hornet [35]

model were developed to identify the path of FCS and structural defects. As a result of the analysis, as shown in Fig. 9c, it was predicted that instability would occur in the lateral control stick open-loop frequency response at 5.8 Hz. Based on the analysis results, as a result of designing a 5.8 Hz notch filter with + 20 dB gain attenuation on the control stick path of the lateral axis, it was confirmed through a flight test that the FCS and structural coupling were eliminated. However, the basic vibration problem remained. To solve this, a design method in which the missiles were rotated, nose up, to maximum of 4 degrees relative to the wing was applied. However, since it was not possible to find a practical wing reconfiguration that would reduce vibration occurred over the entire flight envelope, an active control approach was applied in the Phase II development stage. Active Oscillation Suppression System (AOSS) incorporated into the F-18's existing flight control system could suppress the oscillation known as limit cycle oscillation by utilizing rate gyroscopes and accelerometers to sense anti-symmetric motions and adjusting ailerons through control laws which are based on Mach number and altitude in the interested flight envelope [39, 40]. Also, the LCO was eliminated by increasing the inner wing torsional stiffness 34% as demonstrated by flight testing on the Super Hornet [41].

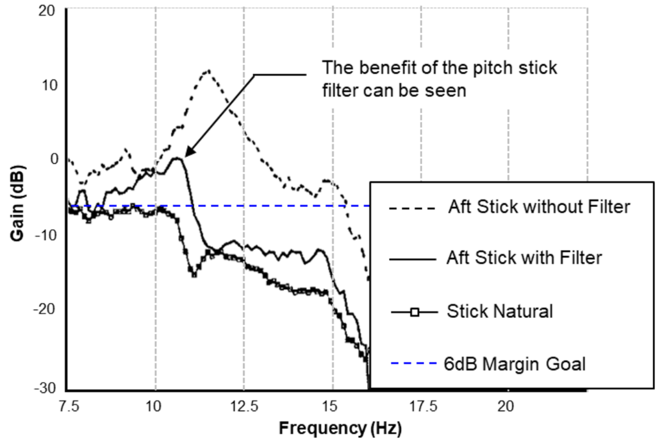
3.5 F-22 Raptor

Lockheed Martin's F-22 Raptor [36] experienced several structural coupling and ride quality problems during the development stage. In this chapter, three issues caused by structural coupling are introduced.

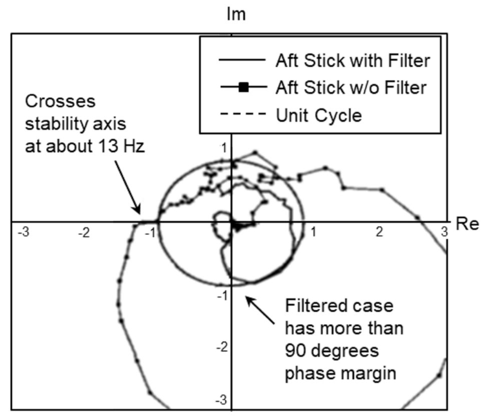
The first issue is the pilot-in-the-loop structural coupling observed during flight testing. This was shown during a 2 g turn maneuver by inputting the aft control stick input in the afterward direction by the pilot, and as shown in Fig. 10b, the instability occurred by the effect of phase, at a frequency in the 13 Hz frequency band slightly higher than the 11 Hz frequency band which is the vertical fuselage bending frequency.

As a result of constructing and analyzing a simple analysis model which applies the pilot gain estimated by calculating the transfer function between the acceleration at the cockpit and the resulting control stick force, it was confirmed that there was a coupling problem between the pilot and the structure in some flight envelope. To improve this, a simple notch filter was designed and applied in the control stick command path by minimizing the phase loss in the low-frequency band, with the requirements of processing the flight control computer as constraints.

Figure 10 shows the pilot-in-the-loop frequency response to symmetric horizontal tails (HT) input with a notch filter, compared with the response without a notch filter. After



(a) Open-loop feedback to symmetric HT input



(b) Nyquist plot

Fig. 10 Pilot in the loop frequency response for filtered and unfiltered [36]

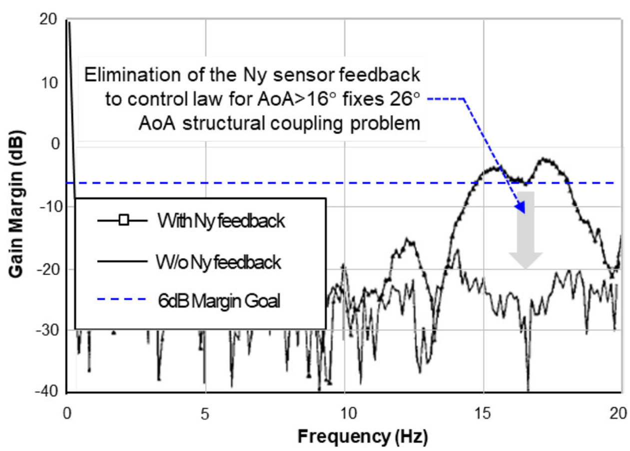


Fig. 11 Open-loop feedback due to anti-symmetric HT input at M0.3/30K/26° AoA for 26° AoA concern with new OFP 25 [36]

applying the notch filter, it can be seen that the gain margin has more than 6 dB when not pulling the stick, although it is still high around 0 dB when the pilot is pulling the stick in the aft direction. And it can be confirmed that the phase margin of the filtered design is more than 90°. Instability caused by the pitch loop did not occur in the flight test anymore.

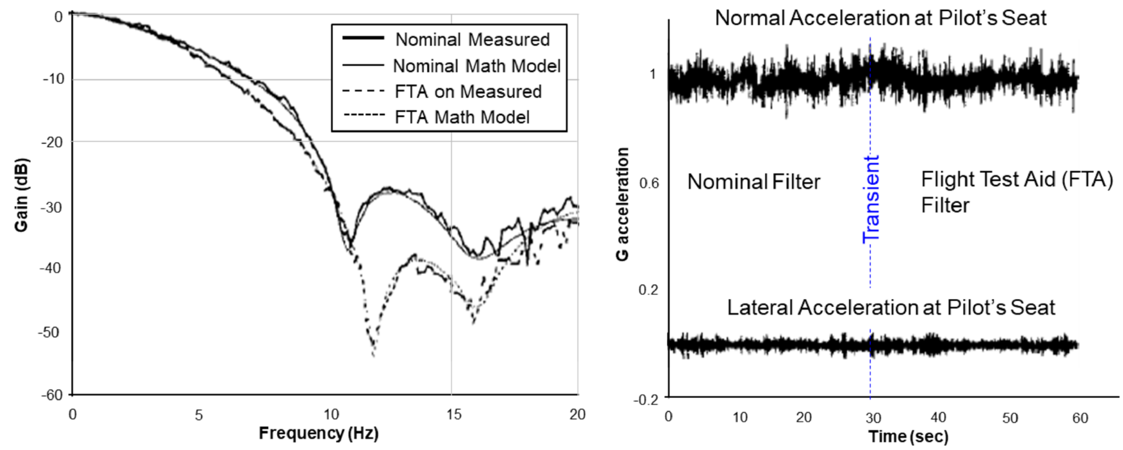
The second issue is the instability due to the coupling between FCS and structural responses by the effect of the control law feedback measured by the Ny sensor. The F-22 lateral-directional control uses roll rate, yaw rate, angle-of-sideslip, and lateral acceleration (Ny), and roll attitude angle (bank angle) as feedback variables. And each feedback control gain is scheduled according to the angle-of-attack. As shown in Fig. 11, the structural coupling test confirmed that there is potential instability as the Ny feedback is coupled with the structural mode at a specific angle of attack. To solve this problem, a design that eliminates Ny feedback above 16° of attack was applied. As a result of this design, it can be seen that it is very stable at 26 degrees of attack.

A final issue is that during the early stages of engineering and manufacturing development (EMD) development a Gravel Road characteristic has been reported by pilots in landing approach, since it feels like the plane is being driven over a rough surface. At this time, the dominant frequency band is about 12 Hz, which is the vertical fuselage bending mode. Since this riding quality was likely caused by the coupling between FCS and structure, a subsequent flight test was conducted after applying a structural filter as an option in the pitch angular rate control command stage using the flight test aid (FTA). Figure 12 shows the level of acceleration measured at the pilot seat with and without the filter applied obtained from the flight test, but there was no difference. In conclusion, the prevailing opinion is that the cause of the rough ride was not the coupling between FCS and structure, but was caused by separated flow impacting the horizontal tails.

3.6 X-35B Joint Strike Fighter (JSF)

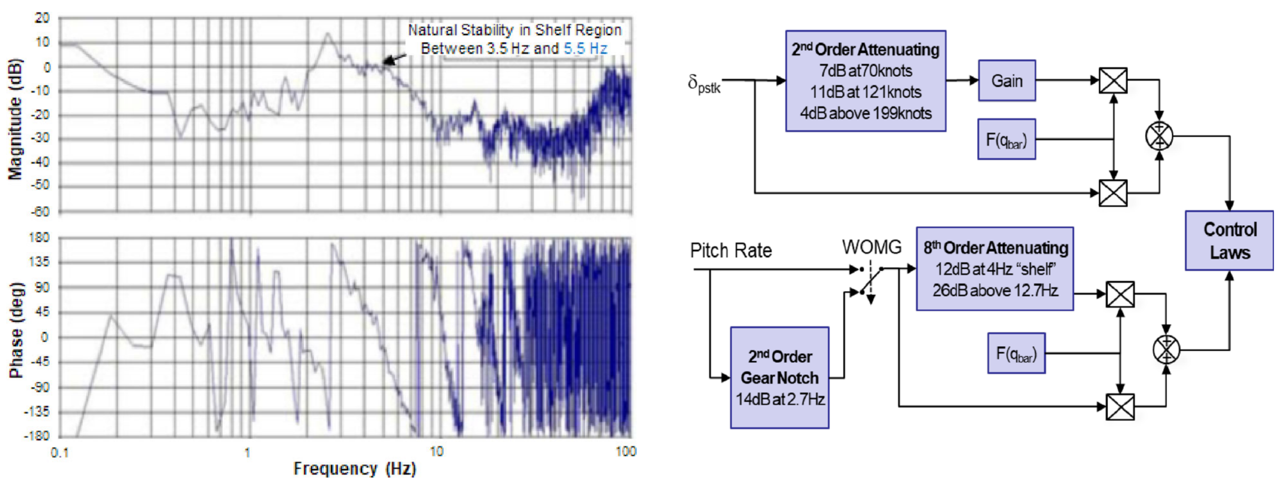
The Short Takeoff and Vertical Landing variant of Lockheed Martin's JSF (X-35B) had some problems in verification for structural coupling stability due to several FCS and structural coupling issues. In particular, instability due to inertial coupling through the pilot and control stick in a relatively low-frequency band due to the high inertia of control surfaces has become a major issue. This section introduces a design method that solves the instability caused by structural coupling while complying with the airworthiness requirements in the low-frequency band.

Rigid body structural coupling can occur when a high-inertia control surface is combined with a high control gain. Typically, these coupling modes appear in the form



(a) Gravel road structural filter vs. nominal (b) Time history of pilot seat acceleration

Fig. 12 Gravel road characteristics for F-22 Raptor [36]



(a) open-loop response (neutral stability case) (b) Structural filters in longitudinal control structure

Fig. 13 Open-loop response and structural coupling filter of pitch axis for X-35B [37]

of a "shelf" in the broad low-frequency range, as shown in Fig. 13a [37]. And the high-inertia control surface coupled with the control system through the pilot's arm dynamics along with the movement of the stick can cause instability. The possibility of such instability is due to the low-speed flight regime designed with a large control gain and the side control stick where the small range of motion of the pilots' hand is required to obtain a relatively large surface deflection. It is most likely to occur in the case of side stick controller.

In the X-35B, this instability was observed in pitch at low speeds where the control system gain is high, and the mode may be quite violent. To address this, a filter attenuation of 10 dB was provided for the pitch stick above 8 Hz in the pilot pitch stick input. As shown in Fig. 13b [37], excessive time-delay in the low-frequency band is prevented by applying a method to schedule the filter according to the dynamic pressure, so that the filter can have maximum attenuation at 120 knots and less attenuation at other speeds.

4 Related Requirements

This chapter describes the main requirements presented by aircraft military specifications in relation to structural coupling characteristics. Stability margin and flying qualities are the main requirements to consider when developing an aircraft.

4.1 Stability Margin Requirement

Regarding stability, MIL-STD-9490D [20] presents the structural mode redundancy requirements, as shown in Table

Table 1 Gain and phase margin requirements [20]

Mode Frequency, f_M \ Airspeed (kts)	V_s to V_{oMIN}	V_{oMIN} to V_{oMAX}	V_{oMAX} to V_L	At 1.15 V_L
$f_M < 0.06$	GM= 6 dB (No Phase Requirement Below V_{oMIN})	GM = ±4.5 dB PM = ± 30°	GM = + 3.0 dB PM = + 20°	GM = 0 dB PM = 0° (Stable at Nominal Phase and Gain)
$0.06 < f_M <$ First Aeroelastic Mode		GM = ± 6.0 dB PM = ± 45°	GM = + 4.5 dB PM = + 30°	
$f_M >$ First Aeroelastic Mode		GM =± 8.0 dB PM = ± 60°	GM =+ 6.0 dB PM = + 45°	

Note :

V_L	: Limit airspeed (MIL-A-8860).
V_{oMIN}	: Minimum operational airspeed (MIL-F-8785)
V_{oMAX}	: Maximum operational airspeed (MIL-F-8785)
Mode	: A characteristics aeroelastic response of the aircraft as described by the aeroelastic characteristic root of the coupled aircraft/FCS dynamic equation of motion
GM(Gain Margin)	: The minimum chain in loop gain at normal phase, which results in instability beyond that allowed as a residual oscillation.
PM(Phase Margin)	: The minimum change in phase at normal loop gain which (Phase Margin) results in instability.
f_M	: Mode frequency in Hz
Nominal	: The contractor's best estimate or measurement of FCS and aircraft phase and gain characteristics Phase and Gain available at the time of requirement verification

1, while MIL-A-8870C requires a phase margin of 60° as the criteria of aeroservoelasticity [42]. In general, structural mode damping increases with increasing speed, but MIL-STD-9490D specifies that a minimum gain margin of ± 6 dB must be satisfied for both aerodynamic and non-aerodynamic closed loops at zero airspeed. For the operational airspeed envelope, it requires a minimum gain margin of ± 8 dB and phase margin of ± 60° in the first aeroelastic mode or higher frequencies. And at frequencies below the first aeroelastic mode, both ± 6 dB gain margin and ± 45° phase margin are considered as phase stabilization requirements. Meanwhile, ± 6 dB gain margin is applied as a gain stabilization requirement, since most of the structural modes are at a frequency too high to maintain the phase relationship. That is, this is because phase can be affected by actuator wear, hydraulic pressure, partial hydraulic system failure, timing changes between calculations performed in different parts of an asynchronous system, and relative changes in the gains of different parts of a multipath system [37].

4.2 Flying Qualities Requirement

MIL-STD-1797A [27], a military standard document, presents the equivalent time-delay standard for aircraft systems, which is up to 100 ms for flight qualities level 1. Equivalent time-delay can be calculated in the frequency domain with the low-order equivalent system (LOES) technique.

From the pilot's point of view, the time-delay in an aircraft system is the dead time it takes for the aircraft response to start after the pilot applies force to the control stick, caused by various sources within the flight control system. Excessive system time-delay slows the response of the aircraft to pilot input, thereby degrading flight qualities and becoming a major cause of aircraft accidents such as pilot-induced oscillation (PIO) [43, 44]. Therefore, the time-delay requirement required by the military specifications must be considered in the flight control system design process.

As mentioned above, the notch filter, which is a representative design method for reducing the structure coupling characteristics in the high-frequency band, causes a phase lag in the low-frequency band that affects the motion of the rigid body. Therefore, a notch filter must be designed considering the phase lag of the low-frequency band as a constraint.

5 Structural Coupling Compensation Design and Testing

5.1 Structural Coupling Loop

As shown in Fig. 14, the feedback control loop consists of an aircraft, sensors, flight control system, and actuators [26]. Here, aircraft which has both rigid and flexible characteristics can be divided into the control surface dynamics such as HT and flaperon, and the airframe such as wings and fuselage.

Aircraft exhibit several vibration modes, which has a mode shape that represents the relative motion between the various parts of the airframe when excited at a resonant frequency. In particular, there are many modes of aircraft within the bandwidth of the FCS depending on fuel states, flight conditions, and external stores [45]. The mode can be excited or forced by vibrating each control surface with input in the ground test,

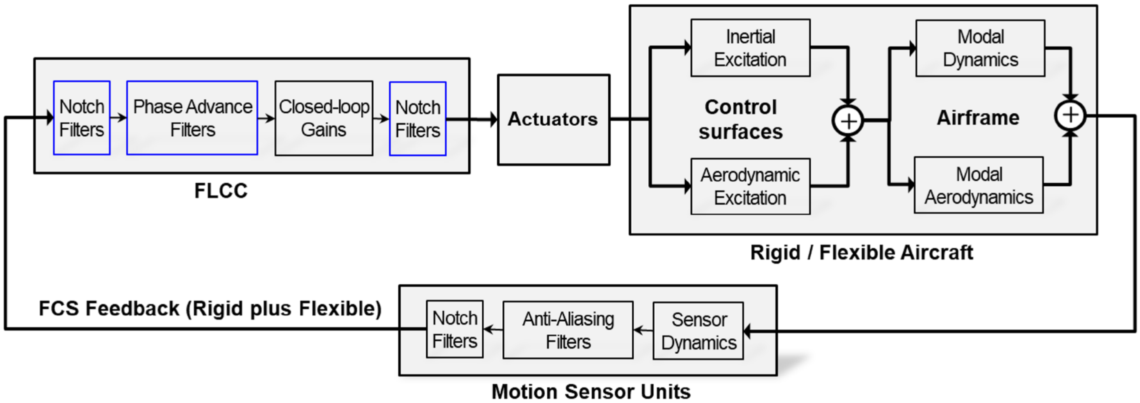


Fig. 14 Structural coupling loop [26]

and the response of the aircraft, such as pitch angular rate, can be measured by monitoring the outputs of the AMSU sensor. Here, the magnitude of the response peak corresponding to the structural mode resonance is a function of the efficiency with which the control surface activates a particular mode and an amplitude of aircraft motion measured by the AMSU sensor.

An excitation force applied to the structure by exciting the control surface is composed of inertial component and aerodynamic component. The inertial component is due to the offset of the control surface's mass from its hingeline and is proportional to the surface's acceleration. And the aerodynamic component is due to a change in the aerodynamic pressure distribution on the control surfaces or adjacent surfaces, which is proportional to the magnitude of control surface defections and the dynamic pressure.

A system response to the excitation force is mainly affected by aerodynamic damping and stiffness, but its effect is secondary in the low-speed flight regime with large control gains, where critical flight conditions for structural coupling are often found. For example, aerodynamic effects are dominant on relatively light flaperon control surfaces mounted on aerodynamically powerful wings, while inertial effects are dominant on rather heavy all moveable HT control surfaces.

Since the control gains in the control laws are generally designed based on the rigid-body flight mechanical model, not considering the aeroelastic model of flexible aircraft, the theoretical control law design requiring high control gain is unrealistic for aircraft with structural coupling. Therefore, the designer must design the control law considering this aspect. SCT, which is performed on the ground prior to the first flight, can prove that there is no instability and the gain margin requirement is satisfied in the closed control loop.

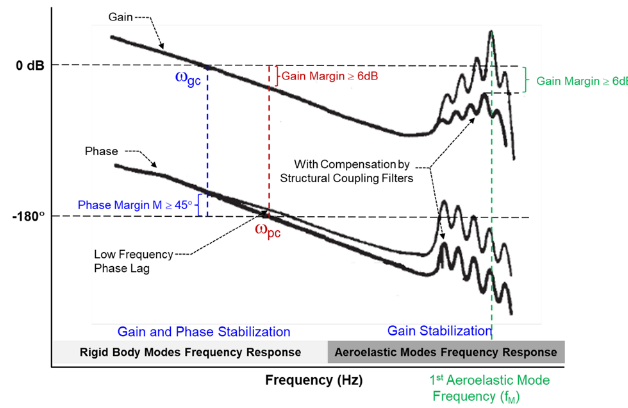


Fig. 15 Structural coupling compensation concept [21]

5.2 Compensation Concept

The control gain in the flight control law is designed with a budget of phase loss due to the application of the notch filter required to minimize the aeroelastic effect. As shown in Fig. 15, the most common design method for minimizing structural coupling characteristics is to apply a combination of notch filters in the structural vibration frequency band in question to satisfy a gain margin requirement of at least 6 dB, so that it can attenuate the structural response [21]. However, this design method introduces additional undesirable phase lag at low frequencies. It is most obvious that the phase lag in the low-frequency band translates into a pilot-perceived time-delay, adversely affecting the handling qualities of the aircraft. Moreover, the effect of time-delay can have even more serious consequences for aircraft with unstable static stability. It is therefore less clear that control law engineers consider that stability requirements must be met at the expense of an aircraft's handling qualities [46].

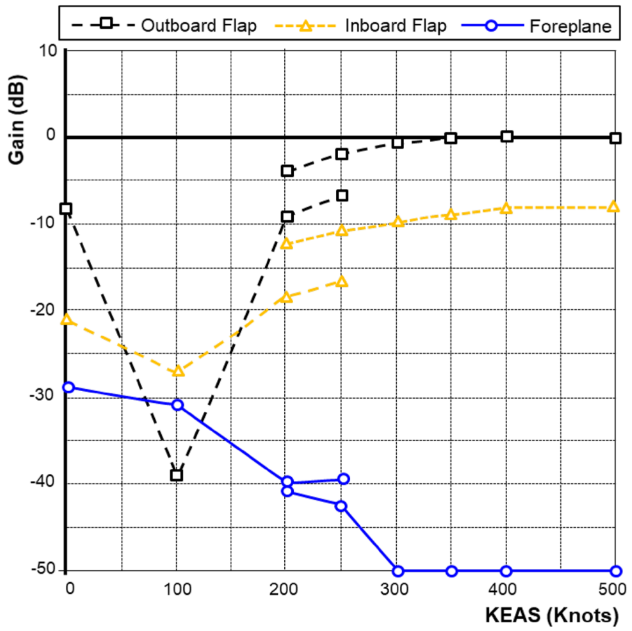


Fig. 16 Variation of first wing bending mode for each control surfaces [26]

5.3 Design Consideration

An integrated aircraft design approach is needed to effectively reduce the undesirable response in the high-frequency band and minimize the phase lag in the low-frequency band in the notch filter design process. This chapter reviews the design methods that should be considered to minimize structural coupling in the aircraft development stage.

5.3.1 Minimum Weight and Inertia of Control Surfaces

As introduced in the previous chapter 3 of the structural coupling of the F-22 [36] and the X-35B [37], the high-inertia control surfaces cause structural mode coupling problems by coupling rigid or airframe flexibility characteristics with high control gains or pilot's dynamics. Figure 16 shows the effect of each control surface on the response of a flexible aircraft in first wing bending mode as the speed changes. Here, low-weight and inertia control surfaces such as flaperons mounted on the main wing have a dominant aerodynamic effect as the speed increases, while high-weight and inertia control surfaces, such as all moveable horizontal tails or foreplanes, are dominated by inertial excitation forces in the low-speed regime, not in the high-speed regime, although the values are small. Therefore, a design that minimizes the weight and inertia of these control surfaces must be considered, since the effect of all moveable horizontal tails or foreplanes' control surfaces is dominant in the low-speed flight envelope with high control gain where structural coupling mainly occurs. This control surface design method significantly reduces the structural coupling effect at high frequencies between 20 and 80 Hz [46].

5.4 Optimum AMSU Positioning

The AMSU, a gyro mounted on an aircraft, measures essential information, i.e., 'rigid' aircraft rates and accelerations, along with 'flexible' aircraft rates and accelerations at the frequencies of the aircraft elastic mode. Because of this, the 'flexible' aircraft angular rates and accelerations measured by the AMSU are passed through the flight control system control paths and multiplied by FCS gains and FCS filters and they are inserted into the control surface actuator input which then drives the control surfaces at the frequencies of the elastic modes of the aircraft, which creates unstable structural coupling characteristics.

As a design method to reduce instability due to structural coupling, the design method of placing the AMSU at an ideal position, or "sweet spot", with the least movement as a whole in an aircraft structure is widely known [45]. Figure 17 shows the pitch angular rate response to excitation of each control surface obtained from the ground structural coupling test of the forerunner of Eurofighter (i.e., the Experimental Aircraft Program, EAP) [26]. In the EAP program, priority was given to the location of the AMSU for easily supplying cooling air to the electronic devices rather than the consideration of the structural coupling of the motion sensor. As a result, the pitch angular rate response to foreplane excitation was large due to the fuselage bending mode in the 15 Hz frequency band. And the high gain at 55 Hz in Fig. 17 was traced to local bending of the AMSU mounting plate between the four units fixed to it. As can be seen in this development case, it can be seen that the mounting position of the motion sensor and the stiffness of the mounting plate have a significant effect on the structural coupling instability.

Figure 18a shows the mounting positions of the motion sensors, RSA (rate sensor assembly) and ASA (accelerometer sensor assembly), in the AFTI/F-16 fighter [47]. The AFTI/F-16 employs the same four RSAs and ASAs as the F-16 production version, of which three sensor signals are integrated with a triplex flight control system. RSA, which measures angular rate, is located at the anti-node where angular motion occurs the least. The ASA, which measures normal and lateral acceleration, is located at a node in front of the aircraft's center of gravity (CG) as close as possible to the center of percussion (CP), to obtain the lowest vertical deflection motion. Similarly, the F-22 Raptor has accelerometers near the cockpit and rate gyros about 150 inches aft of the cockpit [36].

As another design method, in EAP equipped with a motion sensor that simultaneously measures angular rate and acceleration, as shown in Fig. 18b, the AMSU was placed at the anti-node where the pitch rate and yaw rate due to the elastic

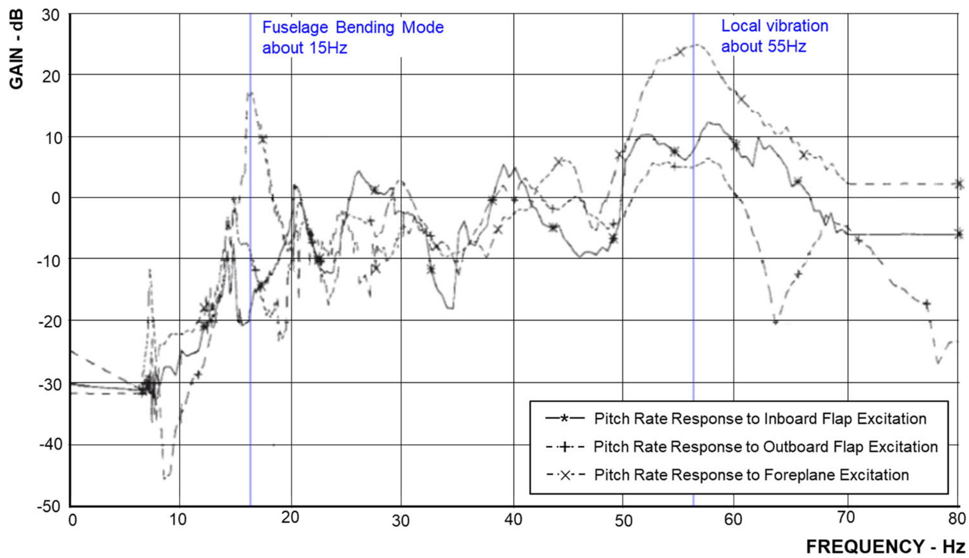


Fig. 17 Typical ground structural coupling test data for EAP aircraft [26]

modes far from the node of the primary fuselage bending mode were minimized [19, 24–26].

5.4.1 Stiffening AMSU Platform

Figure 19 shows the detailed mounting design of the RSA and ASA, the motion sensors of the AFTI/F-16 aircraft [47]. The structure to which the motion sensor is mounted must be very stiff, and its natural frequency must be higher than twice the sampling frequency of the flight control computer [24, 46, 48] to prevent attraction of local elastic frequencies. If the first natural frequency of the structure around the motion sensor, including brackets and trays, is low, structural vibration characteristics may affect signals in the low-frequency signal band due to aliasing characteristics by digital sampling of the flight control computer. In addition, actual manufacturing sensitivity should be considered when selecting motion sensor locations [49]. For this reason, the first high frequency of the AMSU structure must be high enough to minimize the aliasing due to the local vibration of the AMSU structure which is within the sampling frequency of the flight control computer. As an example, Luber W, Becker J, and Sensburg O designed the vibration frequency of an AMSU tray to have a minimum 140 Hz [48].

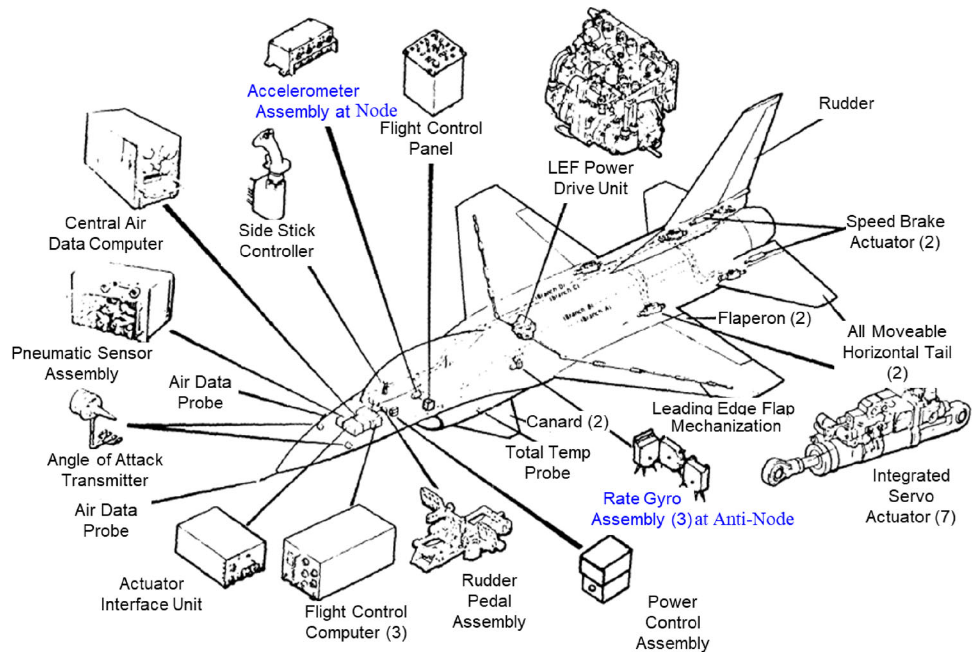
5.4.2 Avoiding Excessive High Gain Condition

In general, control gains in the flight control system are designed based on a rigid aerodynamic model that reflects wind tunnel test results. This theoretical control law design approach requires high control gains to achieve the desire for “Level 1” flying qualities in the low-speed flight regime where dynamic pressure is small and control power is insufficient [37]. As a result, high-performance fighter aircraft with high gain control systems often have multiple closely coupled structural modes which can interact unfavorably with their feedback control systems [32, 37, 50, 51].

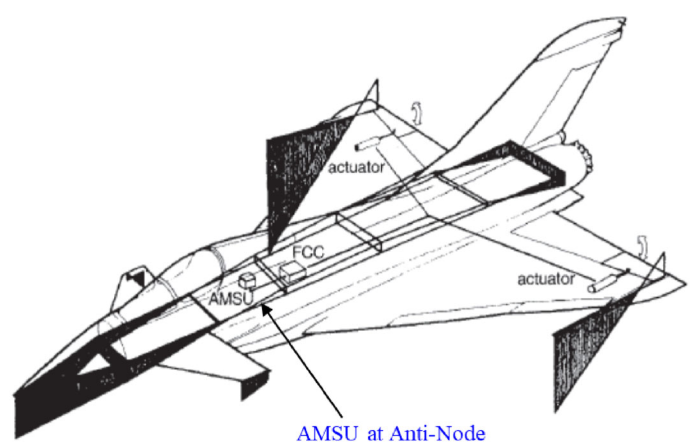
In fact, control gains designed based on mathematical models to satisfy flight performance requirements do not always lead to good flight characteristics. In other words, although the theoretically designed control gain is reduced, flying performance is not always degraded proportionally. In particular, it does since the center of aerodynamics moves forward in the low-speed flight regime and then the aircraft become statically stable. That is, in a case that the control gain designed initial to be high reduces, the effect of the reduced range of the control gain on flight characteristics can be small. And, in general, the control system designed with the LOES technique has a rather large Mismatch Cost Function value of 10 or more in the low-speed flight regime, so the accuracy or reliability of design and analysis is insufficient [27]. Therefore, by scheduling theoretically designed control gains according to altitude, speed, and angle of attack, the large control gain in the low-speed flight regime can be reduced to achieve “Level 1” flying qualities, and to eliminate instability due to structural coupling.

5.4.3 Hardware Design and Minimizing Digital Effect

Flight control computers of fighter-class aircraft are, in general, designed with a sampling rate of 80 Hz or less, such as 64 Hz for F/A-50 [52, 53], 50 Hz for F-16 [54, 55], 80 Hz for EAP [56], and 80 Hz for F-35 [57], and AMSU which measures the motion of the aircraft used as a control law feedback variable operates at a high-frequency sampling rate of 400 Hz or more [58]. The aliasing phenomenon in digital systems occurs when data with a frequency higher than half the sampling frequency, i.e., Nyquist frequency, are reflected



(a) AFTI/F-16 [43]



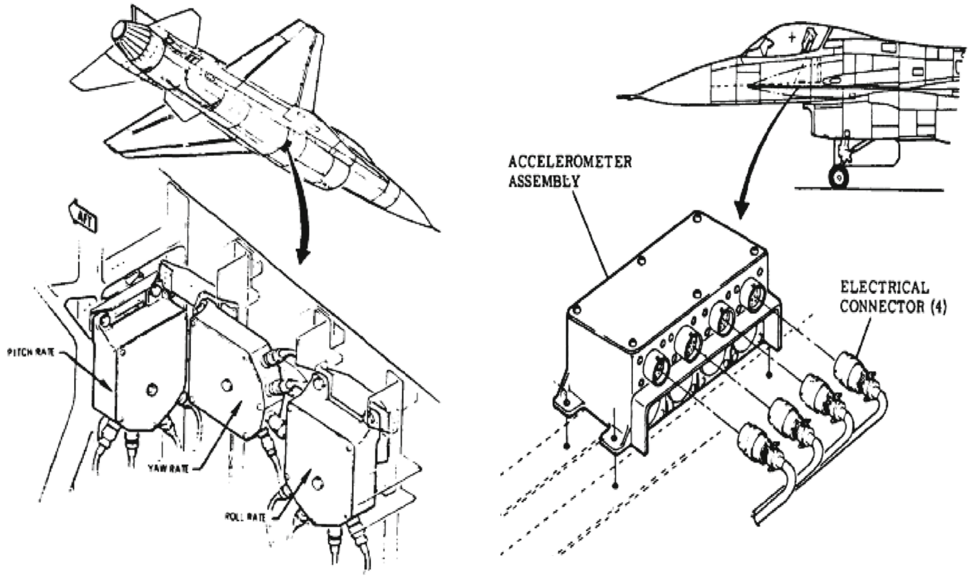
(b) Eurofighter 2000 [20][26][27][28]

Fig. 18 Position of motion sensor assembly for the fighter aircraft

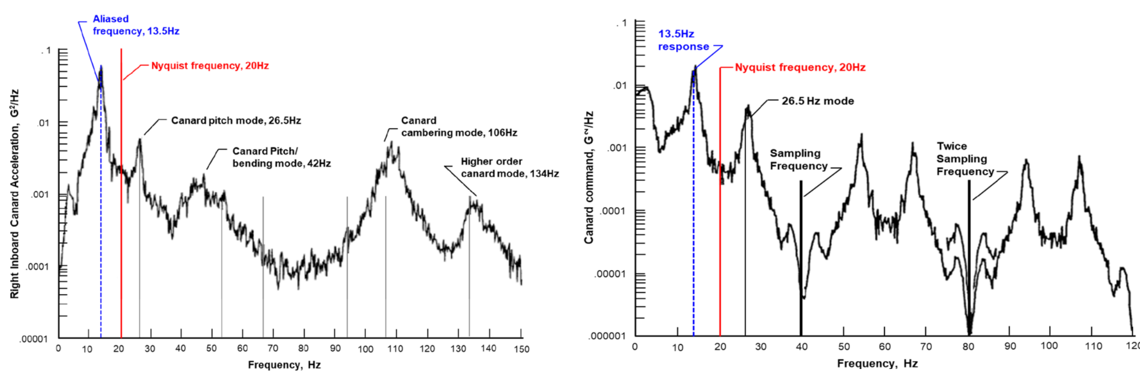
and overlapped to the Nyquist frequency below [45, 59]. For example, the Nyquist frequency of a flight control computer with a 40 Hz sampling rate is 20 Hz, so frequency data above 20 Hz are reflected back to a frequency range of 0–20 Hz.

Figure 20 shows the power spectral density plot for the inboard canard accelerometer response of X-29 [60]. The sampling rate of the X-29's flight control computer was designed to be 40 Hz. A 13.5 Hz response was observed at the canard after FCS was changed to improve aircraft handling qualities. As a result of test data analysis, it was confirmed that the lowest canard structural mode (canard pitch) is a frequency of 26.5 Hz, and there were no structural modes at this frequency. Therefore, the forced response of the 13.5 Hz frequency as the structural-mode response was aliased at each frequency in the range of 20–150 Hz (42 Hz of canard pitch/bending mode, 106 Hz of canard cambering mode, and 134 Hz of higher order canard mode), starting at the Nyquist frequency of 20 Hz. Here, these modes contribute little to the 13.5 Hz signal, since the displacement level of these modes is smaller than the 26.5 Hz canard pitch mode and is mostly attenuated by designing the roll-off of the anti-aliasing filter with 32 Hz, not with low cut-off frequency to have sufficient phase margin [55]. As a result, a 13.5 Hz forcing response was mainly observed at the canard, and it was also observed at a damped level at the stake flap and wing. At a frequency of 13.5 Hz, the amplitude of the canard command was as

Fig. 19 Motion sensor mounting for AFTI/F-16 [47]



(a) RSA (b) ASA



(a) Possible aliased frequencies for a 40 samples/sec sample rate

(b) Canard command zero order hold signal power spectral density plot

Fig. 20 Power spectral density plot for the inboard canard accelerometer response of X-29 [60]

high as 2° peak-to-peak under certain flight conditions. The canard actuator acted as a filter attenuated the signal's amplitude, so that the canard motion was only 0.30° peak-to-peak at 13.5 Hz. This is a good development practice for reducing forced responses due to aliasing, a digital effect. In other words, the digital signals of AMSU sensors operating at high frequencies can be designed to be processed in the form of rolling averages and downsampling to provide very effective anti-aliasing. And the flight control computer sampling and zero-order hold (ZOH) characteristics also can provide an attenuated signal at frequencies higher than the flight control computer (FLCC) sampling rate.

5.4.4 Structural Coupling Filtering and Parameters' Optimization

The AMSU sensor mounted on the flexible structure of the aircraft detects not only the low-frequency rigid-body motion, but also the superimposed higher frequency oscillations due to the resonances of the flexible modes of the structure simultaneously. Therefore, if the large-size signal component of the sensor outputs is not attenuated, the control law can form a closed loop with the control itself by actuating the aircraft's control surfaces and can destabilize the aircraft by amplifying the structural mode. To solve this structural coupling problem, a filtering function for attenuating structural vibration signal, that is problematic with the FCS must be designed, so that the closed-loop control is stable and prevents the performance degradation of the FCS or the damage to the aircraft structure. To design a notch filter at the structural modes in question is a widely used design method to effectively attenuate high-frequency structural modes. To effectively design the filters, design and analysis environments must be built for valid and appropriate representations of the FCS-flexible aircraft system.

Figure 21 shows the longitudinal control structure of EF-2000 [19]. To give stability to statically unstable aircraft and provide excellent maneuverability, the flight control system feeds back and augments aircraft status information, such as pitch angular rate, normal acceleration, and angle of attack from the AMSU sensor. As already mentioned, the design method for damping is to first design anti-aliasing filters in the AMSU sensor hardware to minimize the introduction of high-frequency structural vibration modes into the flight control computer, so that high-frequency signals and structural vibration responses entering the flight control computer can be effectively damped by designing a notch filter.

There are several methods for designing notch filters for structural response attenuation in flight control systems, as shown in Table 2. The first is to design a notch filter in the sensor input path. This design method is effective when the movement of the control surface directly excites the structural modes, or there is no structural response such as biodynamic Feedthrough) related to the control stick command path. The second is to design a notch filter in the sensor input and control stick command path when there is structural mode excited in the sensor input and control stick command path. The third is to design a notch filter in the actuator command path rather than sensor input path, which is an effective design method when there is a problematic structural response coupled with control surface motion and control stick commands as well as sensor feedback. However, these notch filters introduce additional phase lag in the low-frequency band. In particular, the second and third design methods have problems in adding time-delay to the control law in the path between the pilot's command and the control surface deflection, and increasing the overall phase lag of the control system, although the methods can effectively dampen the overall structural response. Therefore, in the process of designing a notch filter, minimization of the phase lag in the low-frequency band should be considered as a constraint, and PIO analysis should be also considered.

For a fighter with feedback control laws, the stability margin for a loop must be determined with the other loop closed. This means that the notch filter design for the structural-mode response with inter-axis coupling results in a complex non-linear optimization problem, and the notch filter in multiple sensor paths affects the stability margin. Notch filters for coupled axis response are usually designed through ad hoc procedures by taking one sensor loop, but the processes not only involve a great deal of trial and error, but also can lead to very conservative designs. To address these issues, various integrated design approaches have been studied [22, 23, 47]. As an example, Animesh [23] suggests an alternative one-shot procedure, in which the notch filter in each individual sensor path can be designed independently of ones in the other sensor paths, to comply with the requirements for stability margin of the whole system. By decomposing the complex optimization problem to several independent optimization problems, it is possible to design a notch filter in this situation with a single-step procedure and it turns out to be less conservative than an ad hoc procedure. And, if necessary, during design iterations, the amount of phase lag injected by the notch filter can be redistributed between different sensor paths at lower frequencies to effectively reduce the phase lag of the entire control system at low frequencies.

However, fighter jets developed in modern times are designed in lighter configurations to improve performance, and more complex designs are required, because a large number of notch filters are used as agility and higher bandwidth control systems are applied. As an example of this, EAP [56], which designed notch filters in such a way as to maximize their effectiveness while minimizing the inherent phase lag effects, applied seven notch filters only on the pitch axis. In addition, X-35B [37] designed an eighth-order notch filter and scheduled the notch filter according to the dynamic pressure, further increasing the complexity of the design.

Although it has not been applied to production aircraft, a structural coupling solution using a Kalman filter has been studied in an effort to solve this problem [25]. A Kalman filter solution can provide comparable performance to a notch filter solution in terms of attenuation of structural modes while reducing phase lag with a simpler design process. However, since the Kalman filter is dependent on the mathematical model of the aircraft, the more flexible the mode, the higher the processing requirements. Therefore, a flight control computer with improved performance considering the calculation throughput is required.

5.5 Test Consideration

SCT is to confirm that there is no instability due to structural coupling prior to flight testing. This test verifies that the gain margin in the structural mode complies with the margin requirements by directly exciting the control surfaces, while the aircraft is on the ground. This chapter describes test environments and procedures that must be considered during SCT, since the reliability and accuracy of test results are determined by the type of input, aircraft conditions, and test environment.

5.5.1 Structural Coupling Test

Among the forces that cause the airframe dynamic response, the aerodynamic and inertial forces induced by oscillating control surfaces play a fundamental role. When the two forces

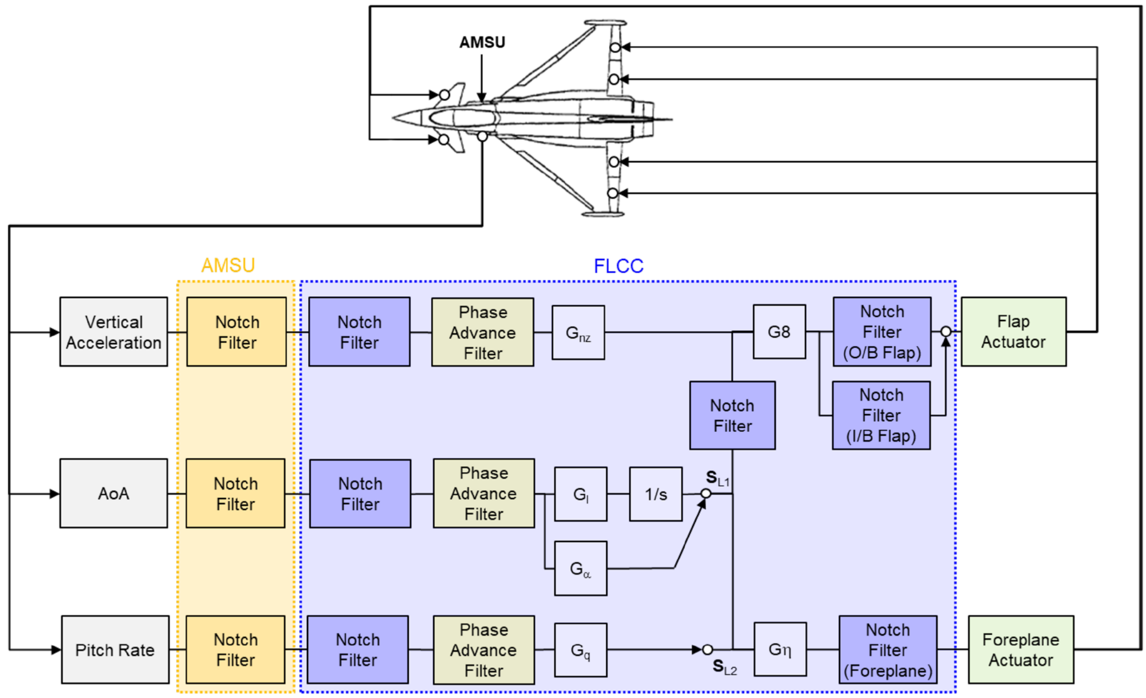


Fig. 21 Longitudinal control structure for EF-2000 [19]

Table 2 Consideration of notch filter design to eliminate structural coupling [43, 44, 51–55, 57, 61]

Location	Control Path	Objective	Remark
Motion Sensor	-	• Anti-aliasing for frequencies higher than the flight control computer sampling frequency (F/A-50: 64 Hz [49][50], F-16: 50 Hz [51][52], EAP-80 Hz [53], F-35: 80 Hz [54]) to minimize the high frequency flexible aircraft signal components	• High frequency response remaining after AMSU filtering was folded back
FLCC	**Sensor Feedback**	• Attenuate the feedback signals (rate and acceleration) from the remaining problematic airframe flexible mode below Nyquist frequency of FLCC sampling frequency	• Low frequency phase lag
	Actuator Command	• Attenuate excitation modes of particular control surfaces	• Overall low frequency phase lag including feedback and command path and PIOs [41][44]
	Control Stick Command	• Prevent the biodynamic feedthrough (BDFT) [58] • BDFT is influenced by a large number of different factors (level of aircraft accelerations at pilot seat, seating dynamics, armrest, control stick dynamics, pilot dynamics, control tasks) • BDFT varies between different persons, as well as within on person over time (highly nonlinear)	• Flying qualities and PIOs concerns due to command path phase lag [41][44]

excite a structure near resonance, it creates a very dangerous loop. Therefore, if the structure vibration contents of the motion sensor signal used as control feedback are not properly filtered and removed, the control surface command amplified by the control gain will cause unstable vibration.

Therefore, SCT is performed not only to examine nonlinearities such as structures, hydraulic, and actuator backlash that are not implemented in the linear model in the absence of aerodynamics, but also to evaluate the appropriateness of the designed notch filter, collect enough experimental data above a certain frequency where the quality of the model predictions is rather poor, and update the aeroservoelastic model. Another important purpose is to optimize the designed notch filter and obtain flight clearance and qualification for safe flight [37, 45, 47].

Simplifying the design process as much as possible and take into account the phase uncertainty of the high-frequency band, the structural coupling stability of the frequency band is demonstrated by the "gain stabilization" method in SCT, which considers only the gain margin without the condition of the phase margin [47].

5.5.2 Test Setup

To perform SCT, a special environment is required for generating excitation signals as input to actuators, acquiring aircraft structure response to excitation input, and analyzing real-time frequency response. Figure 20 shows the conceptual SCT layout of Eurofighter-2000 [18]. Automatic test equipment (ATE) is a key equipment for SCT that performs several functions such as data exchange with FLCC. The main functions of this device are to set up and read/write FLCC parameters, inject the excitation signal into the FLCC, and read the AMSU sensor signals from the FLCC facilities and provide real-time presentation of FLCC signals. A transfer function analyzer (TFA) generates an excitation signal and is integrated into the ATE and interfaced with an external PC. The TFA then performs the calculation of the open-loop frequency response function (OLFRF) and sends the relevant data to a PC for data save and subsequent analysis. The OLFRF data obtained during testing are then compared to theoretical predictions to determine in real time whether any unexpected or undesired effects are affecting the test.

Since SCT is performed in open loop as well as in closed loop, it is necessary to be able to disconnect the FLCC and the actuator. Also, for the efficiency of testing, it can be considered to include a function that disables the notch filter designed in the AMSU and FLCC or changes the main coefficient of the notch filter.

The excitation input during SCT directly affects the structural vibration of the aircraft, increasing structural fatigue and causing structural damage, so special equipment or facilities for aircraft safety are required. Equipment such as "Soft Support System" to exclude the effect of landing gear can be considered to improve the accuracy and reliability of test results. The following chapters describe these equipment and facilities (Fig. 22).

5.5.3 Soft Support System

With the flight control system closed loop, landing gear and tires can affect structural mode coupling, while the aircraft is on the ground. In particular, if the frequencies of the landing gear and tire mode are adjacent to the frequency band of the structural mode of the aircraft, the frequency response of the structural mode obtained during open-loop testing can be masked, so that the aircraft can become unstable during closed-loop testing, which may prevent acquiring a closed-loop structural response. These issues could be filtered out if the frequencies of the structural modes were high enough, so that the landing gear and tire mode frequencies were separable, but these filters would still affect the phase to some extent, potentially affecting the results.

One way to address this problem is to use some kind of "soft support" system during structural mode testing, as shown in Fig. 23, to set landing gear and tire modes at frequencies well below the frequencies of interest that do not interfere with the test. Such a test environment can be particularly useful when the response of low-frequency modes due to rigid bodies or pilot couplings must be examined. The soft support system can use electric suspension [18], pneumatic support [37, 62], or rubber bungee cords [63]. As shown in Fig. 23a, the soft support system using pneumatic support supports the aircraft vertically by installing airbags that replace tires on each landing gear and using the air flotation system below the airbags to provide yaw or side force. It has been applied to X-35B [37, 62]. As shown in Fig. 23b, the soft support system using rubber bungee cords, a method of lifting the aircraft from the ground using rubber bungee cords, was applied to light combat aircraft (LCA) [63]).

5.5.4 Aircraft Status Monitoring and Special Equipment

During SCT, the status of several parameters must be monitored to prevent damage to the aircraft by sustained and excessive vibration [18].

First, the engine is an item that requires attention during testing, since the engine shaft shall be rotate periodically during testing using a crank system or other device to distribute effects of vibration wear on bearings and rotating parts.

Second, high vibration levels can be reached and maintained for several cycles in some parts of the aircraft, due to the excitation input of the control surfaces during SCT. Therefore, it is necessary to monitor the level of vibration by mounting accelerometer and strain gauge sensors at aircraft structure critical points. And a function that automatically

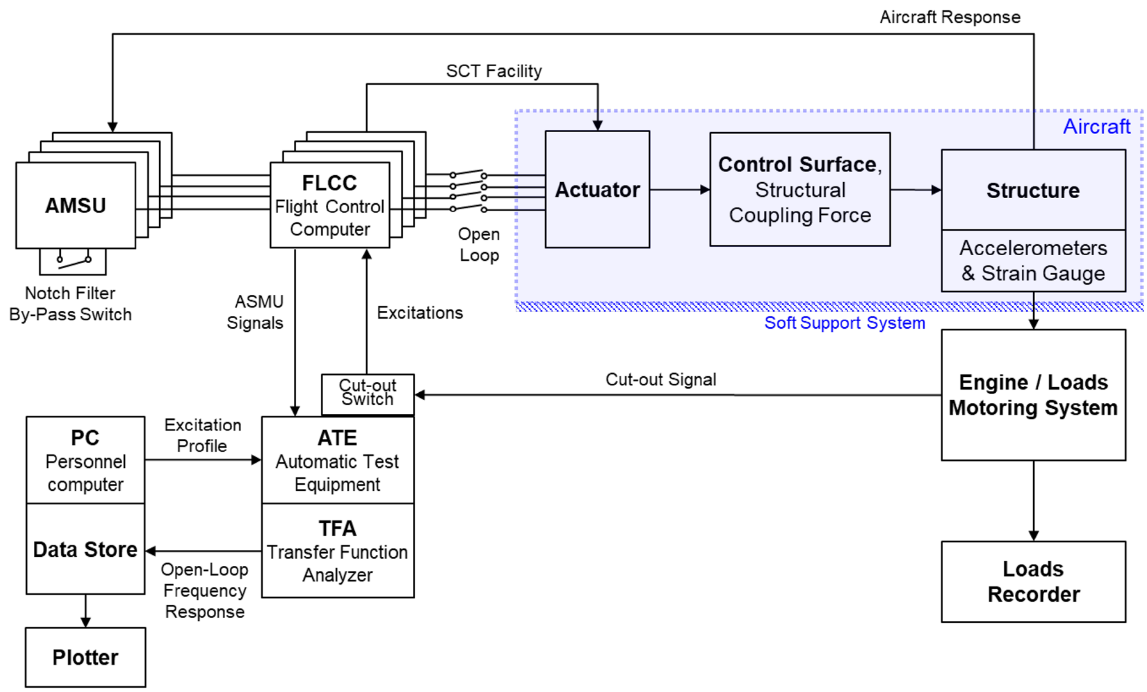


Fig. 22 Structural coupling test layout for EF-2000 [18]

(a) F-35 pneumatic soft support system

(b) LCA rubber bungee Cords

Fig. 23 Soft support system for ground testing [62, 63]

cuts out the excitation when the vibration exceeds a threshold value can be also applied to effectively prevent aircraft damage.

5.5.5 Excitation and Test Procedure

Due to factors such as model uncertainty, modeling-based structural coupling analysis cannot predict the responses of aircraft with sufficient accuracy. Therefore, SCT is essential in the aircraft development process to prove structural-coupling stability. SCT can be generally divided into open-loop testing and closed-loop testing according to its purpose [37].

Open-loop testing is a test performed with the loop being broken between the flight control system and the actuator. The purpose of this test is to obtain aircraft structural response from control surface excitation input to motion sensor output, or to observe the response to flight control system commands for actuators to verify the stability of the entire system including the flight control system. The test result is used to design a structural coupling filter to prevent structural coupling or is used to adjust the control gain.

The excitation signal is in the form of a frequency sine sweep or random signal, which is used in a method of measuring the gain between input and output. The variable frequency sine input has the advantage of obtaining a relatively accurate aircraft response in an arbitrary frequency range, but also has the disadvantage of increasing the time required for testing and accumulating aircraft fatigue due to high vibration levels. Meanwhile, the random input method has the advantage of being able to produce less wear on the actuators and a less violent system response in all frequencies and to reduce the test time, but coherence [64] should be referred to ensure the reliability of the test, since the accuracy is reduced.

If the excitation input is small, it may not drive the system due to non-linear hysteresis effects, whereas if the excitation input is too large, the ratio of the output to the excitation input may indicate a lower gain than it actually is due to non-linear input restriction. Therefore, the amplitude of the excitation input must be set high enough to obtain the level of force necessary for the response of the aircraft while avoiding the non-linear effects of the actuator and obeying the constraints of the load monitoring system (LMS). Typically, the amplitude of the excitation input is high at low frequencies and decreases with increasing frequency to obey the LMS constraints. And sine-wave inputs are preferably scaled in magnitude with frequency, and random inputs are sometimes good to be large enough to produce a non-linear actuator response, so the test engineer gradually increases the excitation input size until the maximum gain is observed. The excitation input of the actuator is an analog signal or a digital signal that is at least four times the sampling frequency of the flight control system in consideration of aliasing, resolution, and phase lag characteristics by the Nyquist frequency, while the accuracy of the test data can be less than four times the sampling frequency considering phase lag and resolution. For example, in the case of the X-35B, the actuator was stimulated with a digital system of 80 Hz, and the test data are reported to be accurate up to about 15 Hz, less than four times the sampling frequency [18].

Since the flight control system includes not only the motion sensor, but also the pilot's command to the control stick, the possibility that structural vibration is input to the control stick through the pilot's arm cannot be ruled out [37]. As mentioned earlier, the possibility of structural coupling caused by the pilot's stick input increases in aircrafts that adopt control surfaces with high inertia and side sticks with small displacement. The open-loop test applies to BDFT. This test is performed with the pilot seated in the aircraft and with his hand gripping the control stick.

Structural response can be obtained by separately exciting the symmetrical and asymmetrical structural modes of the aircraft, in a way which inputs the same signal to the symmetrically operated control surface and the same amplitude but 180 degree phase signal to the asymmetrically operated control surface. In addition, excitation must be input and measured with an extended frequency range up to the sampling rate that characterizes FLCC digital signals, taking into account the effect of high frequency signals folding back to low frequencies due to digital characteristics. For this example, the T-50 with FLCC's sampling rate of 64 Hz has considered excitation signals up to 50 Hz [65]. In addition, the SCT should be performed in single and dual failure states of redundant hydraulic systems and FLCCs where actuator degradation can affect the design of OLFRF and notch filter (NF), as well as normal states [18].

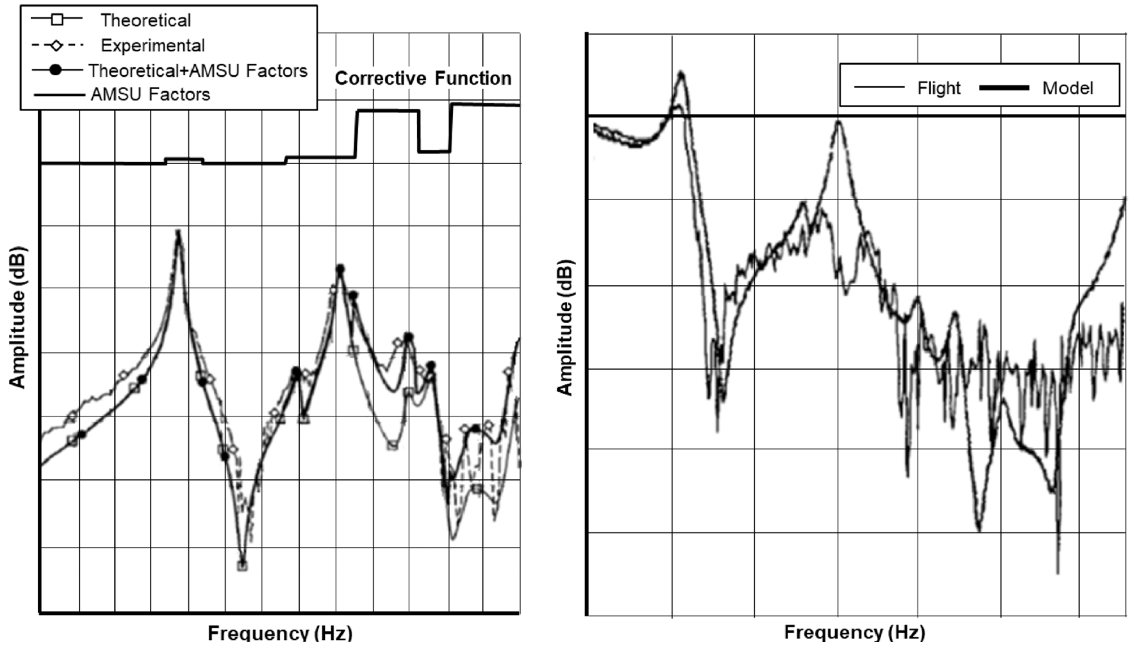
Closed-loop testing which measures power spectrum density (PSD) by amplification ratio is used to verify designs under highest control gain flight conditions. Variable gain is applied to a control law or sensor or actuator path and tested in gradual increments to prevent sudden instability that could damage the aircraft. At each increased gain, the test engineer excites the aircraft by inputting pulses to actuators or tapping the aircraft structure and control surfaces. A potential instability can be observed when the response barely attenuates as the gain increases. And if the variable gain is adjusted to two times the maximum control gain condition and the system is stable (gain margin greater than 6 dB), the test passes.

6 Model Update and Flight Clearance Process

Before the ground test, the structural behavior characteristics are predicted by the finite-element method. To ensure structural coupling stability, a notch filter is designed based on an aeroservoelastic model that integrates predicted structural models and other models of aerodynamic, control law, sensors, actuators, etc. However, among the components of the ASE model, the aircraft structural dynamic model contains a lot of uncertainties, in view of dynamic analysis. Therefore, it is essential to update the ASE model based on test results from ground vibration test (GVT) and SCT to increase confidence of the design and reliability of the analysis [18].

SCT is usually performed in parallel with GVT, but not necessarily in the same configurations. GVT is performed to acquire sufficient data related to aircraft modal identification including specific external stores. Meanwhile, since SCT collects structural response characteristics at the high-frequency band and is directly used in notch filter design, the representative configuration of the most critical ones identified based on the model should be applied.

To reduce errors in the model update process and generate reliable models, the motion sensors' location, fuselage model, matching of the main modal frequencies, and nonlinear effects should be considered with great care. In particular, if the effect of the nonlinearity of the system is large, the effector of nonlinearity on the test results should be carefully considered, because the comparison results with the model is affected by the error distribution that varies according to the frequency. Since the effect on the amplitude of the response is usually greater than the effect on the frequency of the mode, we consider the amplitude associated with the highest level of excitation and we change the original GVT modal damping values based on the SCT data. And for the



(a) Application of a correction functiona

(b) Comparison of model prediction with flight test data

Fig. 24 Example of model update and validation [18]

most critical modes, it is evaluated iteratively with varying excitation levels of control surface input.

Figure 24 shows an example of a model update case based on ground and flight test results [18]. Figure 24a is the result of model update through the OLFRF correction process using a frequency-dependent correction function based on the ground test data before the filter optimization phase. The structural uncertainty of the model can be reduced by matching the structural response at the frequency of the first modes and correcting the structural response above the first-mode frequency with a correction factor differently according to the frequency. Figure 24b shows the comparison between model predictions and flight test data. Additional correction can be implemented using flight test data after updating the ASE model with SCT data. This correction is much simpler in relation to the effectiveness of control surfaces, which are usually overestimated in the model, and gives an idea of the degree of conservatism of the model from the aerodynamic alleviation factors. Improvements in the structural and aerodynamic modeling techniques can reduce the cost and the risks inherent in the design and qualification of notch filters.

After confirming that the structural coupling stability criteria are satisfied through the SCT test and the analysis based on the updated ASE model, the first flight test and flight envelope expansion tests are gradually conducted. However, if the structural coupling stability criteria are not satisfied in some flight envelope, Aircraft Engine Operation Limitation (AEOL) can be set in consideration of the schedule of the aircraft development project, and flight tests can be continuously conducted under the monitoring of engineers in the control room.

7 Conclusion

In modern military fighter aircraft, new materials such as composites are applied to the aircraft structure design to improve structural efficiency by reducing weight, making the structure more flexible. At the same time, to preoccupy the battlefield with rapid maneuverability during air-to-air combats, the configuration is designed to make static stability unstable. Therefore, to make unstable aircraft stable and to ensure excellent handling qualities, there is a trend of designing a flight control system with a feedback concept and gradually increasing the authority of the flight control system. For this reason, structural coupling, the interaction among flexible structural dynamics, unsteady aerodynamics, and flight control systems, is an important design consideration issue in the development of modern fighter aircraft.

Considering this design issue, this paper presented major design considerations that developers must be aware of to properly design and effectively solve structural coupling problems that may occur during aircraft development. To this end, the physical relationship of structural coupling was organized, cases of structural coupling instability experienced during the development process of some aircraft were summarized, and airworthiness requirements such as stability margin and time-delay of the system presented in military specifications of flight control system were analyzed. Key technologies to be considered at the design stage in the aircraft development process, such as structural design to reduce structural coupling-induced instability, motion sensor location selection, and notch filter and phase advanced filter design, were presented. Finally, all considerations related to the test that must be kept in mind during the SCT test to secure flight test safety and obtain approval were described, and based on them, a method for updating the structural dynamics model was presented.

The results of this study will help aircraft developers and students majoring in aeronautical engineering understand the structural coupling characteristics of aircraft. During the aircraft development process, design and test tips that should be considered to effectively reduce and solve aircraft instability caused by structural coupling are presented. Therefore, by utilizing this in the development process, we expect to contribute to successful development by reducing trials and errors.

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Declarations

Conflict of Interest On behalf of all authors, the corresponding author states that there is no conflict of interest.

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