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# RECENT ACTIVITIES IN AEROSERVOELASTICITY AT THE NASA LANGLEY RESEARCH CENTER

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# RECENT ACTIVITIES WITHIN THE AEROSERVOELASTICITY BRANCH AT THE NASA LANGLEY RESEARCH CENTER

by

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## ABSTRACT

The objective of research in aeroservoelasticity at the NASA Langley Research Center is to enhance the modeling, analysis, and multidisciplinary design methodologies for obtaining multifunction digital control systems for application to flexible flight vehicles. This paper discusses recent accomplishments and presents a status report on some current activities within the Aeroservoelasticity Branch. In the area of modeling, improvements to the Minimum-State Method of approximating unsteady aerodynamics are shown to provide precise, low-order aeroservoelastic models for design and simulation activities. Analytical methods based on Matched Filter Theory and Random Process Theory to provide efficient and direct predictions of the critical gust profile and the time-correlated gust loads for linear structural design considerations are also discussed. Two research projects leading towards improved design methodology are summarized. The first program is developing an integrated structure/control design capability based on hierarchical problem decomposition, multilevel optimization and analytical sensitivities. The second program provides procedures for obtaining low-order, robust digital control laws for aeroelastic applications. In terms of methodology validation and application the current activities associated with the Active Flexible Wing project are reviewed.

## INTRODUCTION

Aeroservoelasticity (ASE) is a multidisciplinary technology dealing with the interactions of an aircraft's control system and its flexible structure. Accurate representations of the flexible structure, the steady and unsteady aerodynamic forces acting on the structure, and the flight control system are required to provide predictions of ASE interactions and to design active control systems for flexible vehicle application. There has been much progress by many researchers to numerous to reference in the last few years that demonstrated the usefulness of active controls technology for favorably modifying the aeroelastic response characteristics of flight vehicles. These demonstrations promise significant enhancements in aircraft performance and stability while reducing structural weight. It is apparent that the future will demand high gain control systems and flexible structures, two ingredients requiring significant interdisciplinary communication not only to avoid adverse ASE interactions but also to make maximum use of promising technology.

To prepare for the future, the Aeroservoelasticity Branch (ASEB) of the NASA Langley Research Center (LaRC) has been performing research which addresses four main objectives (Figure 1). These include activities 1) to improve aeroservoelastic modeling and analysis procedures, 2) to develop methodologies for integrating structural and control design functions, 3) to validate new software developments through comparisons with experiment, and 4) to apply ASE methods on advanced NASA and DOD flight projects.

Some of the projects associated with the first three branch objectives are described in this paper. These activities include: the use of Minimum-State[1,2] approximations of unsteady aerodynamics for obtaining low-order ASE models; an evaluation of the Statistical Discrete Gust Method[3-5] and the use of Matched Filter Theory[6,7] and Random Process Theory for predicting time-correlated gust loads; the development of a multilevel, decomposition methodology[8] based on parameter sensitivity[9] for obtaining an integrated structure/control law design capability; and the development of a digital robust active control law synthesis procedure[10,11] using constrained optimization. Finally, a status report on the Active Flexible Wing (AFW) test program[12,13] is provided. To place the AFW project in proper perspective, an overview of the entire program is provided although some of the tasks are not yet completed.

## MODELING AND ANALYSIS

### Minimum-State Approximations of Unsteady Aerodynamics

The classical equations of motion of a flexible aircraft contain unsteady generalized aerodynamic forces that are based on the assumption that the vehicle is undergoing simple harmonic motion. These forces generally take the form of matrices that are Mach-number and reduced-frequency dependent. The availability of efficient linear system algorithms for aeroservoelastic analysis and design has provided a strong motivation to approximate the unsteady aerodynamic forces as rational functions of the Laplace variable. Such rational function approximations (RFA) allow the aeroservoelastic equations of motion to be recast into a linear time invariant state-space form. A disadvantage of using an RFA is that it can significantly increase the size of the state vector. This increase in size is referred to as the aerodynamic dimension. There is, of course, always a trade-off between how well the rational function approximates the aerodynamic forces and the desire to keep the aerodynamic dimension small.

Currently, there are three basic formulations used within the ASEB in approximating unsteady generalized aerodynamic forces using rational functions[1]. These formulations include the Least-Squares (LS), the Modified Matrix-Pade´ (MMP), and the Minimum-State (MS) Methods. Table 1 shows the general form of the approximation and the aerodynamic dimension associated with each method. For the MS Method, the number of augmenting states required to represent the unsteady aerodynamics is a function only of the number of denominator roots in the rational approximation. Recent extensions to these approaches include the capability to enforce selected equality constraints on the RFAs and to optimize the denominator coefficients of the rational functions using nonlinear programming techniques.

Recent studies[2] have shown that by using discretion in the selection of the denominator (lag) coefficients, choosing various equality constraints, and applying physical weighting to the various aerodynamic data terms according to their importance in subsequent analyses, the MS Method can provide very accurate, low-order aeroservoelastic state-space models. The physical weighting procedure produces a measure of importance which allows the aerodynamic approximation to be improved at some reduced frequencies (at the possible expense of others) based upon physical properties without actually enforcing equality constraints at the specified points. The measure of importance is based upon partial derivatives of selected open-loop parameters with respect to the weighted term at a specified design flight condition. For the vibration modes, the weight at each value of reduced frequency is determined by the effective influence on the system flutter determinant; for control modes, by the effect on system gains; for gust modes, by the effect on the response to continuous gusts; and for hinge moment terms, by the hinge moment response to control surface or gust excitations.

Both the MS and the LS Methods were used to develop AFW aeroservoelastic models. Figure 2 shows a root locus plot that compares aeroelastic calculations using the two different models.

As can be observed the differences are quite small. For the MS model, only one-tenth the number of aerodynamic states used by the LS model, an order-of-magnitude reduction, were required. Results using the classical p-k flutter solution technique have been omitted from the figure to simplify the comparisons. Both the MS and the LS models provided accurate solutions as compared to this classical solution.

These various lag-selection, constraint, and weighting techniques provide an effective, systematic approach to generating aerodynamic approximations. Using these techniques, the total size of a typical time-domain aeroservoelastic model can be efficiently reduced by fifty percent. In addition to significant computer time savings for control design and analysis tasks, lower-size models provide more realizable optimal control laws and facilitate near real-time simulations.

## Time-Correlated Gust Loads

One of the major research activities within the ASEB is time-correlated gust loads methodology development. This activity began with a request from the U.S. Federal Aviation Administration for the NASA to investigate a claimed "overlap" between the Statistical Discrete Gust (SDG) Method[3,4] and the Power Spectral Density (PSD) Method[14] for computing gust loads. This investigation[5] led to the development of new time-correlated gust loads analysis methods that use Matched Filter Theory (MFT) and Random Process Theory (RPT)[6,7]. This section of the paper will discuss the investigation of the SDG-PSD overlap and the development of the new methods.

### Statistical Discrete Gust Method

The objective of the SDG Method is to determine analytically the maximum, or worst-case, responses of an airplane to discrete gusts representative of atmospheric turbulence. The method is based on the assumption that atmospheric turbulence is comprised of a family of discrete equi-probable smoothly varying ramp-hold gusts whose maximum magnitudes ($\bar{w}_g$) vary as indicated by the dashed envelope in Figure 3. A representative selection of gusts within this family are indicated by the solid curves.

The SDG Method is carried out in the time domain through the calculation of response time histories. In the general implementation of the SDG Method, an airplane is subjected to all possible combinations of single gusts within the family of gusts, including all possible combinations of spacing between the gusts. But, for an airplane modeled as a linear system, this extremely large number of inputs may be reduced to a manageable number by taking advantage of superposition. The overall worst-case response, $\bar{\gamma}$, is determined by first identifying, in descending order, the largest response peaks in the response time histories; second, grouping them as the largest single peak, the largest two peaks in combination, the largest three peaks in combination, and so forth; and finally, applying probability factors to the combinations of peaks.

### Power Spectral Density Method

The fundamental quantity of the PSD Method is the power spectral density function, or power spectrum. A power spectrum contains statistical information describing a random process, including the root-mean-square (RMS) value. The random processes in question in the present application are atmospheric turbulence and airplane responses. The input is assumed Gaussian, and because the system is assumed linear, the output is also Gaussian.

The input and output power spectral density functions are related to each other through the square of the modulus of the airplane frequency response function, as given by the following equation

$$\Phi_y(\omega) = \Phi_{w_g}(\omega)\,\left|H_y(i\omega)\right|^2$$

where $\Phi_y(\omega)$ is the airplane response power spectrum, $\Phi_{w_g}(\omega)$ is the atmospheric turbulence power spectrum, and $H_y(i\omega)$ is the airplane frequency response function.

The quantity $\bar{A}$ is the normalized response quantity, defined as the ratio of the RMS value of the output to the RMS of the input.

$$\bar{A} = \frac{\sigma_y}{\sigma_{w_g}} = \frac{\left[\int_0^\infty \Phi_y(\omega)d\omega\right]^{1/2}}{\left[\int_0^\infty \Phi_{w_g}(\omega)d\omega\right]^{1/2}}$$

### SDG-PSD Overlap

Jones[3,4], the developer of the SDG method, claims that, under certain circumstances, the SDG and PSD Methods produce essentially the same numerical results. He refers to this situation as the "SDG-PSD Overlap." The quantitative definition of the overlap is given by the equation

$$\bar{\gamma} = 10.4\,\bar{A}$$

where $\bar{\gamma}$ is the SDG worst-case response as defined previously, and $\bar{A}$ is defined above.

The approach taken in the NASA investigation[5] of the SDG-PSD overlap was to perform SDG and PSD analyses for several airplanes at different flight conditions and to compare the corresponding responses from each method to see if the "10.4 factor" was obtained. All the analyses were for symmetric longitudinal conditions with the vertical component of atmospheric turbulence as the disturbance quantity. To maintain impartiality and independence during the investigation, the NASA wrote its own computer codes and chose its own configurations, flight conditions, and responses quantities.

Figure 4 summarizes the SDG and PSD results for a representative configuration, a drone vehicle modeled with two rigid-body modes and four symmetric flexible modes. Ten responses were investigated as indicated in the figure. All ratios of $\bar{\gamma}/\bar{A}$ fall between 8.45 (18.8% below 10.4) and 11.50 (10.6% above 10.4). The mean value of the ratios is 10.45, with a standard deviation of 0.91. These results indicate an approximate, rather than an exact, SDG-PSD overlap.

### Time-Correlated Gust Loads Using MFT and RPT

During the course of the investigation of the SDG-PSD overlap it was recognized that MFT[6] could also be used to determine worst-case responses and time-correlated gust loads. It was further proven that the time-correlated gust loads computed by MFT are theoretically identical to auto- and cross-correlation functions of RPT. Thus, auto- and cross-correlation functions of RPT may be interpreted as time-correlated gust loads.

Figure 5 contains a signal flow diagram of the steps necessary to generate a maximum dynamic response at some point in the aircraft structure using MFT. In the top half, a gust pre-filter is excited by an impulse of unit strength to generate an intermediate gust impulse response which, in turn, is the excitation to the aircraft. Also shown in the top half of the figure are several output load responses to the impulse, one of which, y, is chosen for the maximization process. Response y is then normalized by its RMS value, reversed in time (analogous to convolution) and used as input to the system as shown in the bottom half of the figure. This normalized and reversed signal is referred to as the matched excitation waveform. Intermediate and final outputs due to the matched excitation waveform are the critical gust profile and the time correlated responses, including the maximum response of the system.

Figure 6 contains comparisons of time-correlated gust loads (wing root bending and torsion moments) computed by MFT and RPT. Except for some slight differences in the peaks and troughs, depicted in the insets, results from the two methods show excellent agreement.

Computing time-correlated gust loads by MFT and RPT has the two advantages of being computationally fast and of solving the problem directly. In addition, the MFT and RPT approaches are general enough to be applied a variety of dynamic-response problems, such as taxi, landing, and maneuver loads in addition to gust loads.

**MFT and RPT Relative to Phased Design Loads**

During the course of the development of the MFT and RPT approaches, it was recognized that there was a relationship[7] between time-correlated gust loads computed by MFT and RPT and Phased Design Load Analysis (PDLA), a procedure commonly use in the aerospace industry. The relationship is as follows: Time histories of two time-correlated gust load responses, determined using either MFT or RPT, can be plotted as parametric functions of time and the resulting plot, when superimposed upon the PDLA design ellipse corresponding to the two loads, is tangent to the ellipse. The point of tangency corresponds to the design value of one load and the "phased" value of the other load. Figure 7 illustrates this relationship. Figure 7 contains normalized wing-root-bending-moment and wing-root-torsion-moment responses due to an excitation matched to root torsion. The parametric curve is seen to be tangent to the ellipse in the lower-right-hand corner.

The question is raised of whether or not it is possible for a parametric load plot to extend outside the associated design ellipse. If it is possible, the use of equi-probable loads design ellipses is not a conservative design practice in some circumstances.

## CONTROL LAW SYNTHESIS

### Integrated Structure/Control Law Design Methods

An integrated multidisciplinary aircraft design methodology currently under development within the ASEB is based on hierarchical problem decompositions, multilevel optimization methods, and design sensitivity analyses[8]. This methodology depends on the decomposition of the design problem into vehicle performance requirements and separate aerodynamic, structure, control, and/or propulsion subsystem requirements. The subsystem designs are obtained independently subject to a set of fixed design integration parameters, using existing design methods and tools. An iterative optimization method is used to satisfy the integrated vehicle design requirements through modification of the design integration parameters and repeated subsystem designs. Subsystem design sensitivity data relative to the design integration parameters are used as the gradient information for the optimization procedure. The method is illustrated schematically in Figure 8.

One application of the hierarchal integrated design methodology is to the design of aircraft control laws and structure, including the effects of unsteady aerodynamic forces due to structural and control surface motions. This application requires the use of aircraft dynamic response design requirements and a control law design method which reflects the actual feedback signals of the aircraft. Both of these requirements necessitated development and validation of appropriate design sensitivity expressions. Linear Quadratic Gaussian (LQG) optimal control law methods were selected for the control law design. Aircraft dynamic response criteria considered include time responses to control surface motions and discrete aerodynamic gusts, stochastic responses to random gust environments, closed-loop system eigenvalues, and open- and closed-loop frequency responses.

The sensitivity developments have recently been completed[15]. Results of the application of the approach to an aeroservoelastic aircraft example are summarized in Reference 9. A typical sensitivity expression validation result from ~~Reference 9~~ is shown in Figure 9. This figure shows the percentage error in predicting changes in mean square aircraft pitch rate response due to random gusts using the sensitivity results for parametric variations in the wing bending frequency (stiffness). The sensitivity result used for this figure includes the effects of the change in the LQG control law design due to the changes in the aircraft wing bending frequency. This type of sensitivity result is used in the hierarchal integrated design method as gradient information to determine values for the design integration parameter, which in this case would be the wing bending frequency. In the hierarchal design method, the wing bending frequency parameter would be selected to improve the pitch rate response of the aircraft due to the gust environment. The parameter would influence both the structural and control law designs resulting in improved dynamic response characteristics of the aircraft. The results in the figure show that the analytical sensitivity developments of Reference 15 provide good estimates of the response changes for relatively large variations in the design integration parameter.

### Stability Robustness and Singular Value Constraints

To adequately represent the aeroelastic response characteristics of a flexible flight vehicle the small perturbation dynamic equations of motion need to include the important rigid, flexible and control surface modes. When these equations are transformed into state-space form for control design tasks or for simulation, rational function approximations of the unsteady aerodynamics are required resulting in a large-order design model. A control law design for such a system is expected to satisfy multiple conflicting design requirements on the dynamic loads, RMS responses, control-surface deflection and rate limitations, as well as maintain certain guaranteed stability margins based on the system singular values. Optimal control theory is a procedure for obtaining robust control laws for the linear system. Because the resulting control law is usually of the same order or higher than the design model, it becomes difficult to implement the control law for practical application. One approach[10] for obtaining a low-order, robust multi-input/multi-output (MIMO) digital control law design for application to flexible vehicles is being developed within the ASEB.

This design procedure minimizes an LQG-type cost function while satisfying a set of constraints on design loads, responses and stability margins. Analytical expressions for the gradients of the cost function and the constraints with respect to the control law design variables are used to facilitate rapid numerical convergence. This step in the design process provides the full-order LQG analog control law. To obtain the more practical, low-order system various reduction and optimization techniques are applied prior to the discretization process. When the control law is discretized, the stability robustness generally deteriorates requiring further optimization using constraints on both the responses and on the minimum singular values.

To demonstrate the application of the synthesis procedure[11] a MIMO discrete feedback control system was designed for the gust load alleviation (GLA) problem of a remotely-piloted drone aircraft (Figure 10) being excited by a random vertical gust (Dryden Spectrum). The goal of the application is to design a low-order, robust digital GLA control law to reduce the open loop RMS bending moment and shear force at the wing root by 50% without increasing the outboard torsion and bending moment or exceeding control-surface deflection and rate constraints. The control system used compensated sensors from the fuselage and wing to command symmetric elevator and aileron deflections.

Figure 11 shows a comparison of RMS responses and control surface deflections for the open loop system and for a sequence of second-order GLA control laws. The RMS values of wing root bending moment (WRBM), wing root shear (WRS), wing outboard bending moment (WOBM) and wing outboard torsion moment (WOTM) are normalized to their open loop values and control-surface deflections and rates are normalized to their maximum allowable values. Control law-I is obtained by digitization of a continuous control law obtained via reduction of a full-order LQG design. This control law does not satisfy any of the design requirements. After an unconstrained optimization control law-II is obtained which satisfies all the RMS response requirements except that on the WOBM. An optimization sequence using RMS load constraints provides control law-III which satisfies all the constraints except stability margin. When singular value constraints are used in the optimization process (resulting in control law-IV) stability margins are improved, but at the cost of a slight increase in the RMS responses. The application of control law-IV satisfies all the design load requirements and control surface deflection and rate constraints while providing acceptable stability margins.

## APPLICATION

### Active Flexible Wing Program

To extend the state-of-the-art in active controls into more challenging and rewarding areas of application the NASA LaRC is performing cooperative active control system investigations[12] using the AFW aeroelastic wind-tunnel model with Rockwell International. The objective of these investigations is to obtain experimental data for validating analysis, design and test methodologies associated with multifunction digital systems required to control and use, in a favorable way, the aeroelastic response characteristics of flexible aircraft.

#### AFW Wind Tunnel Model

Figure 12 shows a picture of the AFW wind tunnel model mounted in the NASA Transonic Dynamics Tunnel (TDT). This model is an aeroelastically scaled, full-span representation of an advanced tailless fighter configuration. It has two leading-edge and two trailing-edge control surfaces on each wing panel driven by seven vane-type rotary actuators (Figure 13) powered by an onboard hydraulic system. The model will be mounted along the test section centerline by a sting mount that utilizes an internal ball bearing arrangement and a roll degree-of-freedom brake to allow the model to either roll about the sting axis or to be held fixed. In addition, an actuator located at the model center-of-gravity is available for remotely positioning the angle-of-attack.

To demonstrate flutter suppression, ballast has been installed on the tip of each wing to create low speed flutter instabilities within the operating envelop of the TDT. Besides causing flutter, the tip ballast store was designed such that it can be used as a flutter stopper for model safety. The store (Figure 14) is attached to the wing by a pivot mechanism somewhat related to the decoupler pylon concept[16] conceived and evaluated by the NASA. The pivot mechanism uses a pitch brake such that when the brake is on for flutter testing the attachment between the wing and the ballast is essentially rigid; when the pitch brake is off (either manually or automatically), a spring element internal to the store provides a more flexible pitch stiffness thereby altering the structural dynamics of the model to increase the flutter speed.

#### Flutter Suppression System

The design goal for the digital flutter suppression system (FSS) is to increase the flutter dynamic pressure by a factor of two. Because two flutter modes (symmetric and antisymmetric) fall within this goal, the FSS designs must be capable of suppressing both modes simultaneously. Four control law design approaches are being investigated. These approaches include: 1) a LQG method using order-reduction and optimization techniques with inequality constraints[13]; 2) a direct digital, gain-scheduled method based on LQG techniques; 3) a procedure that uses modal velocities of the critical flutter modes based on a blending of available accelerometer signals; and 4) an eigensystem assignment technique[17] that employs a forward path compensator and a feedback matrix.

A candidate FSS using the leading edge and trailing edge outboard control surfaces with the two collocated accelerometers is shown in Figure 15. The digital controller was designed using the LQG method. The FSS was shown to provide a large increase in flutter dynamic pressure with respectable stability margins without gain scheduling.

#### Rolling Maneuver Load Alleviation System

The AFW approach for roll control is to twist the flexible-wing structure into an optimum shape by actively deflecting multiple leading and trailing edge control surfaces on each wing panel. The design goal for the rolling maneuver load alleviation (RMLA) system is to reduce wing loads at multiple points by 20 percent with direct load feedback (strain-gage signals) while maintaining a fixed roll rate.

#### Digital Controller

One of the primary objectives of the AFW Program is to gain practical experience in designing, assembling, and implementing a real-time MIMO digital controller and in developing the hardware interface associated with the controller. The hardware layout for the interface rack and the digital controller is shown in Figure 16. The interface rack contains the circuitry for processing the signals coming from or going to the wind-tunnel model. The circuitry includes low-pass filters, antialiasing filters, and electrical isolation networks.

The digital controller consists of a Sun 3/160 Workstation with several special purpose processors linked to the workstation via a bus. These processors include a digital signal processor (DSP), an array processor, and data translation boards. The data translation boards provide the analog-to-digital (A/D) and digital-to-analog (D/A) conversions required between the model and the controller. The DSP provides the management of all signal processing and the scheduling of the control laws. As bus master, the DSP sends the digital control commands for the actuators to the D/A, sends commands to the array processor to implement the desired FSS, roll trim, and RMLA control laws, and adds digitized model excitations or bias commands to adjust camber. The array processor provides the high-speed floating-point arithmetic computations for the control laws.

#### Hot-Bench Simulation

To test the functionality of the total system the digital controller will be coupled to a Hot-Bench Simulation (HBS) (Figure 17) of the model/wind-tunnel system. The Advanced Real-Time Simulation (ARTS) System at the LaRC will be used during this program. The ARTS uses CYBER 175 computers connected to the simulation site by means of a 50-megabit-per-second fiber optic digital data network called the Computer Automated Measurement and Control (CAMAC). The CAMAC interface converts CYBER 175 digital signals to analog signals to represent the AFW model. These signals are then passed to the AFW controller through the analog NASA/Rockwell interface rack. The CAMAC also converts analog signals coming from the AFW controller/interface box to digital signals to be sent to the CYBER 175.

The HBS is being used to evaluate the operational characteristics of the flutter stopper, asymmetric effects on control law performance, controller functionality as integrated into the total model, and any nonlinear problems that might be identified.

#### Testing

Ground tests will be conducted to obtain data for validating the math models at zero-airspeed and to verify the model's structural integrity. Ground-vibration tests are being conducted to measure the vibration frequency, mode shape, and damping for each of the important symmetric and antisymmetric modes. In addition, transfer functions are being measured for all control surface actuators using several different amplitude signals to evaluate the nonlinear effects and for certain control surface/sensor combinations.

The goals of the first wind-tunnel entry scheduled for August 1989 are to measure control surface stability derivatives, define the passive flutter boundaries of the model, and demonstrate the RMLA system and the FSS, separately. The goal of the second test entry scheduled one year later is to investigate multifunction digital control law design capability by demonstrating FSS and RMLA, simultaneously.

## CONCLUDING REMARKS

The Aeroservoelasticity Branch at the NASA Langley Research Center is actively involved in advancing the state-of-the-art in predicting and preventing aeroelastic phenomena and adverse ASE interactions on existing and future configurations. Not to long ago "ASE interactions" were considered to be detrimental to the aircraft's stability and control. ASE has now entered the lime light as a viable design consideration for meeting the multimission requirements being imposed on future designs. A major thrust within the Aeroservoelasticity Branch is to enhance and validate modeling, analysis and design methodologies so that aeroservoelasticity can play an increasing and ever demanding role in the design of flight vehicles to assure performance goals while satisfying minimum weight requirements.

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**Table 1 Rational Function Approximation Matrix Formulations**

| Aerodynamic Method | Character of $\hat{Q}$ | Aerodynamic Dimension |
|---|---|---|
| Least-Squares | Common denominator coefficients in each $\hat{Q}_{ij}$ | |
| | $\hat{Q}_{ij} = \sum_{k=1}^{n_L} (A_{(k+2)})_{ij} \frac{s}{s + \beta_k \frac{V}{b}}$ | $j * n_L$ |
| Modified Matrix Pade´ | Different number of and values for denominator coefficients for each column, $Q_j$ | |
| | $\hat{Q}_{ij} = \sum_{k=1}^{n_{L_j}} (A_{(k+2)})_{ij} \frac{s}{s + \beta_k \frac{V}{b}}$ | $\sum_j n_{L_j}$ |
| Minimum-State | Common denominator coefficients in each $\hat{Q}_{ij}$ | |
| | $\hat{Q}_{ij} = \sum_{k=1}^{n} \frac{\{D_i\}[E_j]s}{s + \beta_k \frac{V}{b}}$ | $n$ |

Table 1 Rational function approximation matrix formulations

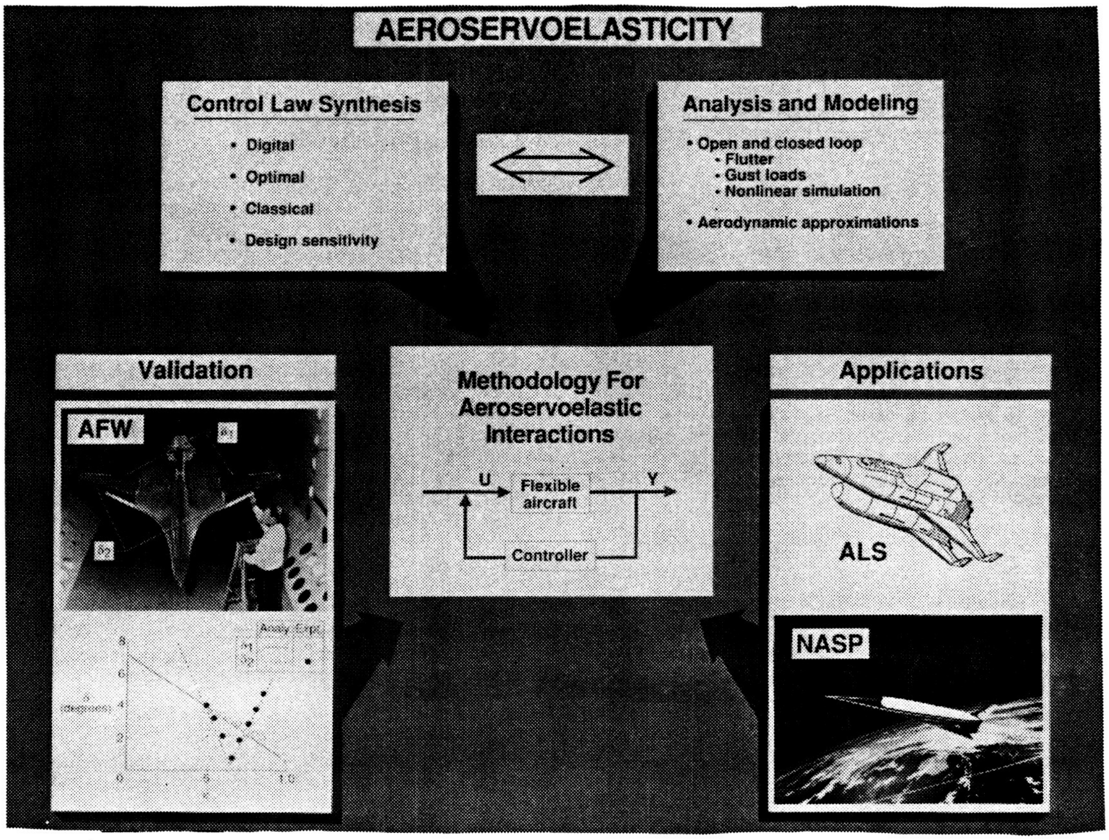



Figure 1 Research activities within the Aeroservoelasticity Branch at NASA LaRC

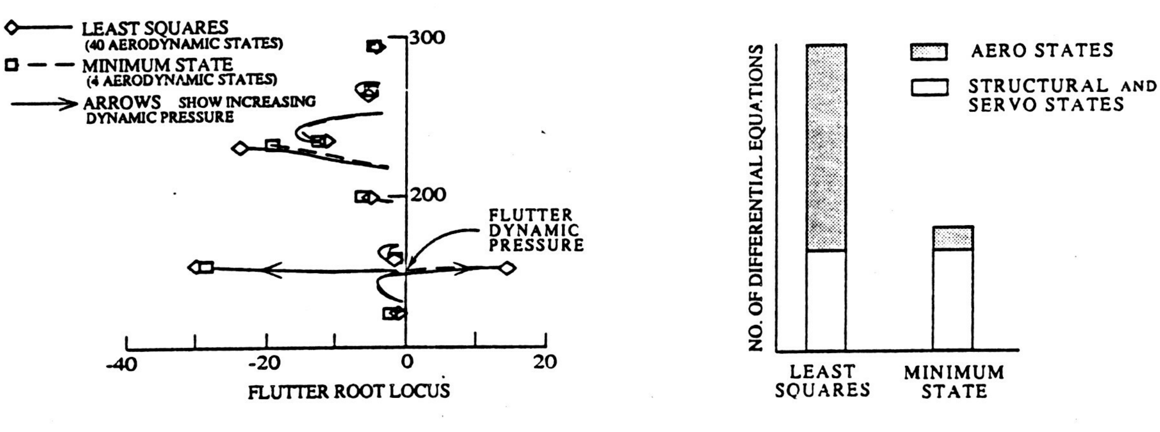



Figure 2 Comparisons between the Least Squares and the Minimum-State Methods for approximating unsteady aerodynamics

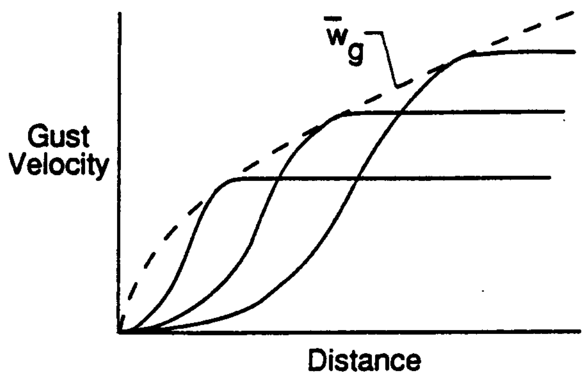
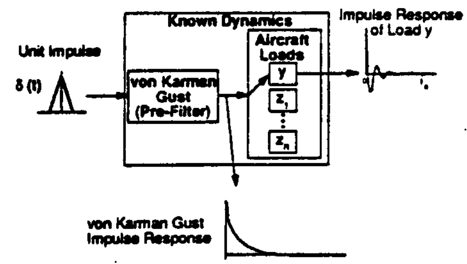



Figure 3 Family of equi-probable ramp-hold gusts for the SDG Method



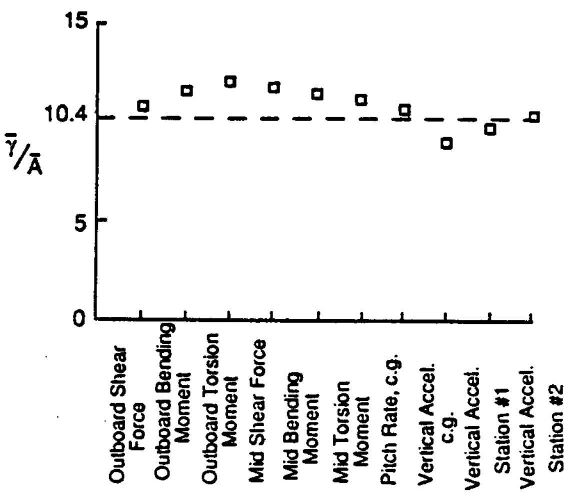




Figure 4 SDG and PSD comparisons for the DAST ARW-2 configuration

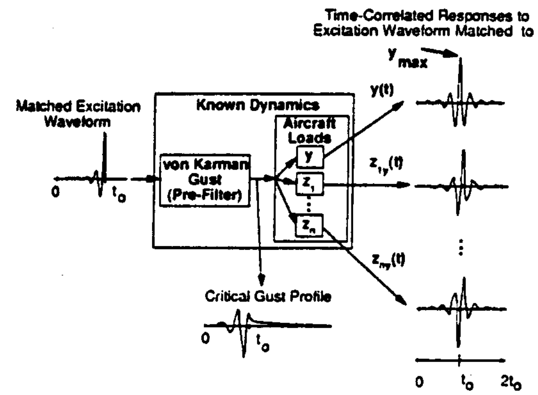



Figure 5 Signal flow diagram for MFT to predict time-correlated loads

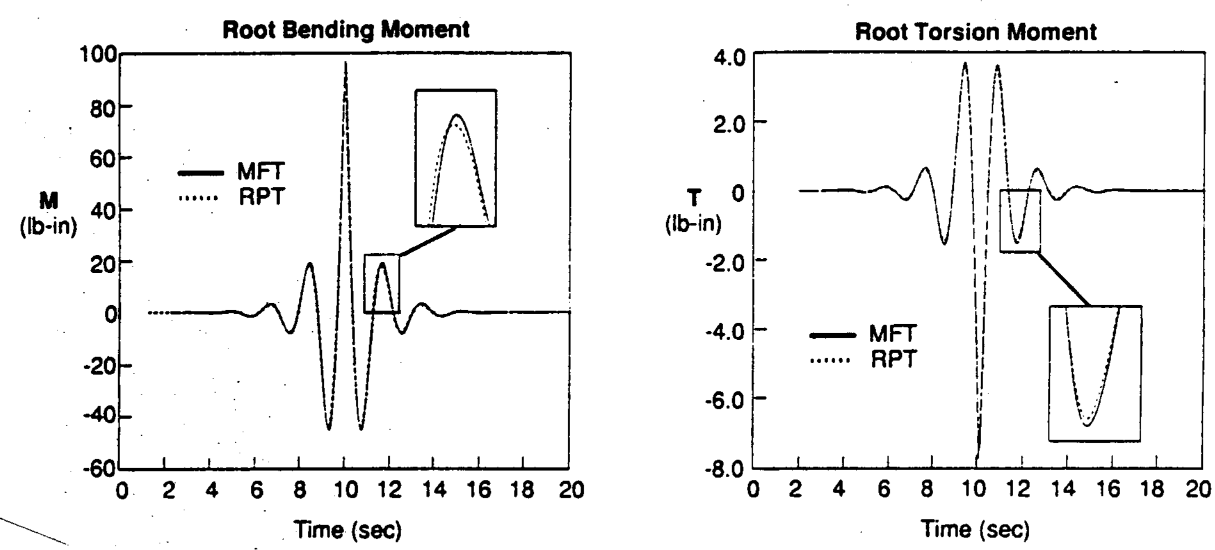



Figure 6 Comparisons of time-correlated gust loads determined using MFT and RPT

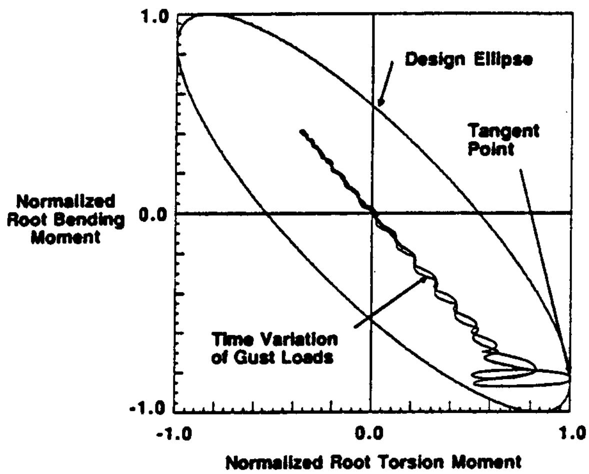



Figure 7 PDLA design ellipse with normalized parametric loads

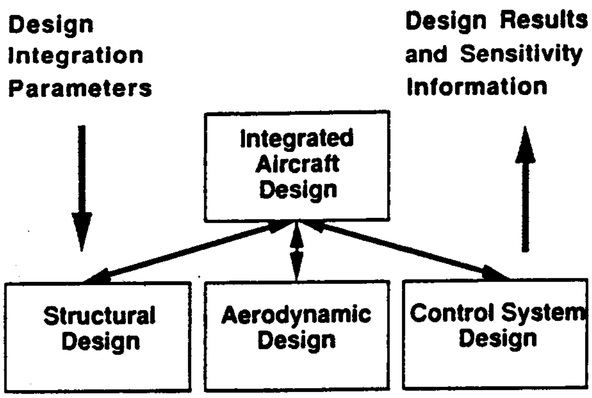



Figure 8 Methodology for multidisciplinary aircraft design

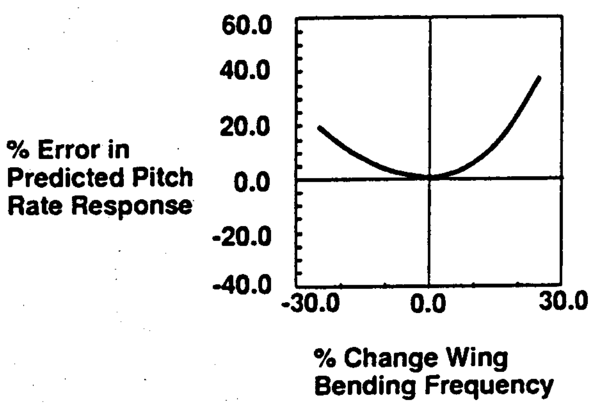



Figure 9 Predicted response based on bending frequency sensitivity gradients

| Physical Quantities | Design Objective |
|---|---|
| Wing Root Bending Moment | 50% reduction |
| Wing Root Shear | 50% reduction |
| Wing Outboard Bending Moment | No increase |
| Wing Outboard Torsion | No increase |
| Elevator Deflection | Within max limit |
| Elevator Rate | Within max limit |
| Aileron Deflection | Within max limit |
| Aileron Rate | Within max limit |

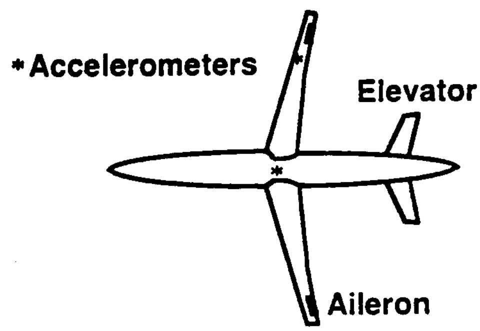



Figure 10 Synthesis methodology for digital robust control laws using constrained optimization

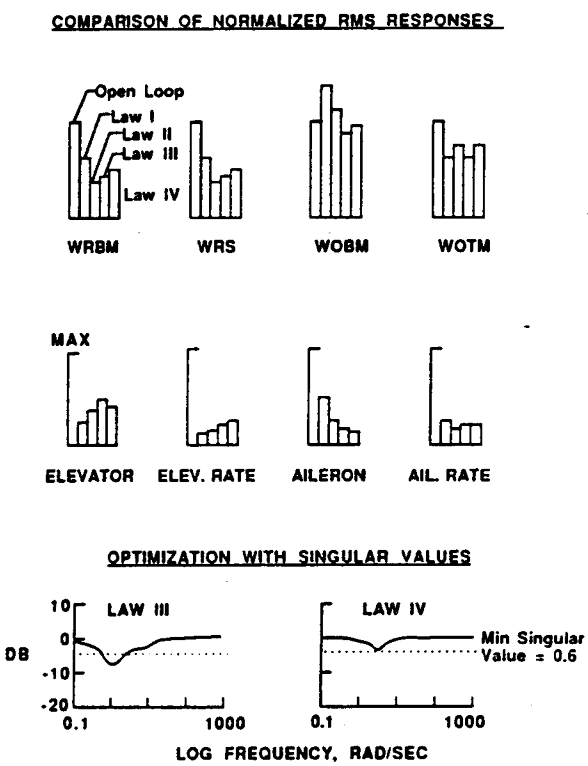



Figure 11 Improved stability robustness using singular value constraints

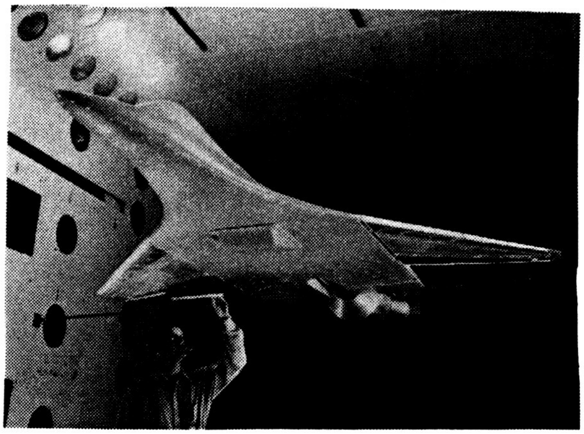


Figure 12 AFW model mounted in the NASA TDT

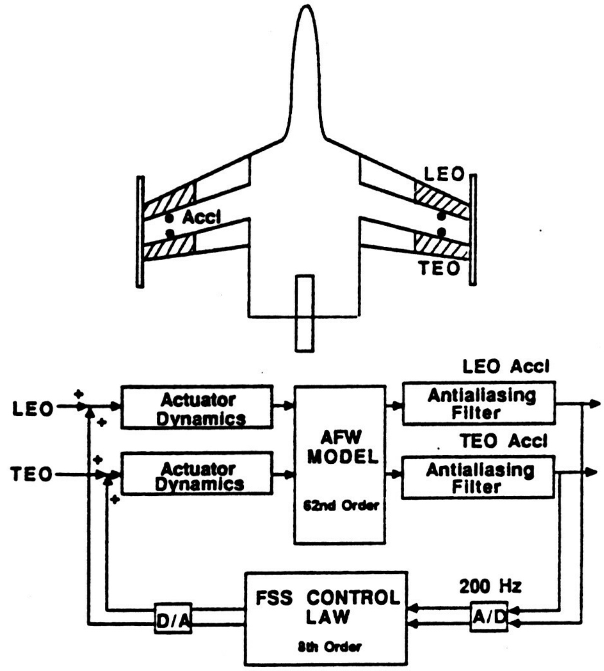



Figure 15 Candidate FSS Design

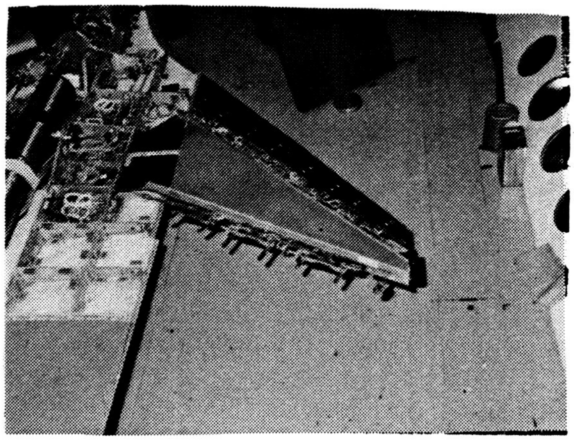


Figure 13 Internal details of the AFW model

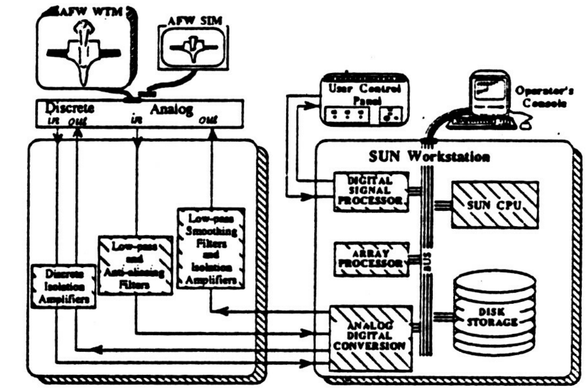



Figure 16 Schematic of the digital controller

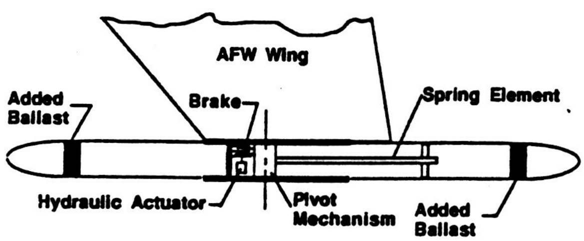



Figure 14 Tip ballast store for the AFW model

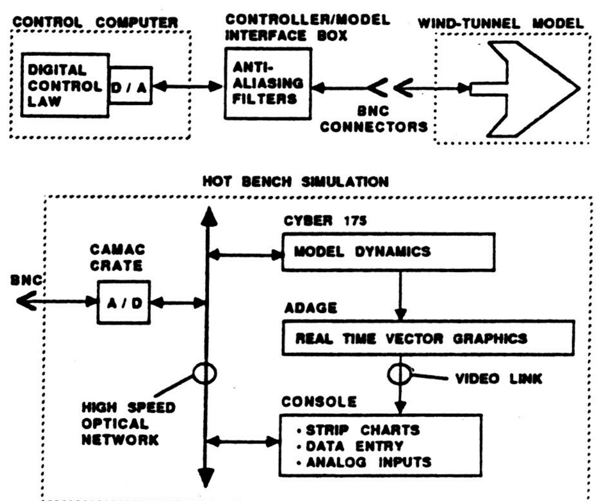



Figure 17 Schematic of near-real-time simulator

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15. Supplementary Notes

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16. Abstract

The objective of research in aeroservoelasticity at the NASA Langley Research Center is to enhance the modeling, analysis, and multidisciplinary design methodologies for obtaining multifunction digital control systems for application to flexible flight vehicles. This paper discusses recent accomplishments and presents a status report on current activities within the Aeroservoelasticity Branch. In the area of modeling, improvements to the Minimum-State Method of approximating unsteady aerodynamics are shown to provide precise, low-order aeroservoelastic models for design and simulation activities. Analytical methods based on Matched Filter Theory and Random Process Theory to provide efficient and direct predictions of the critical gust profile and the time-correlated gust loads for linear structural design considerations are also discussed. Two research projects leading towards improved design methodology are summarized. The first program is developing an integrated structure/control design capability based on hierarchical problem decomposition, multilevel optimization and analytical sensitivities. The second program provides procedures for obtaining low-order, robust digital control laws for aeroelastic applications. In terms of methodology validation and application the current activities associated with the Active Flexible Wing project are reviewed.

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