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Development of an Active Flutter Margin Augmentation System for a Commerical Transport

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A research program was conducted to determine the feasibility of the use of active control systems to achieve increased flutter margins on a commerical transport airplane. Three candidate systems were designed, developed, and installed on an L-1011, and the effectiveness of each system in increasing the damping of a lightly damped wing engine mode was evaluated in a flight test program. The existing data base, which included relevant in-flight structural response characteristics, provided an excellent point of departure for development of a flutter margin augmentation system. Measured in-flight damping increases, which range as high as 118% were accompanied by a substantial attenuation in pertinent structural response levels. Measured damping values, however, typically fell well below those predicted. Considerations relative to certification of flutter margin augmentation systems likely to be incorporated on the present generation commerical transports indicate that current regulatory procedures are satisfactory. Changes to certification requirements will be necessary, however, if far-term benefits are to be realized while maintaining levels of safety for flutter comparable to those of other disciplines such as gust loads.

Introduction

RECENT increases in fuel costs have stimulated the development of active control systems to aid in the development of fuel-efficient aircraft. These systems combined with configuration modifications can provide significant improvements in existing aircraft designs. One of several modifications currently being implemented on an existing wide-body jet transport is the increase in wing span to reduce induced drag. Normally such a change is accompanied by increased loads and possibly reduced flutter margins, necessitating strength and stiffness increases and/or ballasting of the wing. Such a process results in added weight and costly redesign which can significantly limit the benefits of the modifications. Active control systems can be employed to prevent load increases and to maintain original flutter margins without increasing wing stiffness, thereby allowing full advantage to be taken of the modification.

Extensive research, both independently funded and contract supported, has been conducted in the application of active control systems for load alleviation in the design of commercial transport airplanes.[1] In conjunction with these activities, an independently funded research and development program was conducted to evaluate the feasibility of the use of active control systems to achieve increased flutter margins on a commerical transport airplane. One of the principal objectives of this research was a determination of the extent to which existing aircraft systems could be adapted to the flutter margin augmentation task; a related objective was the evaluation of the adequacy of the existing data base for the design of such a system. The related contractual work is part of a NASA Aircraft Energy Efficiency (ACEE) program to investigate the use of maneuver load control (MLC), elastic mode suppression (EMS), and gust load alleviation (GLA) with increased wing aspect ratio applicable to today's commercial transport fleet. Using the related active controls investigations and the hardware developed for these programs as a base, three candidate flutter margin augmentation (FMA) systems were designed, developed, and evaluated in a flight test program.

The modal response selected as the subject of this investigation is one resulting from the interaction of primary wing-bending and wing-engine pitch. At zero airspeed, the frequency of the wing-bending vibration mode is well below that of engine pitch. As airspeed is increased, the wing-bending frequency increases, while the engine pitch frequency is relatively unaffected by aerodynamic forces. At some airspeed, then, the (uncoupled) wing bending frequency becomes equal to that of engine pitch, and continues to increase as airspeed is further increased. For light wing weights, this coalescence occurs at speeds within the design envelope of the airplane and results in a moderate decrease in damping of the engine pitch mode. This decrease in damping is significant, however, in that the engine pitch mode is relatively lightly damped at all airspeeds. As the airspeed is further increased, the damping of the engine pitch mode increases approximately to the precoalescence level, producing a hump mode. For heavy wing weights, the frequency coalescence occurs at speeds beyond the design envelope, at approximately 1.2 times the design speed (V_D). At these higher speeds, the damping characteristics of the engine pitch mode still exhibit the hump properties, but now the increase aerodynamic force levels are sufficient to drive the mode unstable. This modal system is, therefore, an excellent one for use in demonstrating active flutter margin augmentation, since the active system can produce both increased flutter speed margin (1.2 V_D) and increased modal damping within the design envelope. In addition, this latter characteristic is amenable to flight test verification since it occurs within the design envelope of the airplane.

Three distinct active controls configurations were designed and developed as candidates for the accomplishment of the FMA function just described. The three configurations used a total of two sensor locations and two sets of aerodynamic surfaces, so that both parameters were evaluated. Two of the systems were implemented through the ailerons and one through the horizontal stabilizer. One of the aileron systems was combined with the maneuver load control/elastic mode suppression (MLC/EMS) system which was implemented as part of the ACEE program. The aileron FMA system implemented in this manner provides additional symmetric

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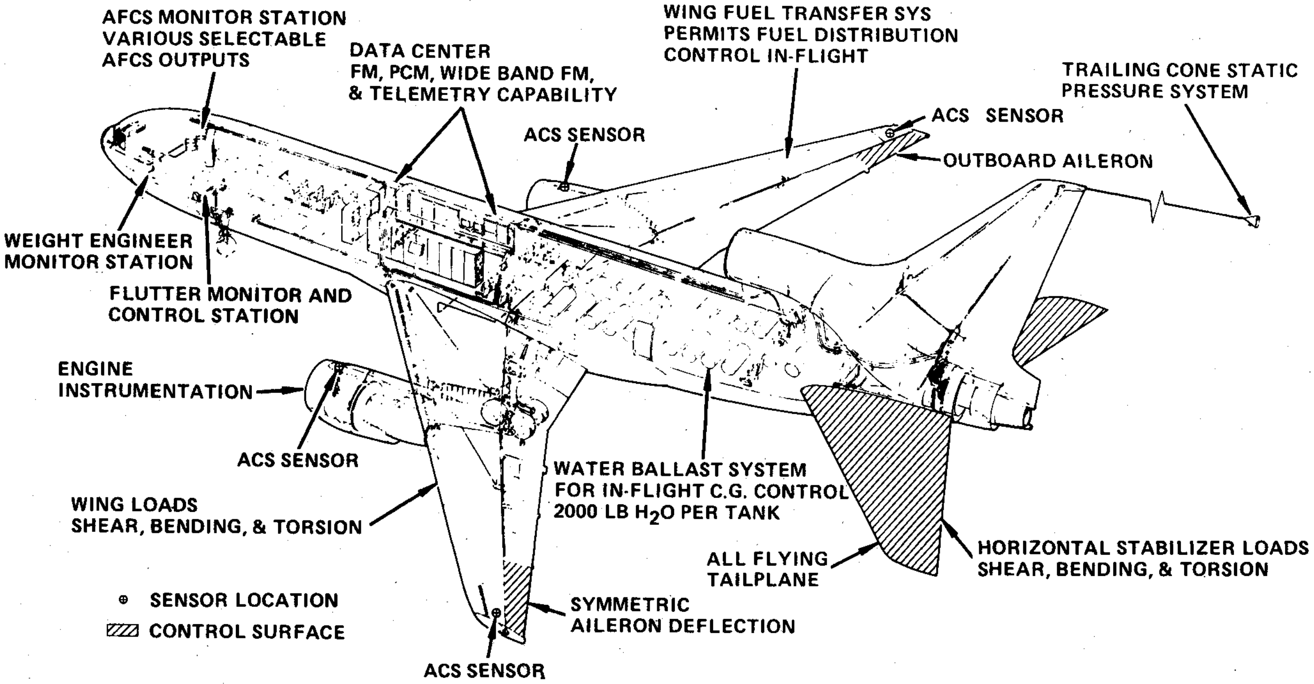

Fig. 1 Flutter margin augmentation system control surface and sensor location.

aileron motion as a function of wing tip acceleration. The remaining aileron and horizontal stabilizer systems sense wing-engine vertical and lateral accelerations. Figure 1 shows the candidate control surfaces and associated sensor arrangement on a current wide-body transport. Synthesis, design, fabrication, ground test, and flight test of the proposed systems as well as results and supporting analyses are presented in the following.

Aircraft Systems Description

The L-1011 and its flight control system characteristics were found to be well-suited for active controls application. All controls are irreversible, powered by multiple hydraulic systems. The outboard ailerons, which are most effective for wing-load alleviation, remain effective throughout the speed range without control reversal. The in-house test aircraft, serial number (S/N) 1001, already contained outboard aileron series servos which are used by the active control system to provide incremental motions to the power servo without interferring with the primary commands from the pilot or autopilot. The all-moving, hydraulically powered horizontal stabilizer has the rapid control power availability required for active controls applications. A series servo in the horizontal stabilizer channel was added to the in-house aircraft early in the program as part of the load alleviation studies. Previous testing associated with the application of MLC/EMS/GLA indicated that the horizontal stabilizer series servo rate limit was preventing the power servo from reaching its peak rate during oscillatory motions. Accordingly, an increase in the series servo rate limit from 2.0 in./s to 4.0 in./s was achieved by a valve change, which would double the flow, and an operating pressure increase from 1000 to 1500 psi. Accelerometers were added to each wing engine pylon in order to sense engine vertical and lateral accelerations. These are in addition to the accelerometers in the wing tip and fuselage which are also used for the MLC/EMS function.

Candidate FMA Systems Development

Available Data Base

The development of the candidate FMA systems is based, for the most part, on the breadboard active control system initially developed for the MLC, EMS, and GLA functions. The MLC and EMS functions require that the active system provide appropriately phased symmetric motions of the outboard ailerons in response to vertical accelerations sensed at the wing tips and fuselage. The EMS provides reduction in wing gust loads resulting from wing response at the first wing-bending mode frequency. The active system further provides appropriately phased horizontal stabilizer motions in response to sensed airplane pitch rate, providing GLA in the range of the airplane short period frequency.

As indicated earlier, the modal response selected for the application of FMA results from the interaction of symmetric first wing-bending and wing-engine pitch. Based on exploratory analyses and the in-flight response characteristics of these modes to sinusoidal excitation provided by the outboard ailerons and horizontal stabilizer, three candidate FMA systems were selected for investigation.

As part of the active controls program, in-flight motion transfer functions were measured for selected aircraft weight and flight conditions using both symmetric outboard aileron and horizontal stabilizer sinusoidal excitation. Typical frequency responses of such parameters as wing-tip normal acceleration, engine normal acceleration, and engine lateral acceleration, which are pertinent to the subject in-flight modal response, are presented in Figs. 2-7. The frequency range of interest for this investigation is from 1.0 to 3.0 Hz, which encompasses the first wing bending and wing-engine pitch mode zero airspeed frequencies of 1.3 to 1.8 Hz (depending on wing fuel) and 2.75 Hz, respectively. The measured in-flight responses indicate that both the outboard aileron and horizontal stabilizer provide sufficient force inputs to excite the modes of interest. The corresponding analyses for the conditions tested are also presented in Figs. 2-7. These analyses were performed using the complete airplane mathematical model used for flutter analysis. This model has the benefit of having been correlated to agree with ground vibration tests of the full-scale aircraft and was used, in conjunction with interactive computer graphics, in deriving and/or checking candidate control laws for the MLC/EMS and GLA fucntions and subsequently for the synthesis of the FMA function. In general, the results of the analyses indicate substantial agreement with the flight test measured frequency responses, both in magnitude and phase, thus providing a good point of departure for this investigation.

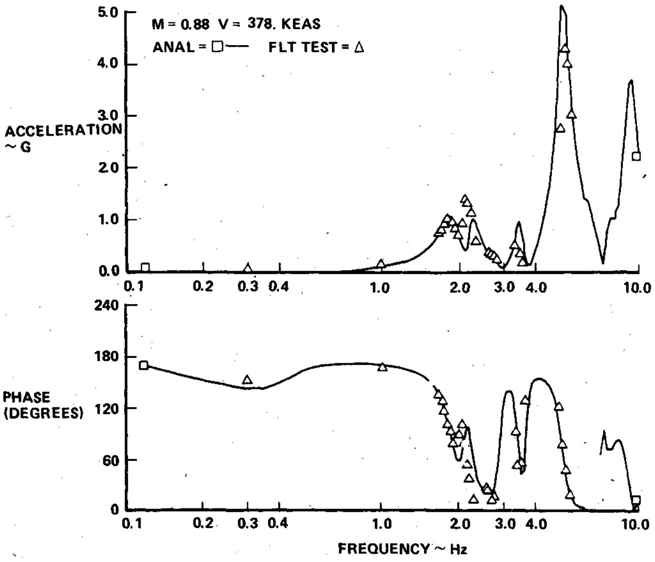

Fig. 2 Wing-tip normal acceleration/degree aileron, open loop.

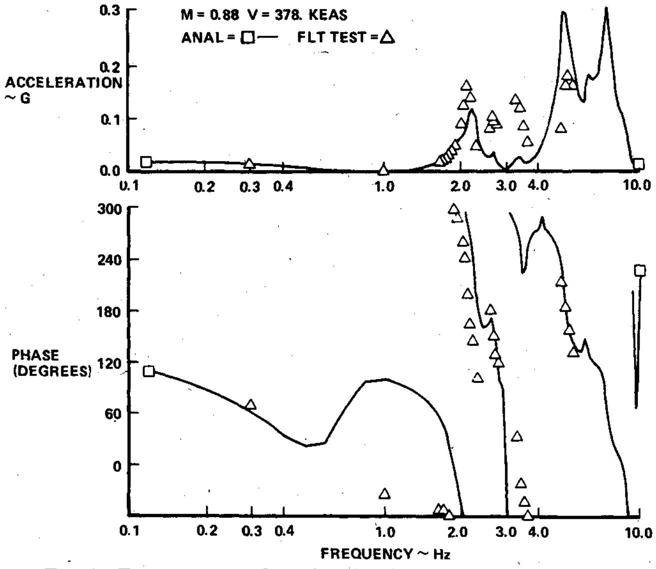

Fig. 4 Engine-1 normal acceleration/degree aileron, open loop.

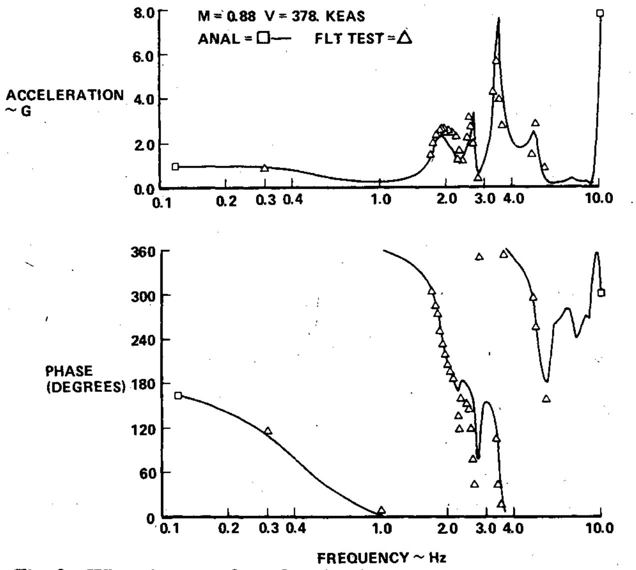

Fig. 3 Wing-tip normal acceleration/degree stabilizer, open loop.

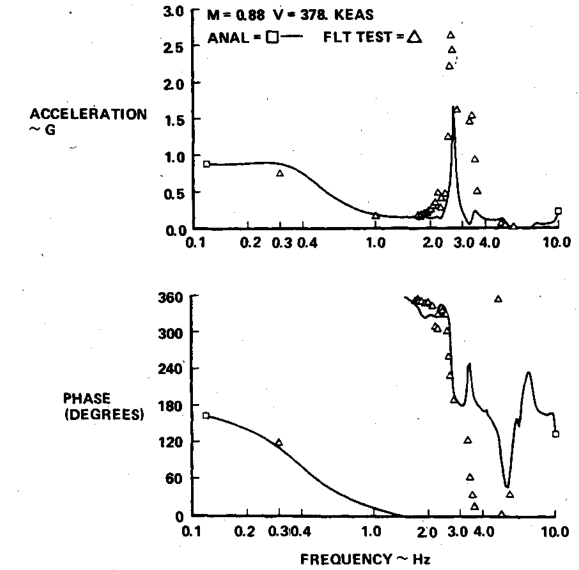

Fig. 5 Engine-1 normal acceleration/degree stabilizer, open loop.

Control Law Development

As indicated earlier, the in-flight coalescence of the first symmetric wing bending mode with the wing-engine pitch mode results in a lightly damped modal response. This coalescence occurs within the flight envelope at a speed which is well above the maximum operating speed (V_{MO}) of the aircraft and at a nonstandard, minimum wing fuel condition. The procedure for developing the control laws is initiated by establishing an objective to be met by the control law, in this case, an increase in the logarithmic decrement g of between 0.03 and 0.04 was specified. Additionally, the flutter-speed margins for all flutter modes may be increased but shall not be degraded below 1.2 V_D by the active control system. The three candidate FMA active controls loops selected for investigation are denoted for convenience as FMA-1, -2, -3 as follows: outboard ailerons sensing wing-tip vertical acceleration (FMA-1); outboard ailerons sensing wing-engine pylon vertical and/or lateral accelerations (FMA-2); and horizontal stabilizer sensing wing-engine pylon vertical and/or lateral accelerations (FMA-3).

Using in-house active controls synthesis programs, gain and phase constraints required to meet the damping improvement objective are computed for each of the candidate FMA loops, at the frequency and airspeed corresponding to minimum in-flight damping. In the case of the outboard aileron/wing tip sensor loops, the FMA gain/phase constraints need to be considered in conjunction with the constraints determined for the MLC/EMS functions and a compatible control law derived accordingly. This interaction may be in the form of having to gain schedule contributions from wing and body as a function of speed, or more precise shaping of the control law, where little frequency separation exists between each of the functions to be performed, invariably resulting in a compromise in effectivity where constraints conflict. In computing gain/phase constraints, the active control system series servos, power servos and sensor characteristics, in terms of transfer function poles, are represented in the basic math model. The control laws are arrived at by curve fitting the gain/phase constraints with polynomials, which are a function of the Laplace transform variable s keeping in mind system hardware capabilities and mechanization considerations. The resulting control laws are then reflected in the math model and flutter analyses performed to verify that the objectives have been met as well as to ensure that other modal characteristics have not been degraded.

The control laws were mechanized on a solid-state analog computer, composed of operational amplifier chips which are "wired" with input and feedback impedances to generate the

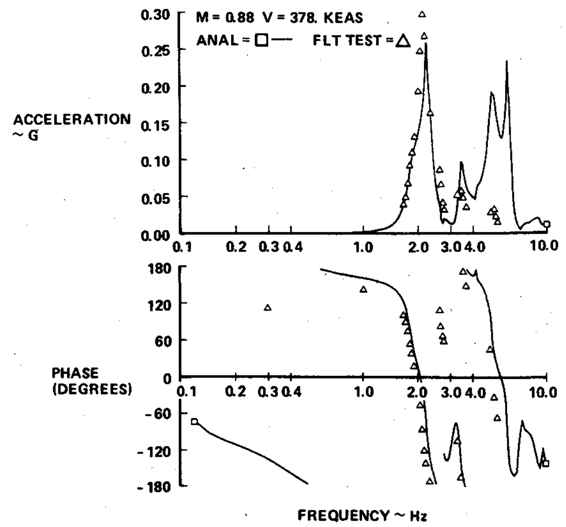

Fig. 6 Engine-1 lateral acceleration/degree aileron, open loop.

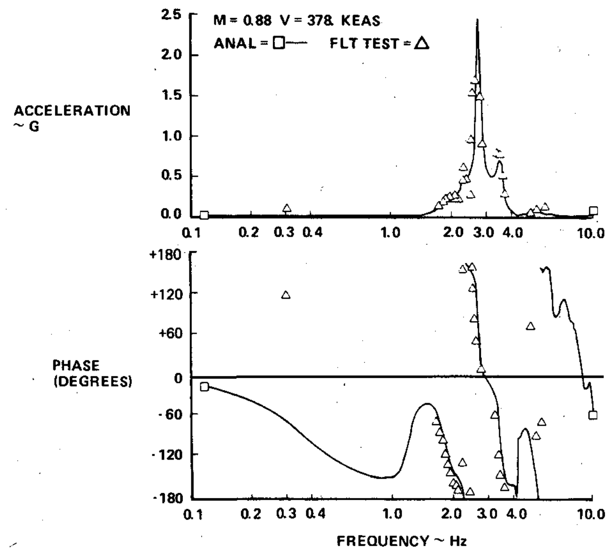

Fig. 7 Engine-1 lateral acceleration/degree stabilizer, open loop.

prescribed transfer function characteristics. A block diagram of the breadboard analog system, showing only those elements pertinent to the FMA function, is presented in Fig. 8. The derived control laws associated with each of the candidate systems are specified in the blocks shown under "Computer" in Fig. 8. As shown, the inputs to the computer are the sensed vertical accelerations of the fuselage (n_{z_B}) wing tip ($n_{z_{WT}}$), and wing engines ($\eta_{z_{ENG}}$), and the lateral acceleration of the wing engines ($\eta_{y_{ENG}}$). The output of the computer is a command signal to the outboard ailerons $\delta_{A_{CMD}}$ and horizontal stabilizer $\delta_{H_{CMD}}$. Sufficient flexibility was provided in the system design to activate each FMA loop independently as well as to vary each loop gain from half nominal to twice nominal. In addition, the wing-engine pylon vertical and lateral accelerations may be sensed independently or in combination.

Laboratory and Aircraft Ground Tests

The laboratory facilities for testing the active control system included the L-1011 Vehicle Systems Simulator (VSS) and the Visual Flight Simulator (VFS). The VSS is comprised of actual aircraft systems hardware enabling simulation of the complete aircraft control system. Elements of the primary flight control system such as pilot input mechanism, servos, and actuators are mounted to support structure which duplicates that of the airplane. Control surface inertial effects and support structure are also duplicated. Each of the active controls subsystems was functionally bench-tested before interfacing or installation into the VSS. The VFS consists of a pilot cab with operating L-1011 control systems and instruments permitting simulated VFR or IFR flight, interconnections with the VSS to operate the VSS control surfaces, and math models of the airplane and its flight

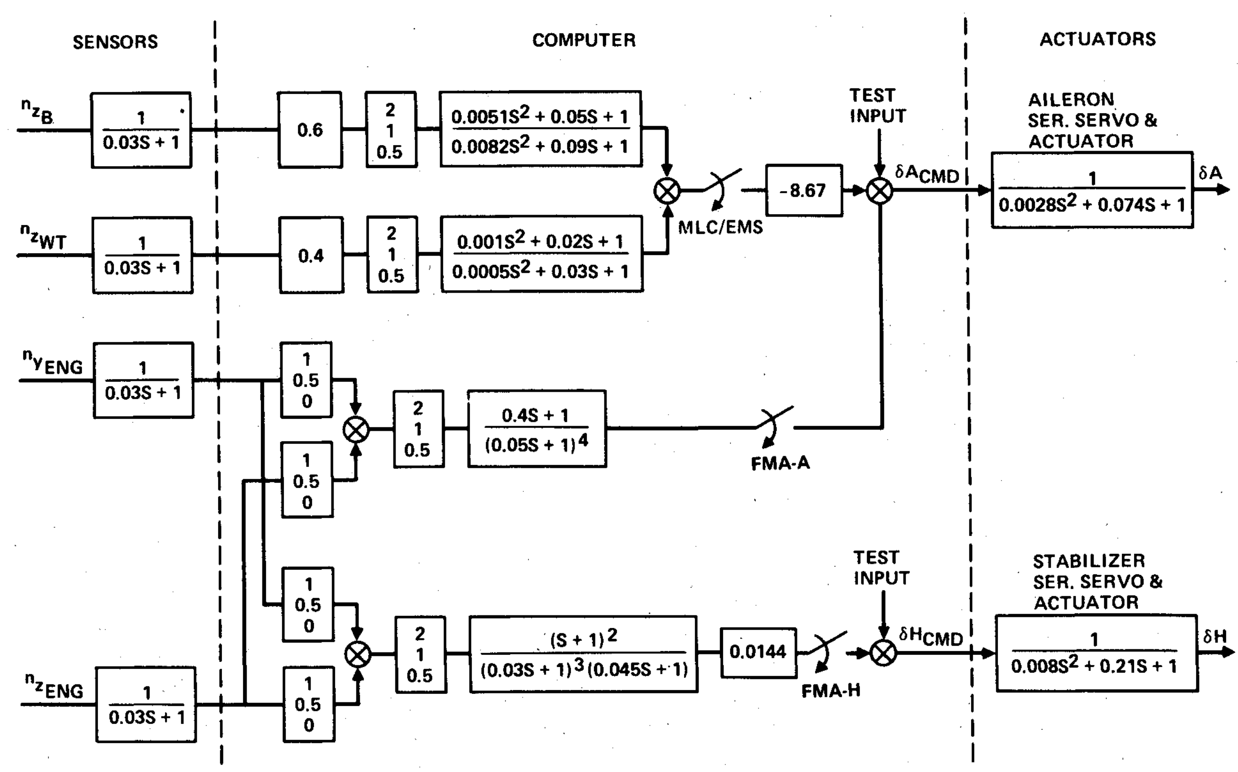

Fig. 8 FMA analog system block diagram.

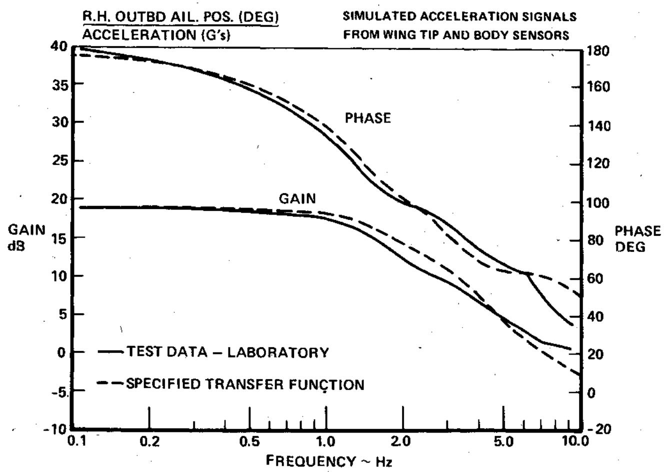

Fig. 9 Outboard aileron, wing-tip acceleration FMA-1, laboratory test vs specified transfer function.

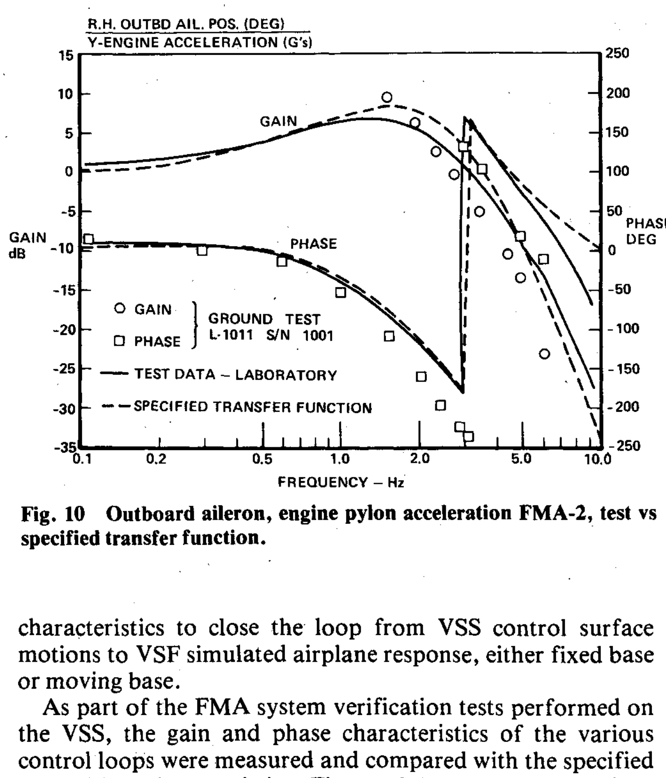

Fig. 10 Outboard aileron, engine pylon acceleration FMA-2, test vs specified transfer function.

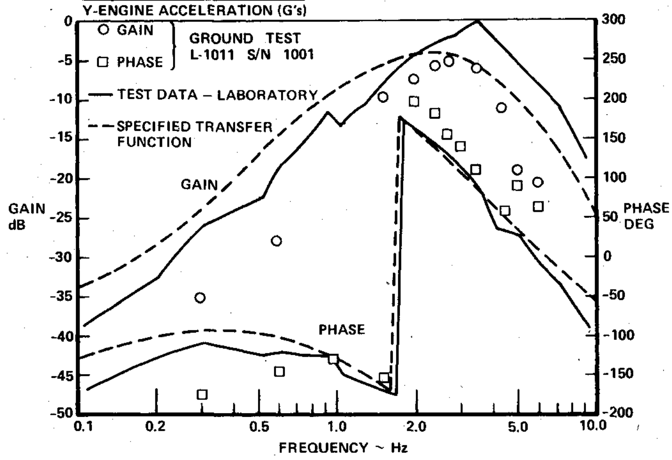

Fig. 11 Horizontal stabilizer, engine pylon acceleration FMA-3, test vs specified transfer function.

characteristics to close the loop from VSS control surface motions to VSF simulated airplane response, either fixed base or moving base.

As part of the FMA system verification tests performed on the VSS, the gain and phase characteristics of the various control loops were measured and compared with the specified control law characteristics. Figures 9-11 show a comparison of the test and specified transfer function characteristics (control surface position relative to accelerometer output signal) for the right-hand outboard aileron/wing tip, right-hand outboard aileron/wing engine pylon, and horizontal stabilizer/wing engine pylon FMA loops, respectively.

The coupled VSS/VFS provided a complete verification of the FMA system closed-loop characteristics and enabled evaluation of the system and aircraft behavior associated with various failure modes. The failure modes simulated included, but were not limited to, loss of sensor signal, loss of control surface feedback signal, loss of series servo command signal, and loss of system pressure supply. These failures resulted in very mild air responses such that only fixed-base simulation was required for these tests.

Prior to flight test, ground tests were performed on the active control system as installed in the in-house structural test aircraft, S/N 1001. In addition to verification of the functional and operational integrity of the system, open-loop, end-to-end frequency response tests were made for each of the FMA loops to verify the installed system transfer functions. Using simulated accelerometer signals as input to the FMA computer, the resulting gain and phase characteristics for the FMA-2 and FMA-3 functions are also presented in Figs. 10 and 11 for comparison with laboratory test results and specified characteristics. Figure 12 presents gain/phase comparisons between aircraft ground test and the specified FMA-1 function; however, unlike Fig. 9, the transfer function herein includes only that portion of the FMA-1 loop which senses wing-tip accelerations. The aircraft ground test results for FMA-1 and FMA-3 show good agreement with the specified functions in the 2.5 to 3 Hz range. The FMA-2 aircraft ground test results show more phase lag than desired in the frequency range of interest. Investigation into the cause of the greater phase lag revealed that the magnitude of the simulated accelerometer signal into the FMA computer was such as to result in aileron displacements of ± 0.29 deg. At these low amplitudes, which are somewhat below the levels expected in flight, the nonlinearities in the outboard aileron actuation loop (servos and linkages) result in the introduction of additional lag.

Flight Tests

Flutter flight tests to evaluate the effectiveness of the three active FMA loops were conducted for a nonstandard minimum wing fuel condition at an altitude of 22,000 ft. The selected airspeed points at which the damping characteristics for active system OFF and ON conditions were evaluated ranged from 290 to 400 knots equivalent airspeed (KEAS) corresponding to a maximum Mach number of 0.93. At the test-fuel condition, approximately 46,000 lb of fuel remained for test purposes allowing approximately 1.5 h of test time per flight. The test procedure used in evaluating each of the FMA functions consisted of discrete tuning of the in-flight modal response of interest using symmetric sinusoidal excitation provided by the outboard ailerons or horizontal stabilizer and performing "quickstops" for active system OFF and ON. For the active system ON condition, the quickstops were preceded by a longitudinal control pulse for an initial check on overall aircraft stability. Although the test aircraft is fully instrumented, only those structural response measurements pertinent to active control system tests and the in-flight modal response being investigated were activated and monitored on board the aircraft using strip-chart recorders and oscilloscope Lissajous displays. In addition to a preliminary on-board damping evaluation derived from the response decays, all measured data were recorded on magnetic tape for further postflight analysis.

Flight Test Results and Evaluation

The damping characteristics of the in-flight modal response involving the interaction of symmetric first wing-bending and

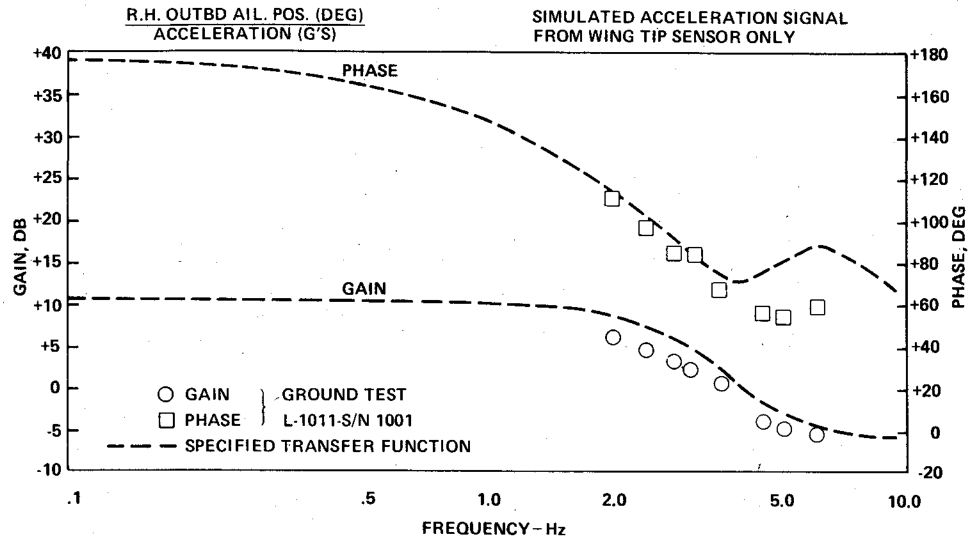

Fig. 12 Outboard aileron, wing tip acceleration FMA-1, aircraft ground test vs specified transfer function.

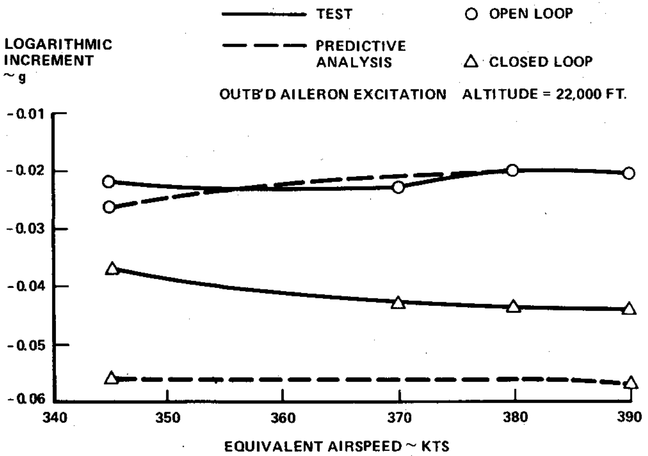

Fig. 13 Comparison of test and predictive analysis, outboard aileron, wing tip n_z, FMA-1.

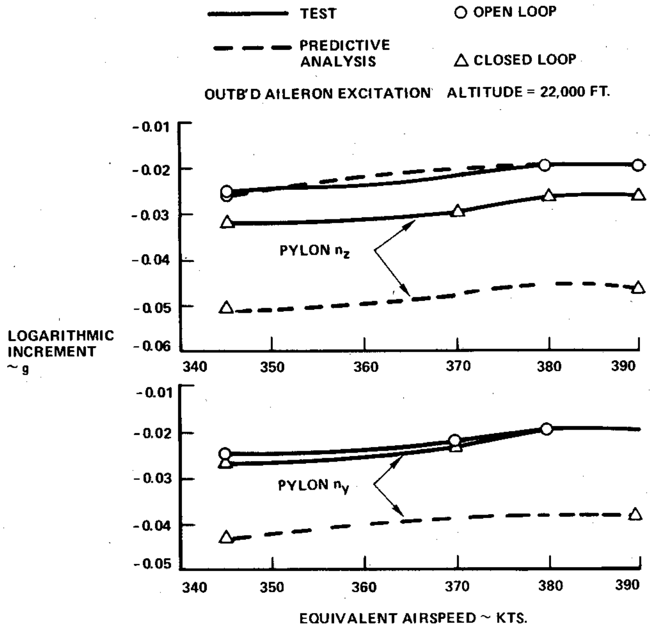

Fig. 14 Comparison of test and predictive analysis, outboard aileron, wing-engine pylon, FMA-2.

wing-engine pitch are best reflected by the wing-engine center of gravity vertical acceleration response. The test damping results, therefore, are derived from the response decays recorded by the engine vertical accelerometers and are presented in Figs. 13-15 for each of the active FMA loops investigated. In addition to the open- and closed-loop test results, corresponding predictive analyses are also presented for comparison. Experience in the prediction of damping characteristics of "hump" modes has shown this type of mode to be extremely sensitive to the usual flutter analysis parameters and, as would be expected, to the assumed levels of structural damping in the engine modes. In order to provide a more direct comparison between test and analysis of the changes in damping associated with the various FMA functions, the assumed structural damping has been adjusted such that the predicted open-loop minimum damping levels coincide with the measured in-flight open-loop minimum damping. The airspeed at which minimum damping occurs for the baseline open-loop condition is approximately 380 knots equivalent. As indicated in Figs. 13 and 15, the damping characteristics for the open-loop condition vary slightly depending on whether the source of excitation is the outboard aileron or the horizontal stabilizer. In general, the increment in damping predicted by analysis is higher than that measured, with the greatest percentage improvement being predicted for the stabilizer/pylon n_z loop followed by the outboard aileron/wing-tip n_z loop. Test results, however, attribute the greatest percentage improvement to the aileron/wing-tip n_z loop (Fig. 13). Table 1 summarizes the percentage increase in damping predicted by analysis and achieved by flight test for all the FMA loops investigated.

Typical in-flight response decays with FMA system OFF and ON are presented in Figs. 16 and 17 for the more effective loops. Presented in each figure are the input oscillator signal, left wing-tip vertical acceleration, and left engine vertical acceleration responses, system OFF and ON, at 380 KEAS and 22,000 ft altitude. It is evident from the closed-loop responses that substantial response attenuation accompanies the increase in damping. The attenuation is greatest for the wing-tip response associated with outboard aileron/wing-tip loop.

Configuration Selection

Whenever several alternate active control system configurations are available to perform a particular function, the selection process involves the consideration of a number of factors. Some of these considerations will be illustrated in terms of the three candidate FMA systems previously discussed. The selection tradeoffs include system complexity, analytical comparisons, flight-test results, reliability, system stability and system interface requirements.

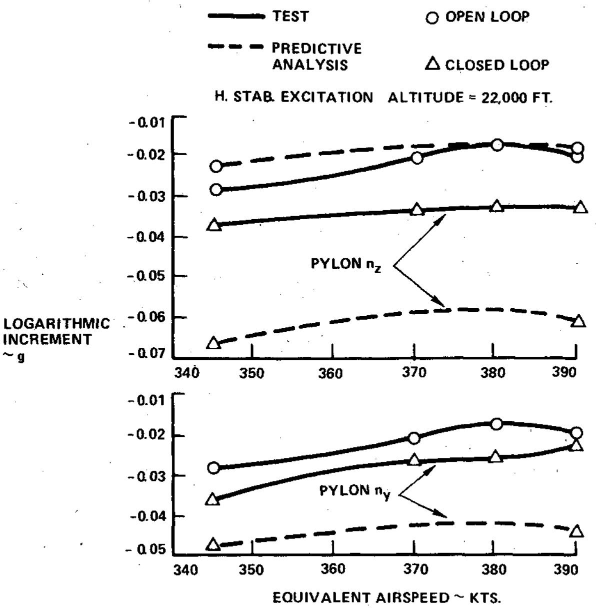

Fig. 15 Comparison of test and predictive analysis, horizontal stabilizer, wing-engine pylon, FMA-3.

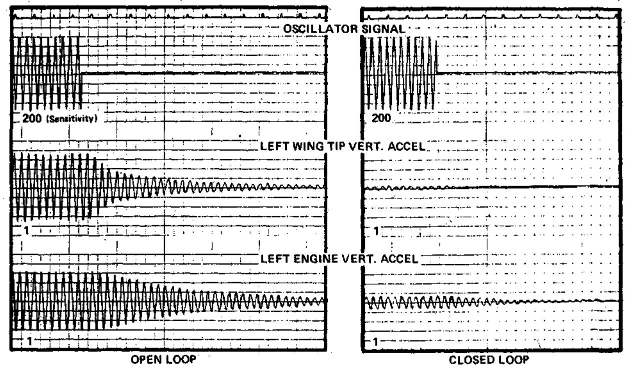

Fig. 16 Typical inflight modal response decays, outboard aileron, wing-tip vertical acceleration FMA loop.

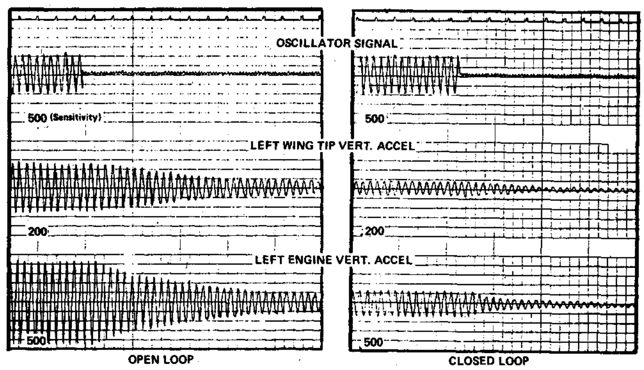

Fig. 17 Typical in-flight modal response decays, horizontal stabilizer, engine pylon vertical acceleration FMA loop.

System Complexity

In terms of system complexity of the FMA system considered independently of other active systems, the system which produces aileron motions as a function of engine pylon accelerations (FMA-2) is the least complex, having the simplest control law and a single set of sensors. Next in complexity is FMA-3, which produces horizontal stabilizer motions as a function of pylon accelerations: this system has a slightly more complex control law. The FMA-1 system, producing aileron motions as a function of wing tip accelerations and body accelerations, with separate control laws for each sensor, is the most complex of the three. It should be noted, however, that the relative complexities of these systems might be quite different if evaluated in conjunction with other active controls functions (MLC/EMS/GLA).

Analytical and Test Results

On the basis of analytical predictions, FMA-3 is indicated to be most effective both in increasing the flutter margin above 1.2 V_D and in increasing the damping of the hump mode. The analysis indicates that FMA-1 is next most effective and that FMA-2 is least effective in both characteristics. The flight-test results, however, show FMA-1 is most effective in increasing the damping of the hump mode; the test results for this system correspond most closely to the analytical prediction. If a selection were to be made based on the data available from the present test program, FMA-1 would be selected. Before committing this system to production, however, it would be desirable to determine why the other two systems produced results which were so deficient relative to the predicted results. One area, which is recognized as having an important effect on the overall damping correlation, is the nonlinear behavior of the servo/actuator package as a function of amplitude. Additional ground test data would be required to define these characteristics. Another area most likely contributing to the poor correlation, but which is somewhat more difficult to adjust for analytically, is associated with predicting open-loop, in-flight structural responses (Figs. 2-7) which are a function of the accuracy of the analytical structural model and unsteady aerodynamic representations. It should be realized, however, that prior to committing any system to production, considerably more analysis and flight testing would be necessary, possibly resulting in several control law iterations in order to achieve the desired level of system effectiveness.

Table 1 Comparison of predicted and measured percentage increase in damping with FMA system

FMA function	Predicted	Measured
Outboard aileron, wing tip n_z, FMA-1	180	118
Outboard aileron, engine pylon n_z, FMA-2	130	35
Outboard aileron-engine pylon n_y, FMA-2	95	0
Horizontal stabilizer, engine pylon n_z, FMA-3	248	94
Horizontal stabilizer, engine pylon n_y, FMA-3	152	52

Reliability

The reliability aspects of the three FMA systems are very similar with the possible exception of the sensor reliabilities resulting from the sensor environment. The sensors of FMA-2 and FMA-3 are subjected to a much more severe environment than are the FMA-1 sensors. It is considered that this factor may be quite significant in the selection of a system for a production program. As a matter of fact, these considerations strongly influenced the selection of the engine pylon as the location of the sensors for the present program. From a functional standpoint, it is clear that the optimum sensor location is on the engine or inlet, where the modal response of interest is much greater than the response of the pylon. It was felt, however, that location of the sensor on the engine might result in unacceptable hazards to the accomplishment of the test program.

System Stability

Any active controls system design for commerical transport service must have adequate stability margins to tolerate the gain and phase variations of the system and its components. Although this subject was only superficially considered in the present research program, some conclusions can be reached by examining the available data. The open-loop engine response due to stabilizer drive shown in Fig. 5 indicates that an FMA system using engine normal accelerations to produce horizontal stabilizer motions could be designed with adequate stability margins with few compromises. The modal response pattern is not dense, and the mode to be controled ($\approx$2.7 Hz) has much greater response than does the only other mode with significant response. As a consequence of this, the control law can be designed for optimum control of the 2.7 Hz mode, with only minor consideration of other modal responses. In contrast to this, Fig. 2 indicates that the design of FMA-1 for proper stability margins could be difficult. On the scale used in the figure, the 2.7 Hz response is obscured by the dense pattern of relatively high modal responses, so that the design of the control law must be strongly influenced by these responses. An example of a configuration intermediate to these two is shown in Fig. 4 for a system using aileron motions as a function of engine accelerations. The modal response pattern here is relatively dense, but the response of the 2.7 Hz mode is the same order of magnitude as the response of the adjacent modes.

System Interface Requirements

The identification of system interface requirements was not a specific part of the present research program since such requirements are highly configuration-dependent. Some basic considerations, however, are demonstrated in the control laws of the three FMA systems.

The FMA systems were developed in conjunction with the load alleviation studies described in Ref. 1, so that FMA-1 was designed to be used with an MLC/EMS system using the same sensors and control surfaces. In order to be compatible with these systems, the constraints applicable to the MLC and EMS functions were combined with those of the FMA function, so that the resulting function, with appropriate speed scheduling, might properly be termed an MLC/EMS/FMA function.

An alternate approach was used on the FMA-3 system. Since this system produces horizontal stabilizer deflection as a function of engine pylon accelerations, the interface of this system and the autopilot at low frequencies must be considered. Since it clearly would be undesirable to modify an existing autopilot to accommodate the FMA system, this latter system was designed to eliminate any significant interference with the autopilot. Accordingly, the control law for this function results in a very low gain at low frequencies, has a maximum gain at the hump-mode frequency, and rolls off fairly rapidly at high frequencies (Fig. 11). A similar consideration influenced the design of the control law for FMA-2, but since the effect of the outboard ailerons on airplane pitch is small, much less shaping was required.

Considerations Relative to Certification of FMA Systems

Present Systems

The process of certification of FMA systems such as described in this paper can, in general, be carried out within the framework of the present regulatory procedures. For certification, paragraph 25.629 of Ref. 2 requires that: 1) the unfailed airplane be designed to be flutter-free to 1.2 times the design speed/Mach number (V_D/M_D) and demonstrated by flight test to be flutter-free to V_D/M_D; and 2) in the presence of any single structural failure, combined with any other reasonably probable single failure, to be designed to be flutter-free to V_D/M_D and demonstrated to be flutter-free to V_{FC}, a speed midway between the maximum operating speed (V_{MO}) and V_D. For active control systems designed for nonflight critical functions such as the FMA system described here, the probability of system failure generally would fall within the probable range (greater than 10^{-5} per flight hour as defined in Ref. 3). The criteria for certification can be met with a failed FMA system, however, since the airplane is flutter-free within the design envelope with the system inoperative.

Future Systems: Near Term

For active controls applications in the future, present regulatory procedures will severely restrict the benefits to be gained through the use of flutter-suppression systems. It is not inconceivable, for example, that the development of active systems can be carried to the point that the in-flight failure of such systems can be shown to be extremely improbable (less than 10^{-9} per flight hour). The present flutter criteria, however, would limit the application of such systems to flutter margin augmentation functions rather than flutter-suppression functions. It is clear that, for future active controls applications, a revised approach is needed.

One such approach might be the revision of the present deterministic flutter criteria such as to increase the range of possible applications of active systems for flutter control. As an example of this, a possible relaxation of the criteria might take the form of requiring that the airplane, with the active system inoperative, be designed to be flutter free to V_{FC} and be demonstrated by flight test to be flutter free to V_{MO}/M_{MO}. The application of such criteria might then be contingent on substantiating that the maximum probability of in-flight failure of the active system meets a specific requirement.

Such a procedure would allow applications of active systems providing flutter margin augmentation at speeds up to V_{FC}, and flutter suppression at speeds above V_{FC}. It is clear, however, that any such procedure will eventually impose the same limitations on the development of active systems as do the present regulatory procedures.

Future Systems: Far Term

Given the continued development of active systems toward failure probabilities of 10^{-9} per flight hour or less, a point in time will be reached when deterministic criteria will no longer permit the effective use of such systems. Flutter criteria must then be developed which recognize the combined probabilities of hazard to continued safe flight and landing, and which prescribe acceptable levels of such hazard. Although this represents a radical departure from present procedures in the area of flutter, an equivalent approach has long been accepted in designing for gust loads.[4-6] For that matter, certain probabilistic considerations can be inferred from the wording of the present deterministic criteria. The consideration of a single structural failure implies that such failure is sufficiently improbable (3×10^{-5} per flight hour) that the combined probability of two such failures would be less than 10^{-9} per flight hour. Similar reasoning would place the probability of "any other reasonably probable single failure" in the range of 10^{-2} to 10^{-5} per flight hour. What must be recognized, in addition to the probabilities of structural and system failure, is the probability of encountering a hazardous flutter condition. It is the simultaneous occurrence, then, of a hazardous flutter condition and failure combination (structural and/or system failure), which must be shown to be extremely improbable. This may be expressed as follows:

> In the presence of system and/or structural failures, flutter conditions which present a hazard to continued safe flight and landing should be sufficiently remote that the simultaneous occurrence of such failure and hazardous flutter condition is extremely improbable.

A criticism of the above criterion that might be anticipated is that the substantiation of the probabilities of encountering a hazardous flutter condition may be a difficult task, with the result that such probabilities may be subject to unacceptably large errors. To prevent this, minimum probabilities of encountering any given condition might be established for each speed regime. Within the V_{MO}/M_{MO} envelope, for instance, the minimum probability might be 1.0, effectively retaining the present deterministic criteria for that regime. For speeds between V_{MO} and V_{FC}, the minimum probability might fall in the range of 10^{-3} to 10^{-5} per flight hour and for speeds between V_{FC} and V_D, minimum probabilities of 10^{-5} to 10^{-7} might be selected. Selection of such minimum probabilities would make the adoption of the criterion practicable both for the airplane manufacturer and the regulatory agency.

Conclusions

The research program described herein demonstrated the feasibility of using active flutter margin augmentation systems on commercial transport airplanes. The flight-test results indicate that the data base resulting from the design of a modern wide-body transport is adequate for the design of such systems, and that existing control system components are reasonably well suited to such applications. Measured in-flight damping increases ranged as high as 118% but, typically, fell somewhat below predicted levels.

The process of selection of an active control system from among several candidates is shown to be the familiar one of evaluating the various tradeoffs and identifying the system most nearly fulfilling the requirements.

It is shown that flutter margin augmentation systems likely to be incorporated on the present generation of commerical transports may be certificated within the constraints of current regulatory procedures, but that existing procedures will impede the development of future active controls systems. Proposals are presented for the certification of near- and far-term systems that would result in levels of safety for flutter comparable to those of other disciplines such as gust loads.

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