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STRUCTURAL DYNAMICS DIVISION RESEARCH AND TECHNOLOGY ACCOMPLISHMENTS FOR FY 1989 AND PLANS FOR FY 1990

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STRUCTURAL DYNAMICS DIVISION

RESEARCH AND TECHNOLOGY ACCOMPLISHMENTS FOR FY 1989

AND PLANS FOR FY 1990

SUMMARY

The purpose of this paper is to present the Structural Dynamics Division's research accomplishments for FY 1989 and research plans for FY 1990. The work under each branch (technical area) is described in terms of highlights of accomplishments during the past year and highlights of plans for the current year as they relate to five year plans and the objectives for each technical area. This information will be useful in program coordination with other government organizations, universities, and industry in areas of mutual interest.

ORGANIZATION

The Langley Research Center is organized into directorates as shown in figure 1. Directorates are subdivided into divisions and offices. The Structural Dynamics Division of the Structures Directorate consists of five branches as shown in figure 2. This figure lists the key people in the division which consists of 66 NASA civil servants and 12 members of the Army Aerostructures Directorate, USAARTA, Army Aviation Systems Command co-located at the Langley Research Center. Recent changes in key positions include the appointment of Mr. Robert V. Doggett, Jr. as Assistant Chief, and the selection of Dr. John B. Malone as Head of the Unsteady Aerodynamics Branch. Each branch represents a technical area and disciplines under the technical areas as shown in the figure.

The Division conducts analytical and experimental research in the five technical areas to meet technology requirements for advanced aerospace vehicles. The research focuses on the long range thrusts shown in figure 3. The Configuration Aeroelasticity Branch (CAB), Unsteady Aerodynamics Branch (UAB), and Aeroservoelasticity Branch (ASEB) all work in the area of the prediction and control of aeroelastic stability and response of aircraft and rotorcraft. The Landing and Impact Dynamics Branch (LIDB) conducts research on the crash dynamics of aircraft structures and on the technology for improving the safety and handling performance of aircraft during ground operations. The Spacecraft Dynamics Branch (SDB) conducts research on the prediction and control of the structural dynamic response of complex space structures.

FUNCTIONAL STATEMENT

The Division conducts analytical and experimental research in the areas of configuration aeroelasticity, aeroservoelasticity, unsteady aerodynamics, impact and landing dynamics, and spacecraft dynamics to meet technology requirements for advanced atmospheric and space flight vehicles. Develops analytical and computational methods for predicting and controlling aeroelastic instabilities, deformations, vibrations, and dynamic response. Investigates interaction of structure with aerodynamics and control systems, landing dynamics, impact dynamics, and resulting structural response. Evaluates structural configurations embodying new material systems and/or advanced design concepts for general application and for specific classes of new aerospace vehicles. Uses a broad spectrum of test facilities to validate analytical and computational methods and advanced configuration and control concepts. Develops research techniques to demonstrate safety from aeroelastic instabilities for new airplanes, helicopters, and space launch vehicles. Test facilities include the Transonic Dynamics Tunnel, the General Rotor Aeroelastic Laboratory, the Impact Dynamics Research Facility, the Aircraft Landing Dynamics Facility, the Space Structures Research Laboratory, and the Structural Dynamics Research Laboratory.

FACILITIES

The Structural Dynamics Division has four major facilities available to support its research as shown in figure 4.

The Transonic Dynamics Tunnel (TDT) is a maximum Mach 1.2 continuous flow, variable pressure wind tunnel with a 16-foot-square test section which normally uses air or a heavy gas (R-12) as the test medium. The maximum Reynolds number obtainable is approximately 10 million per foot in heavy gas and 3 million per foot in air. The TDT is a unique "National" facility that is used almost exclusively for testing of aeroelastic phenomena. Semi-span, side-wall mounted models and full-span sting mounted or cable-mounted models are used for aeroelastic studies of fixed wing aircraft. In addition, the Aeroelastic Rotor Experimental System (ARES) test stand is used in the tunnel to study the aeroelastic characteristics of rotors. The General Rotor Aeroelastic Laboratory (GRAL), located in an adjacent building, is used to setup the ARES test stand in preparation for entry into the TDT and for rotorcraft studies in hover. The new TDT Data Acquisition System, now in full operational use, is capable of simultaneous support of tunnel tests, GRAL tests and model checkout in the Calibration Lab. Because of heavy gas losses to the atmosphere as a result of deficiencies in the heavy gas reclamation system, tunnel testing in heavy gas will be highly restricted and most testing will be conducted in air. A major facility upgrade to improve the heavy gas reclamation system is in progress with scheduled completion in November 1991 after which normal operations will resume.

The Aircraft Landing Dynamics Facility (ALDF) is capable of testing various types of landing gear systems at velocities up to 200 kts. on a variety of runway surfaces under many types of simulated weather conditions. The ALDF consists of a 2800 ft. long rail system, a 2.0 million pound thrust propulsion system, a test carriage, and an arrestment system. Test articles can be subjected to vertical loads up to 65,000 lbs. or sink rate of 20 ft./sec on a wide variety of runway surface conditions. The facility provides for testing at speeds and sizes pertinent to large transport aircraft, fighter aircraft, and the Space Shuttle Orbiter.

The Impact Dynamics Research Facility (IDRF) is capable of crash testing full-scale general aviation aircraft and helicopters under controlled conditions. The facility is a 220 ft. high, 400 ft. long gantry structure which is the former Lunar Landing Facility. General aviation aircraft and helicopters weighing up to 20,000 lbs. can be tested up to 60 mph using a free-swinging pendulum approach or up to 100 mph with rocket assist. Attitudes can be adjusted for desired pitch, roll, and yaw parameters. Impact surfaces can be concrete or dirt. High speed motion pictures and 90 data channels are available to record the crash event. A vertical test apparatus is attached to one leg of the facility for drop-testing structural components. The facility is used to support in-house research and other agency programs (Army, Air Force, FAA).

The Structural Dynamics Research Laboratory (SDRL) is designed for conducting research on the dynamic and control response of spacecraft structures. The facilities in this laboratory include the 16 meter thermal vacuum chamber, and the main backstop area, and the tower test area. These facilities provide a variety of environmental simulation capabilities, including acceleration, vacuum and thermal radiation. The chamber has a 55-foot diameter, hemispherical dome with a 64-foot high peak, flat floor and option for a large centrifuge or a rotating platform. Access is by an airlock door and an 18-foot by 20-foot test specimen door. A vacuum level of 10 torr can be achieved within 120 minutes and, with diffusion pumps, 10^{-4} torr vacuum can be achieved within 160 minutes. A temperature variation of 100°F can be obtained in the chamber by using 250-sq. ft. of portable radiant heaters and liquid nitrogen cooled-plates. The Backstop Area is dominated by the 38-foot-high back-stop of I-beam construction. Test areas around this fixture are 15 x 35 x 38 feet high and a tower 12 x 12 x 95 feet high, both equipped with hydraulic and pneumatic supply lines. There are various size hoists and accessible platforms for suspension systems instrumentation and viewing. Closed circuit television is available for monitoring research studies. Test articles can be excited by several types of actuators and small shakers. State-of-the-art capability is available for signal conditioning and processing including GenRad 2515 digital signal processing systems and a VAX 11-780/EAI 2000 hybrid computer system for simulation and on-line test control.

The Space Structures Research Laboratory (SSRL) is a large open room of dimensions 75 x 84 x 67 feet high. It has a truss framework spanning most of the ceiling area from which lightweight test articles can be supported. A hoist having a capability of approximately 2 tons is available. Access is through a 12' x 12' door. A 10 x 10-foot backstop is available in one corner. A control room equipped for structural and structural dynamics data acquisition and analyses is available. Test equipment such as shakers, sensors, and signal conditioning equipment is shared with the SDRL and similar closed-circuit test monitoring and support are available.

FY 1989 ACCOMPLISHMENTS

Configuration Aeroelasticity Branch

The Configuration Aeroelasticity Branch conducts research (figure 5) to develop the aeroelastic understanding and prediction capabilities needed to apply new aerodynamic and structural concepts to future flight vehicles and to determine and to solve the aeroelastic problems of current designs; to evaluate the aeroelastic characteristics of new rotor systems; and to determine, analytically and experimentally, effective means for predicting and reducing helicopter vibrations. This work is more clearly identified in figure 6 which shows the five year plan of the four disciplines and their expected results.

The Configuration Aeroelasticity FY 1989 accomplishments listed below are highlighted in figures 7 through 22.

Aircraft Aeroelasticity:

- Effects of Planform Curvature on Flutter of Swept Wing Determined in TDT
- High Speed Portable System Developed For Dynamic Data Acquisition/Analysis
- Analysis of Highly Swept Delta Wing Correlates Well With Experiment
- Flutter and Divergence of All-Moveable Delta Wing Determined
- Aileron-Buzz Characteristics For a Generic NASP Model Are Determined in TDT
- Open- and Closed-Loop Flutter Characteristics of Modified AFW Model Determined in TDT
- Flutter Characteristics of a Joined-Wing High-Altitude Vehicle Determined
- F-106B Flight Envelope Flutter Cleared For Vortex Flap Experiments

Aircraft Aeroelastic Validation:

- Effects of Three Tip Shapes on Wing Flutter Determined in TDT
- TDT Tests Show That Ground Wind Loads Are No Problem for Atlas II

Rotorcraft Aeroelasticity:

- Use of Blade Non-Structural Mass Reduces Rotorcraft Vibrations
- BVI Noise Reduced Using Higher-Harmonic Pitch Control

Rotorcraft Structural Dynamics:

- Significance of Warping Determined Analytically for Composite Rotor Blades with Extension Twist Coupling
- P-Version Tapered Beam Finite Element Developed to Improve Vibration Analyses
- Ground Vibration Test of Bell All-Composite (ACAP) Airframe Conducted as Part of Difficult Components Studies
- Anomaly Identified in Use of Damping Treatments For Reducing Structural Vibrations

Unsteady Aerodynamics Branch

The Unsteady Aerodynamics Branch conducts research (figure 23) to develop, validate and apply a set of computational methods for predicting steady and unsteady aerodynamic airloads and the aeroelastic characteristics of flight vehicles. The branch also supports research activities aimed at the generation of experimental data bases needed for computer code validation. Current research topics reflect a major emphasis on accurately predicting transonic aeroelastic phenomena, such as wing "flutter-speed dip" and aileron buzz. Additionally, research topics such as dynamic vortex-structure interactions, dynamic loads and buffet prediction are becoming major areas of investigation. Interest in these latter topics is due to the emerging importance of the high angle-of-attack maneuvering flight capabilities demonstrated by a number of current high-performance aircraft. The branch research program is outlined in figure 24 which shows the five year plan for aerodynamic analysis methods, aeroelastic prediction techniques and experimental investigations.

The Unsteady Aerodynamics FY1989 accomplishments listed below are highlighted in figures 25 through 35.

Aerodynamic Analysis Methods:

Structured Grid Methods -

- Finite-Difference Mesh Design for the Transonic Small Disturbance Equation
- Unsteady Vortex-Dominated Flows Computed Using a Structured-Mesh, Navier-Stokes Solver

Unstructured Grid Methods -

- Adaptive Mesh Refinement Capability Developed for Accurate Vortical Flow Computation
- New Unsteady Euler Algorithm Improves Accuracy and Efficiency of Unstructured Grid Methodology
- Temporal Adaptive Euler Method Developed for Unsteady Aerodynamic Analysis
- Graphics Code Developed to Permit Rapid Visualization of CFD Results for 2-D Unstructured Meshes

Aeroelastic Prediction Techniques:

Structured Grid Methods -

- CAP-TSD Viscous Code Confirms Simple Criteria for Aileron Buzz
- Flutter Analysis of a Highly Swept Delta Wing with Thickness
- CFL3D Code Modified for Aeroelastic Analysis

Unstructured Grid Methods -

- Euler Flutter Analysis of Alrfoils Demonstrated Using Unstructured Dynamic Meshes

Experimental Investigations:

- Clipped Delta Canard Transonic Flowfield Visualized

Aeroservoelasticity Branch

The Aeroservoelasticity Branch (figure 36) conducts research to develop or enhance methods for the modelling, analysis and synthesis of multifunctional active control systems which have application to flexible flight and space structures; to perform wind tunnel and flight experiments for obtaining data to validate the new and improved methodologies; and to provide technical support to advanced NASA and DOD flight vehicle projects for insuring that ASE technology is available for use and that the flight envelope of the vehicle is free of unstable aeroelastic phenomena or adverse aeroservoelastic interactions. The scope of this work is more explicitly identified in figure 37 which shows the five-year plan of the three major thrusts and their expected results.

The Aeroservoelasticity Branch FY 1989 accomplishments listed below are highlighted by figures 38 through 46.

Analysis and Modelling:

- Novel Approach Developed for Determining Maximized and Time-Correlated Gust Loads
- Nonlinear Method Predicts Static and Resultant Dynamic Aeroelastic Behavior
- Hypersonic Aeroservoelasticity Method Reduces Thermal Heating Effects on Flutter

Control Law Synthesis:

- Multilevel Method Integrates Independent Structure and Control Law Designs
- Advanced Integrated Structure/Control Design Method Improves Space Structure System Performance
- Low-Order Digital Flutter Suppression Control Laws Designed for AFW Wind-Tunnel Model

Applications and Validations:

- Real-Time Procedures Developed for Evaluating the Performance and Stability of Digital, Multivariable Controllers
- AFW Digital Controller Designed, Assembled, Coded, Validated and Tested
- First Active Flexible Wing Wind-Tunnel Test Successfully Completed in the TDT

Landing and Impact Dynamics Branch

The Landing and Impact Dynamics Branch (figure 47) has two major facilities, the Aircraft Landing Dynamics Facility (ALDF) and the Impact Dynamics Research Facility (IDRF), for conducting research. The landing dynamics group of the branch conducts research to advance technology for safe, economical all-weather aircraft ground operations including the development of new landing gear systems. The group coordinates in-house research, grants, and contracts with U.S. tire industry to achieve the technology required. The impact dynamics group conducts research to obtain a better understanding of response characteristics of generic composite aircraft components subjected to crash loading conditions and to develop/enhance analytical tools capable of predicting response of composite structures. In-house research, grants and contracts are also utilized to achieve the technology and to develop better structural concepts capable of providing energy absorption and reduced crash loads. The work of the Landing and Impact Dynamics Branch is more clearly identified in figure 48 which shows the five-year plan of the disciplines in both landing and impact dynamics along with their expected results. The Landing and Impact Dynamics Branch FY 1989 accomplishments listed below are highlighted by figures 49 to 58 along with descriptive material.

Impact Dynamics:

- Role of Structural Components Evaluated from Impact Response of Graphite-Epoxy Floor Sections
- Impact Scaling of Unidirectional Composite Beams Verified Experimentally
- Experimental Determination of Composite Beam Bending Stiffness Corrects Overprediction by Lamination Theory in Large Deflection Beam Analysis
- Substantial Reductions in Tensile Strength of Fiber Composites Observed Between Scaled Models and Full Scale Structures
- DYCAST Analysis of Possible NTF Accident Scenario Matches Post-Mishap Observations
- New Analytical Sensitivity Derivatives Provide Capability to Evaluate Lamination and Material Effects on the Vibrational Response of Composite Fuselage Frames

Landing Dynamics:

- Tire Analysis Codes Validated with Experimental Measurements of Space Shuttle Nose Gear Tire
- Aircraft Landing Dynamics Facility Capabilities Enhanced
- Performance of F-106 Actively-Controlled Nose Gear Improved
- Preliminary ALDF Test Results Define Mechanical and Friction Characteristics of Bias-Ply, Radial-Belted, and H-Type Tires

Spacecraft Dynamics Branch

The Spacecraft Dynamics Branch conducts research (figure 59) on the dynamics and control of advanced spacecraft. Analytical methods are developed and verified to advance the state-of-the-art in large flexible complex space structures such as the Space Station Freedom, earth observation platforms and large area antennas. A major activity involving the advancement of ground test methods for control/structure interaction is ongoing along with laboratory upgrading necessary to accomplish this goal. The scope of this work is more explicitly identified in figure 60 which shows the five year plan of the organization's major thrusts and their expected results.

The Spacecraft Dynamics Branch FY 1989 accomplishments listed below are highlighted in figures 61 through 69.

Ground Test Methods:

- Readiness of MINI-MAST CSI Testbed Facility Demonstrated
- Analytical Simulation Provides Pretest Checkout of Mini-Mast Closed Loop Experiment

Optimal Spacecraft Performance:

- New Adaptive Modal State Estimator Provides Improved Estimator Performance for Active Control
- Candidates Identified for Line-of-Sight Controller on a Proposed CSI Structure
- Integrated Design for Earth Pointing Satellite
- Learning Control Method Developed and Applied in Flexible Panel Slewing Analysis

Multibody Dynamics:

- Reboost Analysis Developed and Utilized for Space Station Structural Characterization Experiment
- Mars Evolutionary Space Station Studies Indicate Minimal Impact on Solar Dynamic Performance During Reboost
- Space Crane Dynamics and Operations Investigated Using LATDYN

PUBLICATIONS

The FY 1989 accomplishments of the Structural Dynamics Division resulted in a number of publications. The publications are listed below by organization in the categories of journal publications, formal NASA reports, conference presentations, contractor reports, tech briefs, and patents.

Division Office

Journal Publications:

1. Pinson, L. D.: Recent Advances in Structural Dynamics of Large Space Structures. Acta Astronautica, Volume 19, No. 2, February 1989, p. 161-170.

Formal NASA Reports:

2. Doggett, R. V., Jr.: Some Effects of Aerodynamic Spoilers on Wing Flutter. NASA TM-101632, July 1989.

3. Doggett, R. V., Jr. and Soistmann, D. L.: Some Low-Speed Flutter Characteristics of Simple Low-Aspect-Ratio Delta Wing Models. NASA TM 101547, January 1989.

4. Doggett, R. V., Jr.; and Soistmann, D. L.: Some Low-Speed Flutter Characteristics of Simple Low-Aspect-Ratio Delta Wing Models. NASA TM-101547, January 1989.

5. Doggett, R. V., Jr.; Soistmann, D. L.; Spain, C. V.; Parker, E. C.; and Silva, W. A.: Some Experimental Transonic Flutter Characteristics of Two 72° Sweep Delta Wing Models. NASP TM-1079, August 1989.

6. Doggett, R. V., Jr.; Soistmann, D. L.; Spain, C. V.; Parker, E. C.; and Silva, W. A.: Experimental Transonic Flutter Characteristics of Two 72° Sweep Delta-Wing Models. NASA TM-101659, November 1989.

7. Gardner, J. E.: Structural Dynamics Division Research and Technology Accomplishments for FY 1988 and Plans for FY 1989. NASA TM-101543, January 1989.

Conference Presentations:

8. Doggett, R. V., Jr.; Soistmann, D. L.: Some Low-Speed Flutter Characteristics of Simple Low-Aspect-Ratio Delta Wing Models. AIAA Paper No. 89-1325-CP. Presented at the AIAA, ASME, et. al., 30th Structures, Structural Dynamics and Materials Conference, April 3-5, 1989, Mobile, AL.

9. Pinson, L. D.: NASA Space Objectives. Presented at the AIAA/AFOSR Workshop on Microgravity Simulation in Ground Vibration Testing of Large Space Structures, November 1989, Denver, CO.

Contractor Reports:

10. Joào Luiz Filgueiras de Azevedo, Stanford University, Stanford California: Transonic Aeroelastic Analysis of Launch Vehicle Configurations. Prepared for Langley Research Center under Grant NGL 05-020-243, NASA CR 4186.

Configuration Aeroelasticity Branch

Journal Publications:

11. Hinnant, H. E. and Hodges, D. H.: Nonlinear Analysis of a Cantilever Beam. AIAA Journal, Vol. 26, No. 12, December 1988, p. 1521-1527.

12. Lakin, W. D.; and Kvaternik, R. G.: An Integrating Matrix Formulation for Buckling of Rotating Beams Including the Effects of Concentrated Masses. International Journal of the Mechanical Sciences, Vol. 31, No. 8, March 1989, p. 569-577.

13. Young, M. I.: Spot Damping Anomalies. Journal of Sound and Vibration, Vol. 131, No. 1, May 1989.

14. Young, M. I.: On Dynamic Stability Boundaries for Binary Systems. Accepted for publication as a Letter to the Editor in the Journal of Sound and Vibration, July 1989.

15. Young, M. I.: On Lightly Damped Linear Systems. Accepted for publication as a Letter to the Editor in the Journal of Sound and Vibration, November 1989.

Formal NASA Reports:

16. Brooks, T. F.; Booth, E. R., Jr.; Jolly, J. R., Jr.; Yeager, W. T., Jr., and Wilbur, M. L.: Reduction of Blade-Vortex Interaction Noise Using Higher Harmonic Pitch Control. NASA TM 101624 and AVSCOM TM 89-B-005, July 1989.

17. Bruce, R. A.; Hess, R. W.; Rivera, J. A., Jr.: A Vapor Generator for Transonic Flow Visualization. NASA TM 101670, October 1989.

18. Cole, S. R.: Flutter of a Low-Aspect-Ratio Rectangular Wing. NASA TM 4116, June 1989.

19. Hinnant, H. E.: Derivation of a Tapered P-Version Beam Finite Element. NASA TP 2931 and AVSCOM Technical Report 89-B-002, August 1989.

20. Kvaternik, R. G.; Murthy, T. S.; and Nixon, M. W. (Contributing Authors): Integrated Multidisciplinary Optimization of Rotorcraft: A Plan for Development. NASA TM 101617; AVSCOM TM 89-B-004, May 1989.

21. Kvaternik, R. G.: Langley Rotorcraft Structural Dynamics Program--Background, Status, Accomplishments, Plans. NASA TM 101618, June 1989.

22. Kvaternik, R. G. and Murthy, T. S.: Airframe Structural Dynamic Considerations in Rotor Design Optimization. NASA TM 101646, August 1989.

23. Rivera, J. A., Jr.: Experimental and Analytical Investigation of the Effect of Spanwise Curvature on Wing Flutter at Mach Number of 0.7. NASA TM 4094, February 1989.

24. Sandford, M. C.; Seidel, D. A.; Eckstrom, C. V.; and Spain, C. V.: Geometrical and Structural Properties of an Aeroelastic Research Wing (ARW-2). NASA TM 4110, April 1989.

25. Wilbur, M. L.: Application of a PC Based, Real-Time, Data-Acquisition System in Rotorcraft Wind-Tunnel Testing. NASA TM 4119 and AVSCOM TM 89-B-003, July 1989.

Conference Presentations:

26. Cazier, F. W., Jr.; Ricketts, Rodney H.: Supersonic Transport Studies in the Transonic Dynamics Tunnel. Presented at the High Speed Airframe Integration Workshop, Langley Research Center, June 1989.

27. Cazier, F. W., Jr.: Understanding and Preventing Flutter. Presented at the Experimental Aircraft Association Annual Convention in Oshkosh, WI, August 1989.

28. Durham, M. H.; Ricketts, R. H.: Flutter of a Joined-Wing High-Altitude Vehicle. Presented by Mike Durham at the Aerospace Flutter and Dynamics Council Meeting, October 1989, Orlando, FL.

29. Hamouda, N.: Aeromechanical Stability. Presented at the Army Aerostructures Directorate/NASA Rotorcraft Technology Transfer Meeting, March 1989, Langley Research Center.

30. Hinnant, H.: P-Version Beam Finite Element Analysis. Presented at the Army Aerostructures Directorate/NASA Rotorcraft Technology Transfer Meeting, March 1989, Langley Research Center.

31. Kvaternik, R. G.: Ground Vibration Test of Helicopter Airframe Identifies Important Contributors to Vibratory Response. Presented at the LaRC Senior Management Research Review (SMRR), February 1989.

32. Kvaternik, R. G.: Rotorcraft Structural Dynamics Research. Presented to visitors from ONERA and the Fluid Mechanics Institute of Marseille, May 1989, Langley Research Center.

33. Kvaternik, R. G.: Rotorcraft Structural Dynamics Research at NASA Langley. Presented to McDonnell Douglas Helicopter Company, August 1989.

34. Lake, R.: Structural Dynamics Analysis of Composites. Presented at the Army Aerostructures Directorate/NASA Rotorcraft Technology Transfer Meeting, March 1989, Langley Research Center.

35. Mirick, P.: ARES Models Capabilities. Presented at the Army Aerostructures Directorate/NASA Rotorcraft Technology Transfer Meeting, March 1989, Langley Research Center.

36. Nixon, M.: Structural Tailoring with Composites. Presented at the Army Aerostructures Directorate/NASA Rotorcraft Technology Transfer Meeting, March 1989, Langley Research Center.

37. Nixon, M. W.: Analytical and Experimental Investigations of Extension-Twist Coupled Structures. A Thesis submitted to the Faculty of The School of Engineering and Applied Science, The George Washington University in partial fulfillment of the requirements for the degree of Master of Science. May 1989.

38. Ricketts, R. H.: Langley Transonic Dynamics Tunnel- A National Facility. Presented to the George Washington University Wind Tunnel Techniques Course, March 1989, Hampton, VA.

39. Ricketts, R. H.: Langley Transonic Dynamics Tunnel (TDT). Presented at the OAST Wind-Tunnel Calibration Workshop, NASA Langley Research Center, April 1989.

40. Ricketts, R. H.; Spain, V.; Parker, E.; Soistmann, D.; and Linville, T.: Generic NASP Studies in the Langley Transonic Dynamics Tunnel. Presented at the meeting of the Aerospace Flutter and Dynamics Council, May 1989, Dallas, TX.

41. Ricketts, R. H.: NASA Langley Aeroelastic Models Program. Presented at the meeting of the Aerospace Flutter and Dynamics Council, May 1989, Dallas, TX.

42. Ricketts, R. H.: High-Speed Rotorcraft Aeroelasticity and Structural Vibration. Presented at the High-Speed Rotorcraft Meeting, LaRC, November 14, 1989.

43. Rivera, J. A., Jr.: Laser Light Sheet Flow Visualization System developed for the TDT. Presented at the LaRC Senior Management Research Review, April 1989.

44. Soistmann, D.; Gibbons, M.: Some Analytical Transonic Flutter Characteristics of a Highly Swept Delta Wing. Presented at the 7th NASP Technology Symposium, October 1989, NASA Lewis Research Center.

45. Singleton, J.: Hub Design. Presented at the Army Aerostructures Directorate/NASA Rotorcraft Technology Transfer Meeting, March 1989, Langley Research Center.

46. Wilbur, M.: Rotor Aeroelastic Tailoring. Presented at the Army Aerostructures Directorate/NASA Rotorcraft Technology Transfer Meeting, March 1989, Langley Research Center.

47. Wilkie, K.: Track and Balance Studies. Presented at the Army Aerostructures Directorate/NASA Rotorcraft Technology Transfer Meeting, March 1989, Langley Research Center.

48. Yeager, W.: Rotor Aeroelasticity Research Overview. Presented at the Army Aerostructures Directorate/NASA Rotorcraft Technology Transfer Meeting, March 1989, Langley Research Center.

49. Yeager, W.: Rotor systems/Growth Blackhawk/Tailored Bearingless Rotor. Presented at the Army Aerostructures Directorate/NASA Rotorcraft Technology Transfer Meeting, March 1989, Langley Research Center.

50. Yeager, W. T., Jr.: Rotorcraft Aeroelasticity Overview. Presented to visitors from ONERA and the Fluid Mechanics Institute of Marseille, May 1989, Langley Research Center.

51. Young, M. I.: On Passive Spot Damping Anomalies. Presented at the Damping '89 Conference sponsored by the Air Force Flight Dynamics Laboratory, February 1989, West Palm Beach, FL.

Contractor Reports:

52. Cronkhite, J. D.; Dompka, R. V.; Rogers, J. P.; Corrigan, J. C.; Perry, K. S.; and Sadler, S. G.: Coupled Rotor/Fuselage Dynamic Analysis of the AH-1G Helicopter and Correlation with Flight Vibrations Data. NASA Contractor Report 181723, January 1989.

53. DiTaranto, R. A. and Sankewitsch, V.: Calculation of Flight Vibration Levels of the AH-1G Helicopter and Correlation With Existing Flight Vibration Measurements. NASA Contractor Report 181923, November 1989.

54. Dompka, R. V.: Investigation of Difficult Component Effects on Finite Element Model Vibration Prediction for the Bell AH-1G Helicopter. NASA Contractor Report 181916, Vol. 1 and Vol. II, October 1989.

55. Dompka, R. V.; Hashish, E.; and Smith, M. R.: Ground Vibration Test Comparisons of a NASTRAN Finite Element Model of the Bell ACAP Helicopter Airframe. NASA Contractor Report 181775, May 1989.

56. Dompka, R. V.; Sciascia, M. C.; Lindsay, D. R.; and Chung, Y. T.: Plan, Formulate, and Discuss a NASTRAN Finite Element Vibrations Model of the Bell ACAP Helicopter Airframe. NASA Contractor Report 181774, May 1989.

57. Gabel, R.; Lang, P. F.; Smith, L. A.; and Reed, D. A.: Plan, Formulate, Discuss and Correlate a NASTRAN Finite Element Vibrations Model of the Boeing Model 360 Helicopter Airframe. NASA CR 181787, April 1989.

58. Gold, Ronald R. and Reed, Wilmer H., III: Design of Thermoelastic Experiments Applicable to the National Aero-Space Plane. NASP Contractor Report 1056, August 1989.

59. Reed, D. A.; and Gabel, R.: Ground Shake Test of the Boeing Model 360 Helicopter Airframe. NASA CR 181766, March 1989.

60. Spain, C. V.; Soistmann, D. L.; and Linville, T. W.: Integration of Thermal Effects Into Finite Element Aerothermoelastic Analysis With Illustrative Results. NASP CR 1059. Also presented at the 6th NASP Symposium, April 1989, Monterey, CA.

Unsteady Aerodynamics Branch

Journal Publications:

61. Batina, J. T.: Unsteady Transonic Algorithm Improvements for Realistic Aircraft Applications. Journal of Aircraft, Volume 26, No. 2, February 1989, p. 131-139.

62. Batina, J. T.: Unsteady Transonic Small Disturbance Theory Including Entropy and Vorticity Effects. Journal of Aircraft, Volume 26, No. 6, June 1989, p. 531-538.

63. Batina, J. T.; Seidel, D. A.; Bland, S. R.; and Bennett, R. M.: Unsteady Transonic Flow Calculations for Realistic Aircraft Configurations. Journal of Aircraft, Volume 26, No. 1, January 1989, p. 21-28.

64. Bennett, R. M.; Batina, J. T.; and Cunningham, H. J.: Wing-Flutter Calculations with the CAP-TSD Unsteady Transonic Small-Disturbance Program. Journal of Aircraft, Volume 26, No. 9, September 1989, p. 876-882.

65. Edwards, J. W.; Thomas, J. L.: Computational Methods for Unsteady Transonic Flows in AIAA Progress in Astronautics and Aeronautics Series- Unsteady Transonic Aerodynamics, Volume 120, 1989, p. 211-261.

66. Hess, R. W.; Seidel, D. A.; Igoe, W. B.; and Lawing, P. L.: Transonic Unsteady Pressure Measurements on a Supercritical Airfoil at High Reynolds Numbers. Journal of Aircraft, Volume 26, No. 7, July 1989, p. 605-614.

67. Seidel, D. A.; Eckstrom, C. V.; and Sandford, M. C.: Transonic Region of High Dynamic Response Encountered on an Elastic Supercritical Wing. Journal of Aircraft, Volume 26, No. 9, September 1989, p. 870-875.

Formal NASA Reports:

68. Bland, S. R. (Compiler): Transonic Unsteady Aerodynamics and Aeroelasticity - 1987. NASA CP-3022, Part 1, February 1989, 267 p.

69. Bland, S. R. (Compiler): Transonic Unsteady Aerodynamics and Aeroelasticity - 1987. NASA CP-3022, Part 2, February 1989, 385 p.

70. Gibbons, M. D.; Soistmann, D. L.; and Bennett, R. M.: Flutter Analysis of Highly Swept Delta Wings by Conventional Methods. NASA TM-101530, November 1988, 19 p.

Conference Presentations:

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FY 1990 PLANS

The FY 1990 plans for the Structural Dynamics Division are broken out by each of the branches (technical areas) and selected highlights of proposed FY 1990 milestones are presented.

Configuration Aeroelasticity Branch

Figure 70 summarizes accomplishments planned for FY 1990 selected from the Branch's broad based research program on dynamic and aeroelastic phenomena of aircraft and rotorcraft.

A large portion of this work is associated with wind-tunnel tests in the Langley TDT with companion theoretical studies. Research studies are planned for both aircraft and rotorcraft. Aircraft studies will include further flutter clearance testing of the Navy A-12 for the Department of Defense, additional flutter investigations in support of the HSCT effort, and several investigations in support of the NASP program.
Basic studies related to aircraft will include testing of a NACA 0012 rectangular wing on a flexible mount to provide flutter data and a limited amount of corresponding unsteady pressure data and an investigation into coupling of periodic aerodynamic oscillations with structural modes.

The rotorcraft aeroelasticity work will include testing of a new design hingeless rotor. Initial tests will also be conducted on ARES second generation testbeds. Research studies in rotorcraft structural dynamics are aimed at developing advanced finite element-based dynamics analysis methods for predicting airframe vibrations, with particular emphasis on developing computational procedures for application in airframe dynamics design work. In-house studies and studies by the major airframe manufacturers and universities will be pursued in such areas as airframe finite element modeling, dynamics of coupled rotor-airframe systems, and airframe structural optimization under vibration constraints. In addition, work will continue toward demonstrating the potential of extension-twist coupling for passively controlling blade twist for improved aerodynamic performance of tiltrotor aircraft.

Highlights of proposed FY 1990 research for the four technical areas of Aircraft Aeroelasticity, Aircraft Aeroelastic Validation, Rotorcraft Aeroelasticity, and Rotorcraft Structural Dynamics are shown in figures 71 through 74.

Aircraft Aeroelasticity:
- Aircraft Aeroelasticity

Aircraft Aeroelastic Validation:
- Aircraft Aeroelastic Validation

Rotorcraft Aeroelasticity:
- Rotorcraft Aeroelasticity

Rotorcraft Structural Dynamics:
- Rotorcraft Structural Dynamics

Unsteady Aerodynamics Branch

For FY 1990 there will be continuing activity in developing computational methods to solve nonlinear, unsteady fluid flow equations for application to aeroelastic analysis (figure 75). There will be continued applications of the CAP-TSD code to aeroelastic response problems in order to further define its range of accuracy. As part of this activity, the branch will continue to provide support for the CAP-TSD code via a contract programmer. In addition, flutter and aeroelastic response calculations will be performed using higher-order methods, such as solution procedures based on the Euler and Navier-Stokes equations.Correlations of Euler/Navier-Stokes numerical results with potential equation solutions and experimental data will further delineate appropriate regions for use of the various flow solvers. This year will also see key developments in the field of vortex and viscous dominated flows and their roles in aeroelastic response phenomena known to occur on high-performance aircraft. These research efforts with higher-order CFD methods will be carried out using both structured and unstructured grid flow solvers. Finally, a new initiative in FY1990 will be the development of an in-house capability to generate computational grids for use with both structured and unstructured grid CFD procedures.

Aeroservoelasticity Branch

Figure 76 lists the major tasks being pursued by the Aeroservoelasticity Branch in FY 1990. In the Analysis and Modeling area, the focus of attention involves the development of procedures for using nonlinear transonic aerodynamics represented as Volterra series for ASE applications. In addition, means of approximating aerodynamics in the time domain rather than the Laplace domain are also being investigated to improve our ASE analysis and design methodologies. Studies to evaluate the effects of temperature distributions and thermal stresses and gradients on the aeroelastic characteristics of generic hypersonic vehicles will continue. Various active control concepts which offer the potential for improving the loss of aircraft stability or performance due to thermal effects will be investigated. In the Control Law Synthesis area, the design procedures developed for obtaining low-order, MIMO, robust digital control laws will be applied in the development of a combined FSS/RMLA control scheme for the AFW wind tunnel model for FY 91 test and evaluation. The development of an integrated structure and control law design approach based on hierarchal multilevel decomposition and optimization techniques will be applied to vibration problems associated with large space structures. In addition, activities to develop aerodynamic sensitivity information, initially using a finite difference approach and then based on analytical expressions, will be developed for the hierarchial aircraft design approach. Investigations will continue in evaluating the feasibility of employing "smart structures" technology for aeroelastic and ASE application through more detailed analyses and simple wind-tunnel demonstrations. In the Applications and Validations area, the cooperative effort between LaRC and Rockwell International will continue with the design of multipoint and multifunction digital control laws for evaluation during the second series of wind tunnel tests. Activities have also been initiated to provide preliminary aeroelastic and ASE evaluations for the generic supersonic transport configuration associated with the LaRC High-Speed Aircraft Multidisciplinary Research Program. This effort will also involve the application of integrated hierarchal design procedures, the use of ASE active control design algorithms, and wind-tunnel test demonstrations using semispan, actively controlled, aeroelastic models of the flight vehicle.

Selected highlights of ongoing FY 1990 research are shown in figures 77 through 84.

Analysis and Modeling:
- Maximized Gust Load Computations for Nonlinear Systems
- Modelling of Nonlinear Aerodynamics Using the Volterra-Wiener Theory
- Investigate Time-Domain Aerodynamic Approximation Methods for Aeroservoelastic Analysis

Control Law Synthesis:
- Evaluate Smart Structures Technology for Application to Aeroelasticity
- LaRC High-Speed Airframe Integration Research
- Develop Unsteady Aerodynamic Force Sensitivity Analysis Capability

Applications and Validations:
- Multifunction Digital Active Control Law Design and Validation
- Apply Integrated Multidisciplinary Design Methods to CSI Problems

Landing and Impact Dynamics Branch

Figure 85 lists the areas of continuing activity in the landing dynamics side of the branch for FY 1990. The activities include continuation of the development of the tire modeling strategies through in-house efforts, a contract study, and several university grants. Work will continue to develop a data base on the operational behavior characteristics of radial and H-type aircraft tires. Efforts will also be focused on research with the active control landing gear for the F-106B aircraft with the goal of demonstrating the potential of the active control gear during landing and ground operations. The ALDF part of the Runway Traction Program, which began in FY 89, will continue in FY 1990 as well as the continued support for the Heavy Rains Simulation tests using the ALDF with a test wing mounted on the carriage. These experimental and analytical programs will develop the landing gear technology necessary for safe ground handling operations of tomorrow's aircraft.

Figure 86 lists the areas of continuing activity in the impact dynamics group of the branch for FY 1990. Various static and dynamic tests will be conducted on concepts of composite frames, subfloors, and energy absorbing components for potential application in composite aircraft structures. Tests and analysis will be completed of scale model studies with composite beams under impact loads which will compliment the static data already completed. Efforts will continue to update the element library in the DYCAST computer code for enhancing the composite structures analysis capability under crash loads. Various analyses of composite structures will be carried out on the branch's MicroVAX computer and the LaRC CRAY super computer complex utilizing such codes as NIKE2D, DYNA3D, PAM KRASH, and MSC/DYNA as well as DYCAST. This analysis effort will be coupled with the experimental program to provide a fundamental understanding of the physics associated with the response characteristics of composite structures subjected to crash loads. Finally, initial tests utilizing the composite aircraft fuselage components should be underway in FY 1990.

Spacecraft Dynamics Branch

During FY 1990, two major areas of research will be emphasized (figure 87). The first is the control of flexible spacecraft under the Controls Structures Interaction (CSI) Program. Focusing on ground test methods, system-level experimental studies will begin on a large model which evolves in complexity with time, the CSI Evolutionary Model. These experiments will emphasize dynamic characterization of the basic structure and control studies using a 15-thruster pneumatic-jet control system. A second structure, the 20-meter-long Mini-Mast will continue to serve as a focus for evaluating flexible spacecraft control algorithms developed by guest investigators from three universities.

The second major area of research, structural dynamics analysis and test methods, will emphasize system identification and test methods for space structures which cannot be accurately tested for dynamic behavior in Earth gravity. Because of its large size and expected flight schedule, the Space Station Freedom will be used as a focus spacecraft in the expectation that methods developed can ultimately be verified using on-orbit data. Tests of a 1/10th-size, 1/5th-frequency, hybrid-scale model of an early SSF build configuration will begin and requirements for practical on-orbit tests and data acquisition on SSF will be developed. Advanced configurations of SSF for Human Exploration missions will be studied to assess the importance of dynamic behavior on design and control. The important problem of predicting and controlling the behavior of multiple flexible bodies undergoing large maneuvers will continue to be studied and a workshop will be conducted on the use of the LATDYN computer program for analysis of such problems. System identification will continue to be a research area with experimental evaluation of algorithms which learn in repeated tests being emphasized.

CONCLUDING REMARKS

This publication documents the FY 1989 accomplishments, research and technology highlights, and FY 1990 plans for the Structural Dynamics Division.

LANGLEY RESEARCH CENTER

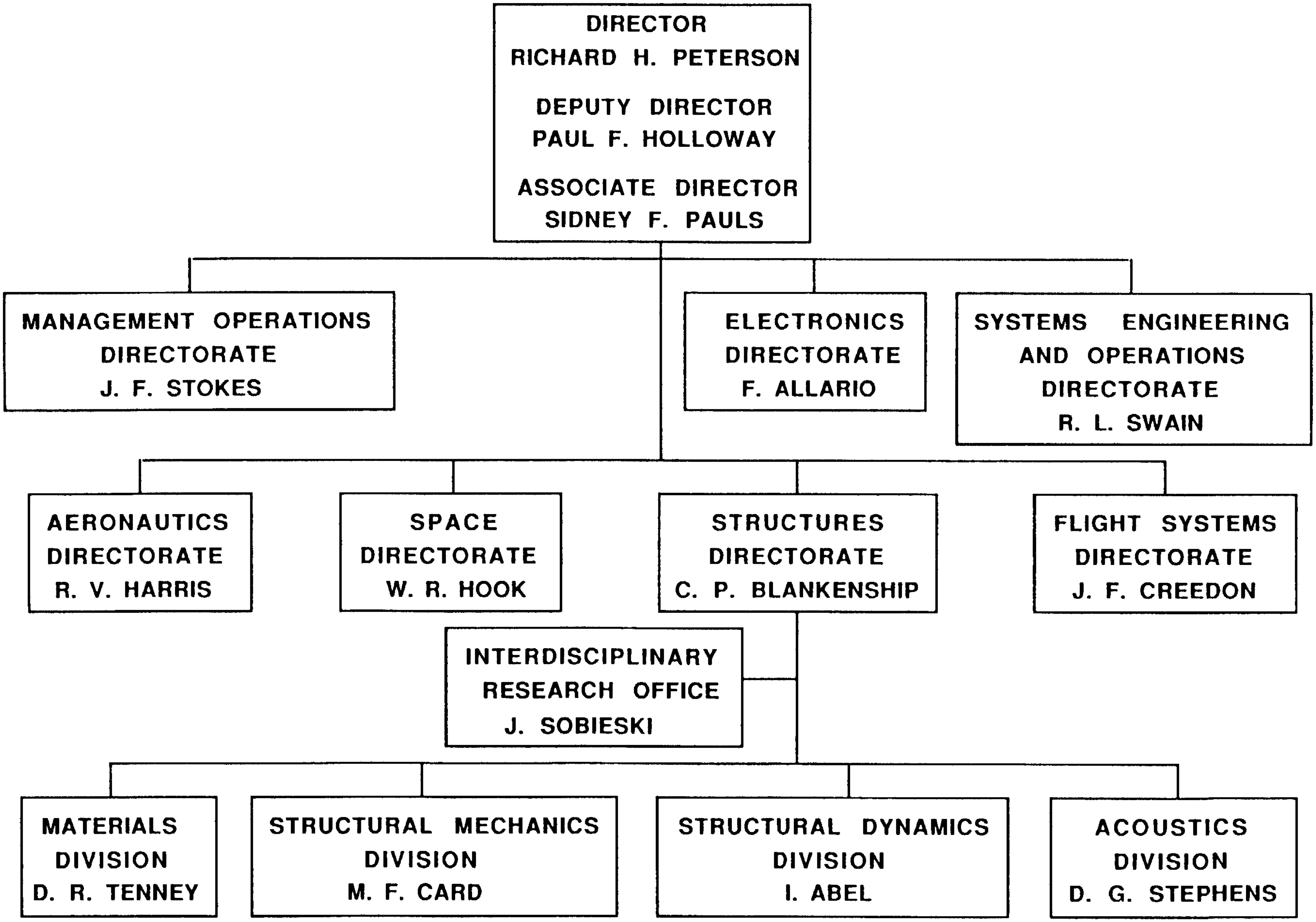

Figure 1.

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STRUCTURAL DYNAMICS DIVISION

IRVING ABEL, CHIEF
LARRY PINSON, ASSISTANT CHIEF
ROBERT DOGGETT, ASSISTANT CHIEF
JACQUELINE SMITH, ADMIN. OFFICER

CONFIGURATION AEROELASTICITY BRANCH

RODNEY RICKETTS, HEAD

- A/C AEROELASTICITY
- R/C AEROELASTICITY
- R/C STRUCTURAL DYN.

AEROSERVOELASTICITY BRANCH

THOMAS NOLL, HEAD

- ANALYSIS METHODS
- DESIGN METHODS
- APPLICATIONS/ VALIDATIONS

UNSTEADY AERODYNAMICS BRANCH

JOHN MALONE, HEAD

- THEORY DEVELOPMENT
- APPLICATIONS
- VALIDATION

SPACECRAFT DYNAMICS BRANCH

BRANTLEY HANKS, HEAD
JERROLD HOUSNER, ASST. HEAD.

- CSI GROUND TEST METHODS
- OPTIMAL SPACECRAFT PERFORMANCE
- MULTIBODY DYNAMICS

LANDING & IMPACT DYNAMICS BRANCH

JOHN TANNER, HEAD
HUEY CARDEN, ASST. HEAD

- LANDING DYNAMICS
- CRASH DYNAMICS

Figure 2.

STRUCTURAL DYNAMICS DIVISION

LONG RANGE THRUSTS

AERONAUTICS

- TRANSPORT AIRCRAFT
  - * AEROELASTICITY (LEAD)
  - * LANDING AND IMPACT DYNAMICS (LEAD)
- HIGH PERFORMANCE AIRCRAFT
  - * AEROELASTICITY (LEAD)
- ROTORCRAFT
  - * AEROELASTICITY (LEAD)

SPACE

- LARGE SPACE STRUCTURES
  - * STRUCTURAL DYNAMICS (LEAD)

Figure 3.

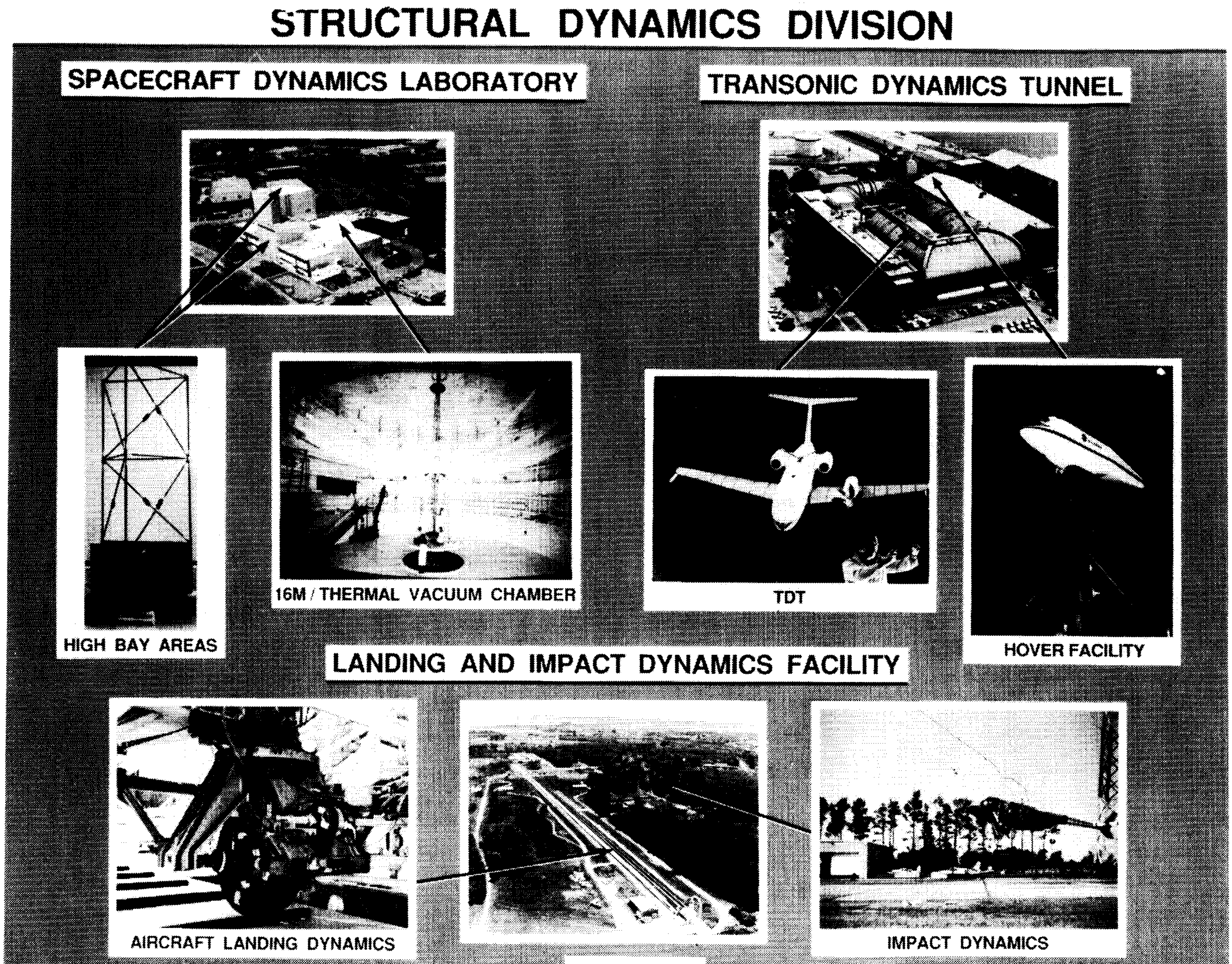

STRUCTURAL DYNAMICS DIVISION
SPACECRAFT DYNAMICS LABORATORY
TRANSONIC DYNAMICS TUNNEL
HIGH BAY AREAS
16M / THERMAL VACUUM CHAMBER
TDT
HOVER FACILITY
LANDING AND IMPACT DYNAMICS FACILITY
AIRCRAFT LANDING DYNAMICS
IMPACT DYNAMICS

Figure 4.

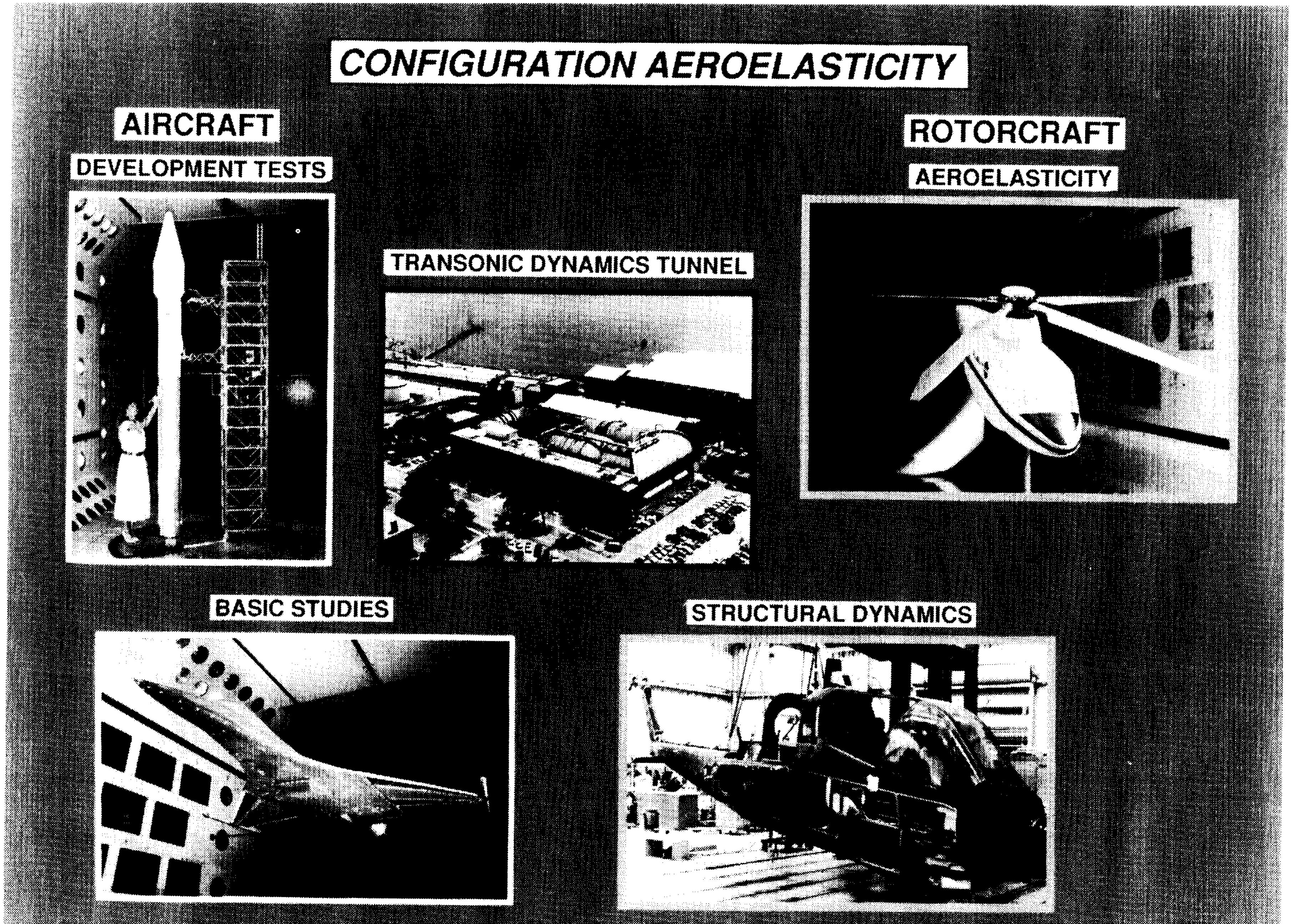

CONFIGURATION AEROELASTICITY
AIRCRAFT
DEVELOPMENT TESTS
TRANSONIC DYNAMICS TUNNEL
ROTORCRAFT
AEROELASTICITY
BASIC STUDIES
STRUCTURAL DYNAMICS

Figure 5.

CONFIGURATION AEROELASTICITY
FIVE YEAR PLAN

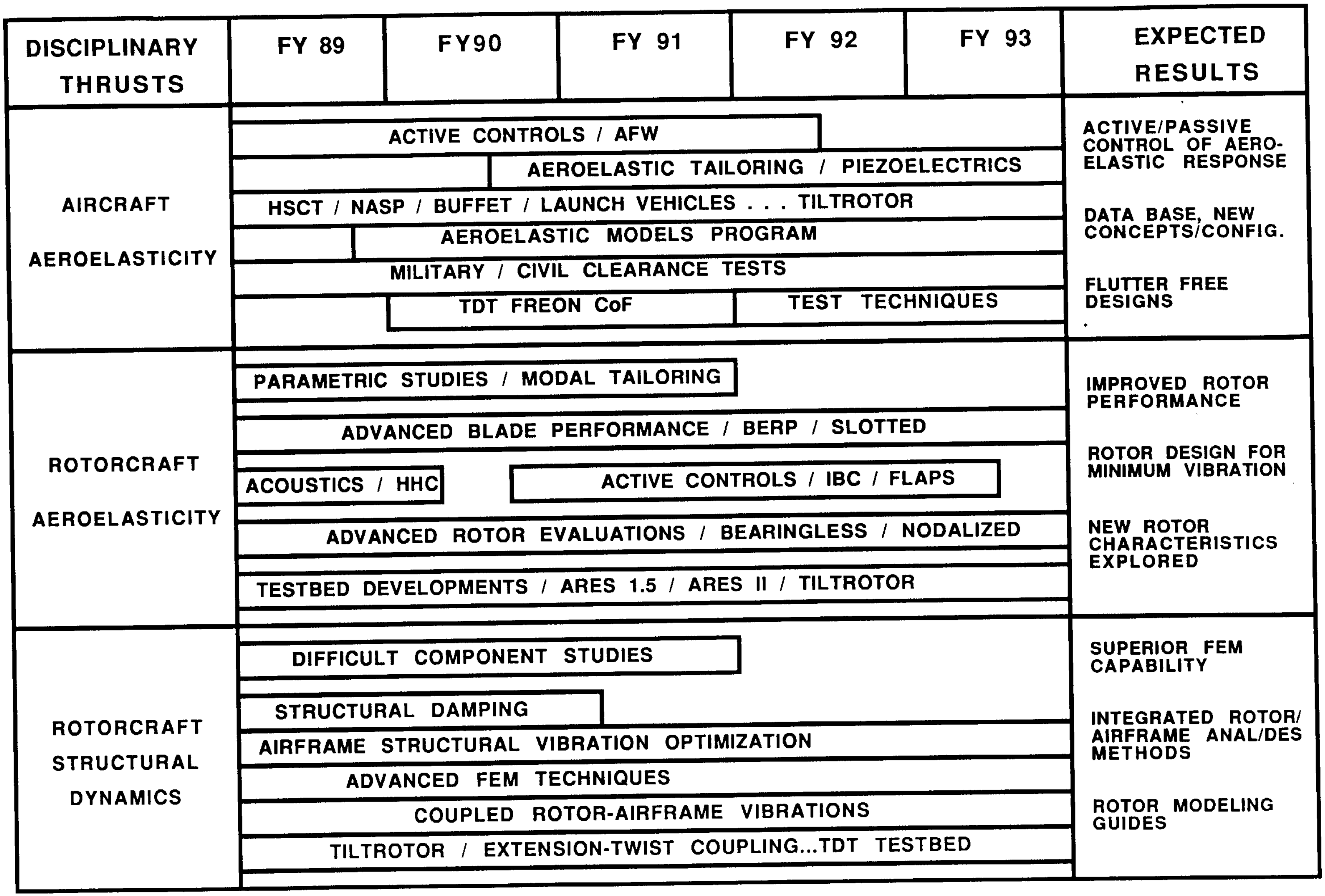


Figure 6.

EFFECTS OF PLANFORM CURVATURE ON FLUTTER OF SWEPT WING DETERMINED IN TDT

Donald F. Keller and Maynard C. Sandford
Configuration Aeroelasticity Branch

RTOP 505-63-2

Research Objective: Due to increasing interest in High Speed Civil Transports (HSCT's), NASA and several airframe manufacturers have developed new concepts and technologies to meet the configuration requirements for a Mach 3+ HSCT. One design being studied involves curving the leading and trailing edges of the outboard portion of a high speed transport wing to improve aerodynamic performance. The objective of this test was to experimentally determine the effects of planform curvature on transonic flutter of a series of moderately swept wing models and to compare these to analytical results obtained by using various unsteady aerodynamic codes.

Approach: Three generic flutter models were designed and fabricated to represent the outboard portion of a HSCT wing. Each of the three semi-span, cantilevered models consisted of a 0.188-inch-thick aluminum plate to which was bonded balsa wood that was contoured to form a 3%-thick bi-convex airfoil. The radius of curvature of the leading edges of the three models were infinity (no curvature), 200 in. and 80 in., respectively. The radius of the trailing edge for each model was determined so that the planform area of each model was 900 square inches. The length and location of the root and tip chords were the same for all three models. The leading-edge sweep angle of the uncurved wing was 53 degrees. A photograph of the moderately curved model mounted in the TDT is presented in figure 7b. Flutter tests on all three models were conducted over a Mach number range of 0.6 to 1.0.

Accomplishment Description: The flutter boundaries presented in the figure are plotted as Flutter Speed Index (FSI) versus Mach number. The FSI in a non-dimensional velocity parameter that is frequently used to correlate flutter results obtained for different models. Its value depends on flow conditions, structural stiffness, and planform geometry. The experimental boundaries were obtained using air as the test medium. The analytical boundaries were calculated using FAST, a flutter prediction program using subsonic kernel function unsteady aerodynamics. Mode shapes and natural frequencies used as input into FAST were obtained from NASTRAN finite-element models and ground vibration tests, respectively. The experimental data in the figure show that the FSI increases as the planform curvature is increased. This trend is also seen in the analytical results which agree well with the experimental data.

Significance: The results indicated that increasing the planform curvature raises the FSI of a cantilevered swept wing. The experimental results from this test form a data base that can be used in the evaluation of various analytical methods in predicting flutter boundaries for similar planform configurations.

Future plans: All experimental and analytical results obtained from this study will be published in a NASA report.

Figure 7 (a).

EFFECTS OF PLANFORM CURVATURE ON FLUTTER OF SWEPT WING DETERMINED IN TDT

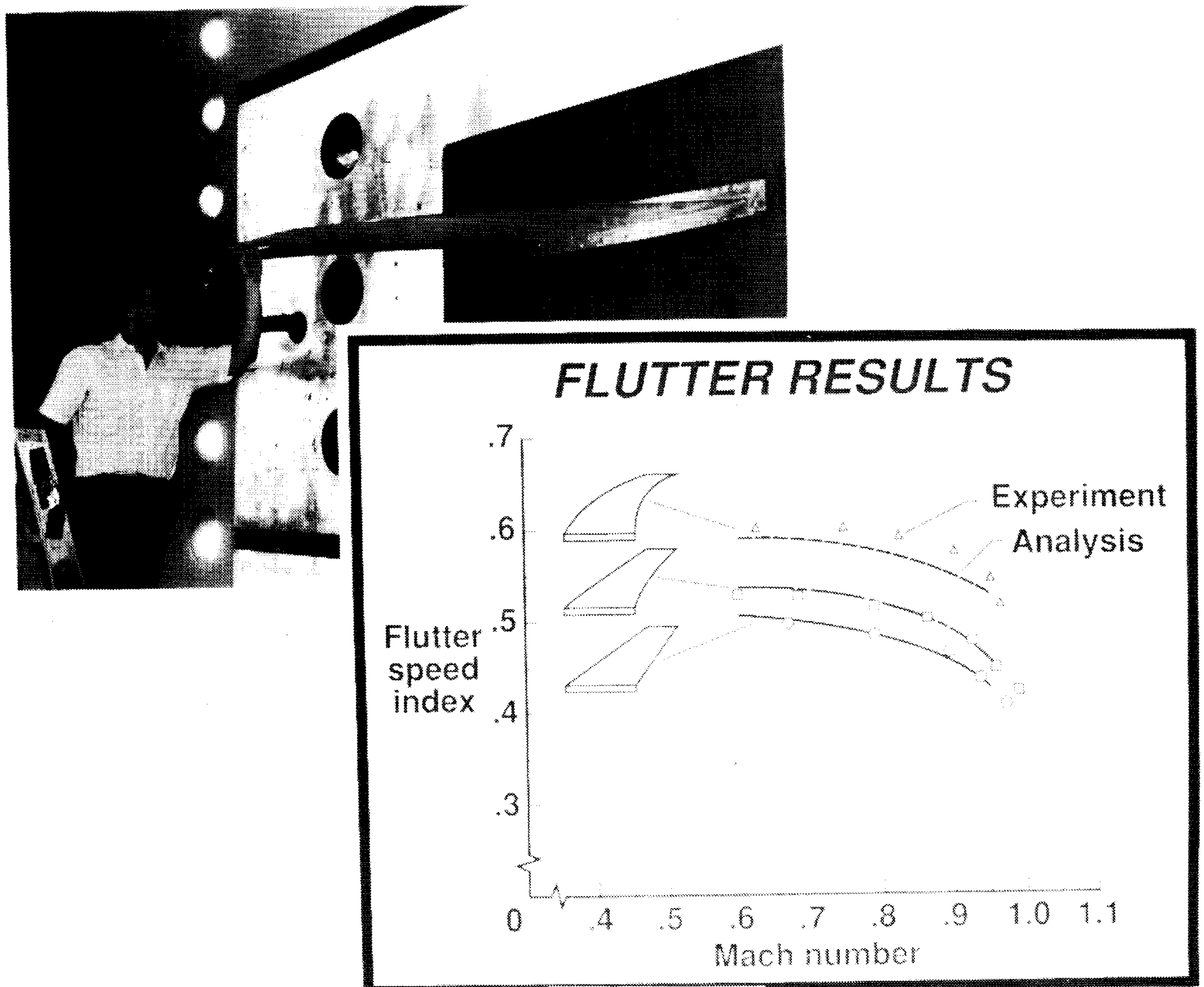


Figure 7 (b).

HIGH SPEED PORTABLE SYSTEM DEVELOPED FOR DYNAMIC DATA ACQUISITION/ANALYSIS

David C. Rosser, Jr.
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective: Experimental studies of dynamic phenomena conducted in laboratories and wind tunnels require the acquisition of digital data at high sampling rates because response frequencies of interest may be high. For facilities where dynamic testing is normally performed, suitable data acquisition and analysis systems are usually in place. In situations where dynamic testing is not routinely done, however, suitable systems are not usually available on site and must be provided by the user. The object of this effort was to develop a system to provide a high-sample-rate, multi-channel, portable, digital data acquisition, and data analysis that could be conveniently used where on-site systems are not available for dynamic testing.

Approach: Existing equipment and software that could be assembled into a personal-computer-based system were identified. The system was to provide for multi-channel capability, high digital sampling rates, ease of programing to meet varied test requirements, and near real-time data analysis, and graphic display of results. Key components identified consisted of a Macintosh II desktop computer with associated peripherals, and special interface equipment and LabView software developed by National Instruments. A photograph of some of the equipment is shown in figure 8b. The interface equipment provides for analog/digital conversion, signal multiplexing, and timing. The LabView software provides a complete integrated environment with user-interactive controls for scientific and engineering applications involving instrument control, data acquisition, data analysis, data display, data management and report generation.

Accomplishment Description: A 64-channel high-sample-rate, portable, digital data acquisition system has been developed which can be used to acquire and analyze, in near real time, dynamic response data from a variety of test situations, laboratories and wind tunnels. The system can operate at a sampling rate of 1000 samples per second each for up to 64-channels. If fewer channels of data are required, then the maximum sampling rate per channel is increased proportionally. The data acquisition/analysis system has been used to support dynamic testing of a delta-wing model in a small wind tunnel. A photograph showing the model mounted in the test section is at the lower left in the figure. A sample of the dynamic response data acquired during testing is shown at the lower right in the figure. This decay trace was printed by the Macintosh computer.

Significance: This system provides a means for acquiring and analyzing dynamic data from experimental studies in facilities that are not equipped for such testing thus providing the capability for obtaining higher quality dynamic data than was possible in the past.

Future Plans: The system is operational and ready for use as needs arise.

Figure 8 (a).

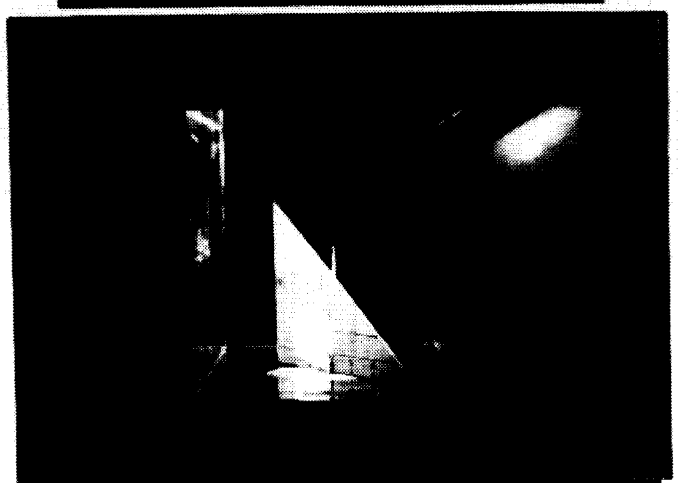
HIGH SPEED PORTABLE SYSTEM DEVELOPED FOR DYNAMIC DATA ACQUISITION ANALYSIS

ASSEMBLED SYSTEM

CHARACTERISTICS

- Portable digital system
- Laboratory/wind-tunnel use
- Near real time data display/analysis
- Multi-channel capability. 64 max
- High sample rate/channel. 64,000 divided by number of channels

WIND-TUNNEL MODEL



FREE OSCILLATORY DECAY

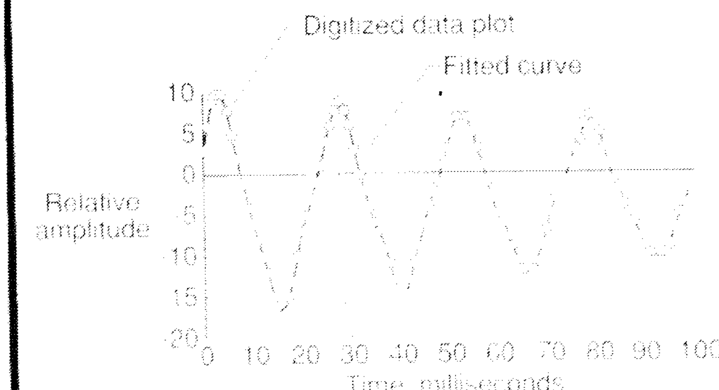


Figure 8 (b).

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ANALYSIS OF HIGHLY SWEPT DELTA WING CORRELATES WELL WITH EXPERIMENT

David L. Soistmann (Lockheed) and Michael D. Gibbons (Lockheed)
Configuration Aeroelasticity Branch and Unsteady Aerodynamics Branch

Robert V. Doggett, Jr.
Structural Dynamics Division

RTOP 506-80-31

Research Objective: Because highly swept delta wings are likely candidates to be used on hypersonic vehicles such as the NASP, there is a need to investigate dynamic instabilities which may be involved with this type of wing. The purpose of this investigation is to verify some analytical methods in predicting flutter boundaries for a highly swept delta wing. Experimental data used were obtained from a flutter test in the Transonic Dynamics Tunnel (TDT) and reported in NASP Technical Memorandum 1079.

Approach: The model which was analyzed was a 72 degree leading edge sweep delta wing with a 3% circular arc airfoil. There were two configurations of this model; each was clamped to a splitter plate in the wind tunnel. The first configuration was a delta wing with an aspect ratio of 0.65. A photograph of this model is shown in figure 9b. The second configuration was a clipped tip delta wing with an aspect ratio of 0.54. The clipped tip area is shown in the drawing in the upper right of the figure. A finite element model of the two configurations was constructed to determine the mode shapes and frequencies. Two aerodynamic methods were used to calculate generalized aerodynamic forces. The first was a kernel function code and the second was a non-linear small disturbance code (CAP-TSD).

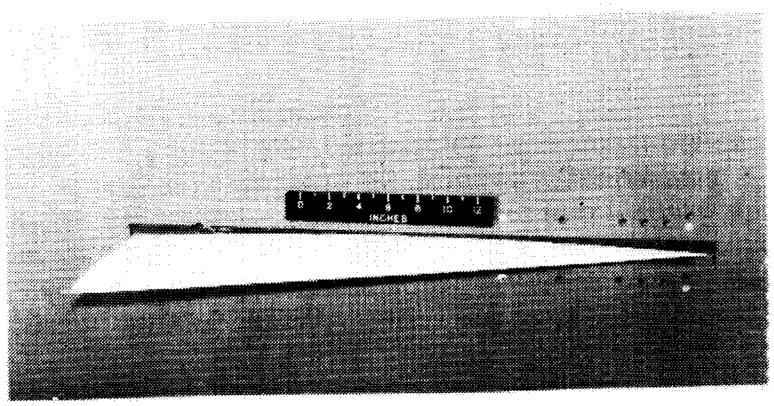
Accomplishment Description: Flutter results were obtained with the two analytical methods through the transonic range and are plotted with the experimental data. Results are shown for both configurations in the figure. For the delta wing configuration, both methods gave conservative results in the subsonic regime. In the supersonic regime, CAP-TSD gave slightly conservative results and the kernel function gave slightly unconservative results. Overall, CAP-TSD correlated better with the experimental data than the kernel function. For the clipped delta wing configuration, the kernel function results were slightly conservative, and the CAP-TSD results were unconservative in the subsonic regime. Although no supersonic experimental data were available for the clipped delta configuration, analytical flutter points were obtained with both methods and were in good agreement with each other. Referring to the flutter results for both configurations, it may be noted that the analytical methods yielded consistent results with respect to each other.

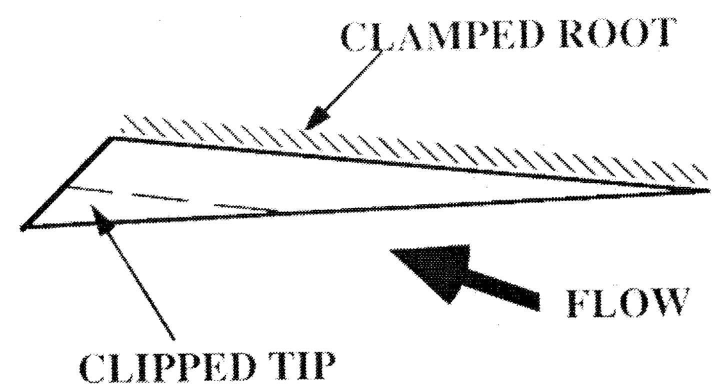
Significance: This investigation proved the usefulness of two analytical methods in determining flutter of a highly swept low aspect ratio delta wing similar to those proposed for the NASP vehicle.

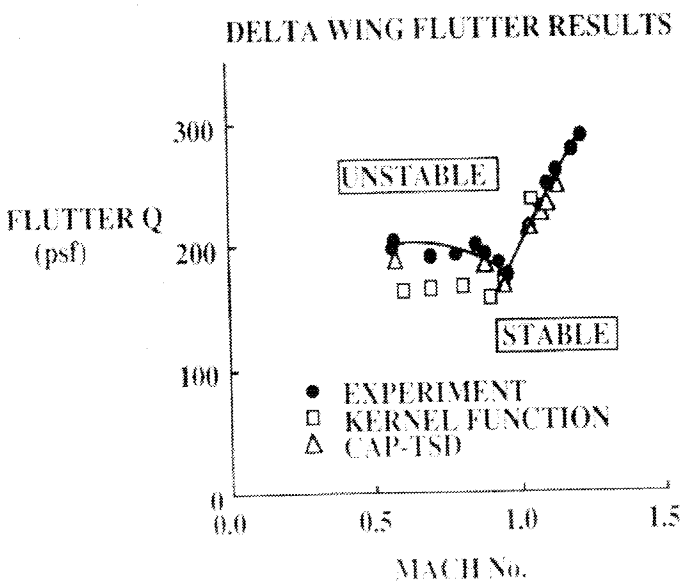
Future Plans: A full wing/fuselage generic hypersonic flutter model is currently being designed to test for the effects of large flexible fuselages on flutter. The wings for this model will be highly swept delta wings.

Figure 9 (a).

ANALYSIS OF HIGHLY SWEPT DELTA WING CORRELATES WELL WITH EXPERIMENT







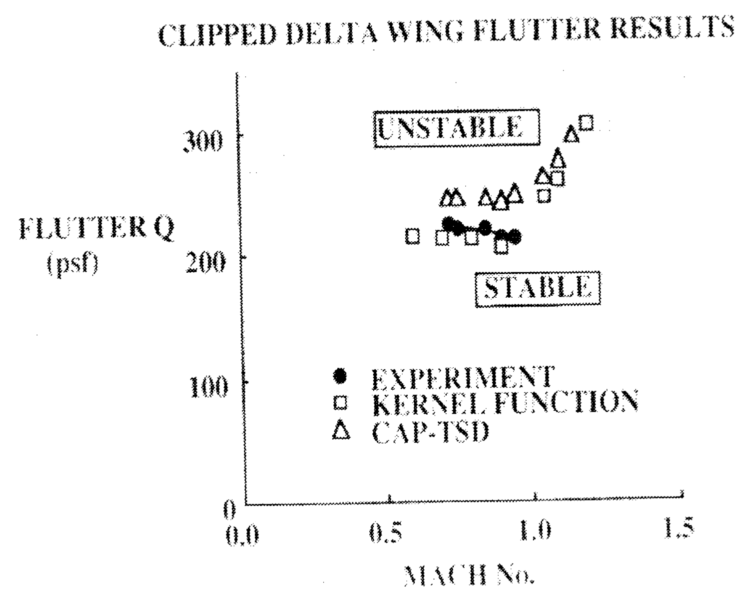


Figure 9 (b).

FLUTTER AND DIVERGENCE OF ALL-MOVEABLE DELTA WING DETERMINED

Charles V. Spain, David L. Soistmann, Ellen C. Parker, and Thomas W. Linville
Lockheed, Configuration Aeroelasticity Branch

RTOP 506-80-31

Research Objective: The National Aero-Space Plane (NASP) will employ highly swept delta (or clipped delta) wings. Some configurations will use all-moveable wings or fins to generate both lift and control forces. Because of the NASP mission and environment, these wings will be mounted on relatively flexible lifting body type fuselages. The objectives of this effort are to obtain subsonic and transonic flutter and divergence characteristics of all-moveable delta wings and to correlate the results with existing aeroelastic analysis methods.

Approach: A typical all-moveable delta wing model (photo in upper left of figure 10b) was mounted on a flexible support system with variable mass and stiffness. The flexible support system (upper right of figure) was designed to allow for variable pitch and plunge stiffnesses as well as for variations in total mass and mass imbalance. Another variable was the attachment, or pivot point, along the wing root chord. The wing model had a 40-inch root chord, and a 72-degree leading edge sweep. It was constructed of a 1/8-inch thick aluminum plate with balsa wood forming a three percent thick circular arc airfoil. The wing root was stiffened with an aluminum rib. The wind-tunnel tests were conducted in the Langley Transonic Dynamics Tunnel using air as a test medium. Analysis methods included Engineering Analysis Language (EAL) for structural modeling, a subsonic and supersonic kernel function code for unsteady aerodynamics, and an aeroelastic stability analysis flutter and divergence code.

Accomplishment Description: The tests were successful in validating present methods for flutter and divergence of this configuration. The flutter boundary curve on the figure demonstrates the correlation of theory with experimental flutter for the baseline configuration. The experiment ranged in Mach number from 0.3 to 1.05. Of interest is the transonic rise in flutter dynamic pressure and the lack of a distinct transonic dip. As shown on the lower right of the figure, divergence was also predicted reasonably well. Prior to the divergence test, analysis indicated that a pivot location of 0.6 mean aerodynamic chord would produce static divergence at a dynamic pressure less than 300 psf. This was confirmed during testing.

Significance: These tests provided experimental data for validating aeroelastic analysis methods applicable to the subsonic and transonic speed ranges for a NASP vehicle with all-moveable wings.

Future Plans: The next step in this research is to analyze and test a full span fuselage/wing model mounted on a similarly flexible support system. Scaled models of generic NASP configurations with flexible fuselages will be used. This work is being performed in coordination with the three NASP airframe contractors.

Figure 10 (a).

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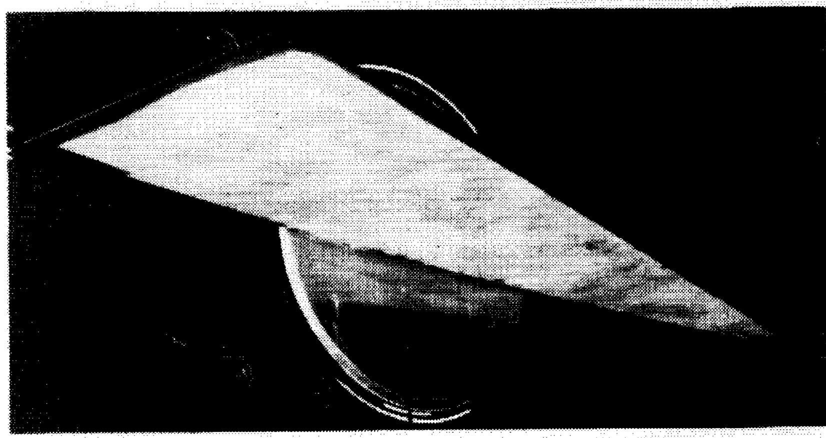
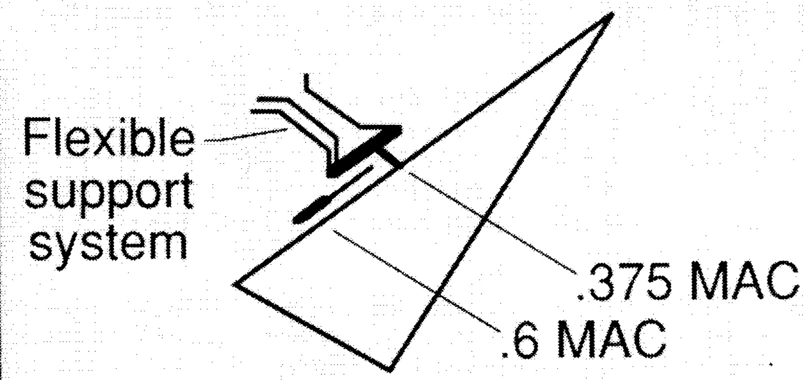
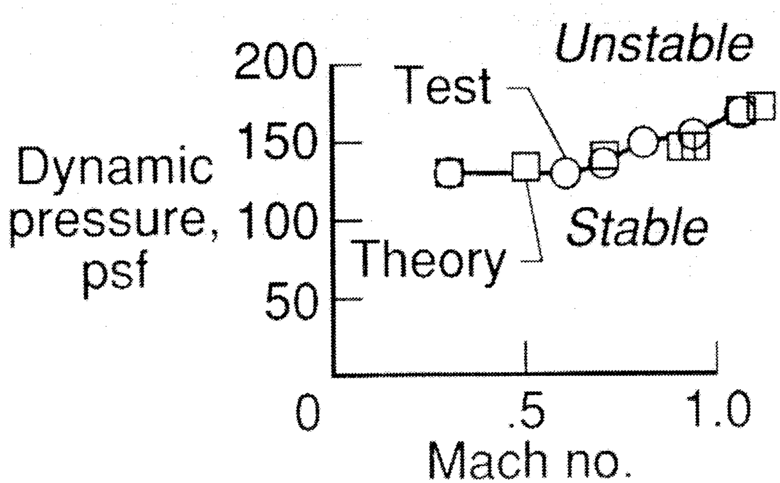
FLUTTER AND DIVERGENCE OF ALL-MOVEABLE DELTA WING DETERMINED

MODEL IN TDT





FLUTTER BOUNDARY
PIVOT AT .375 MAC



DIVERGENCE
PIVOT AT .6 MAC

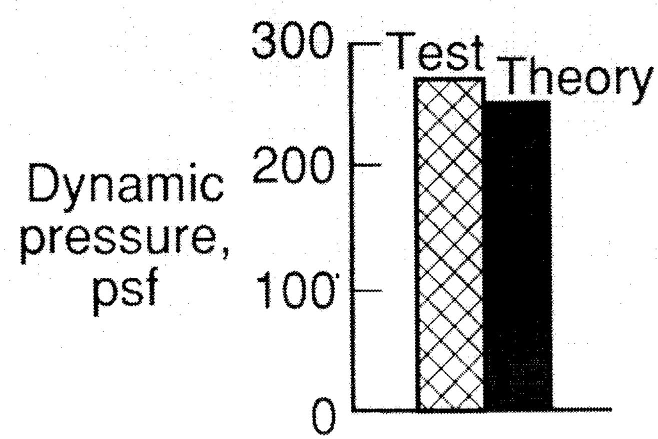


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Figure 10 (b).

AILERON-BUZZ CHARACTERISTICS FOR A GENERIC NASP MODEL ARE DETERMINED IN TDT

Ellen C. Parker (Lockheed) and Charles V. Spain (Lockheed)
Configuration Aeroelasticity Branch

RTOP 506-80-31

Research Objective: Aileron buzz is a phenomenon which can occur with trailing-edge control surfaces on an aircraft that is operating at transonic speeds. The phenomenon is usually characterized by high frequency aileron oscillations. Although aileron buzz is often a low amplitude vibration, it sometimes results in catastrophic dynamic instabilities which can lead to a loss of control of the aircraft. Because the National Aero-Space Plane (NASP) vehicle may have a full-span aileron, the objective of this study is to determine the buzz characteristics for this type of control surface on a generic NASP configuration.

Approach: The photograph in figure 11b shows the first delta-wing buzz model tested in the Transonic Dynamics Tunnel (TDT). It was made from an existing semi-span flutter model which consisted of an aluminum plate and balsa wood contoured to form the airfoil shape. A steel rib was added to increase the wing stiffness and an aileron was made by removing the aft 4 inches of the wing. The aileron was then reattached to the wing by two steel flexures located at approximately 17% and 83% of the span. There is a 0.1 inch gap between the control surface and the wing which remained constant throughout the test. A further description of the model geometry is given in the attached figure. The control surface spring-rate was varied by using three different flexure stiffnesses.

Accomplishment Description: In the subsonic range, aileron oscillations were not observed for dynamic pressures up to 200 psf (tunnel limit). In the transonic range, each of the three configurations tested exhibited aileron buzz. Aileron oscillations in this region ranged from a low amplitude, steady-state condition to an explosive, or divergent condition. The figure on the lower left indicates the tunnel conditions existing when undamped aileron oscillations were first observed (in shaded symbols) and when divergent oscillations occurred (in black symbols) for each flexure stiffness. When the aileron was exhibiting undamped oscillations at dynamic pressures exceeding 60 psf., testing was interrupted because the flexure safety limits were reached. Only for the configuration employing the stiffest flexures was a complete transonic pass possible. Although the aileron oscillations never diverged during this pass, undamped oscillations were observed with maximum amplitudes occurring at Mach 1.04 and a dynamic pressure of 40 psf (not included in plot). The wind-off and buzz frequencies, corresponding to the test conditions in the lower left figure, are indicated in the lower right figure. Aerodynamic stiffening was evident in all configurations tested with a 20% to 30% increase in aileron frequencies from a wind-off to divergence condition.

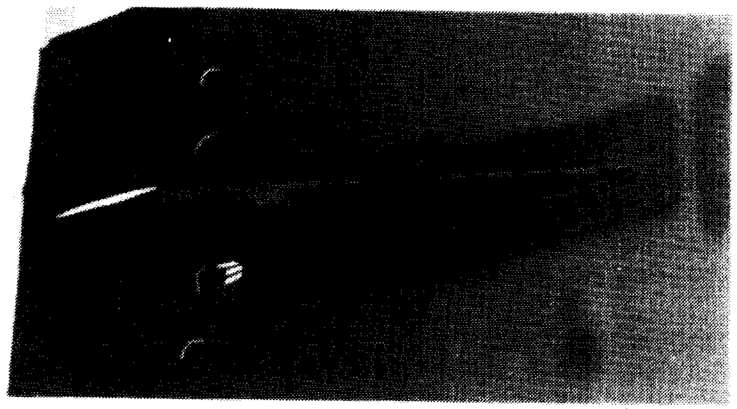
Significance: This test defined aileron-buzz characteristics for a generic NASP model. Because aileron oscillations did not occur in the subsonic range, even at higher dynamic pressures, it is probable that all oscillations observed throughout this test were shock induced. Although testing was limited to lower dynamic pressures in the transonic range, certain trends were suggested. First, increasing the aileron spring-rate may increase the Mach number at which steady-state oscillations occur. Secondly, the amplitude of the undamped oscillations (for a given aileron spring-rate) appears to be proportional to dynamic pressure. Experimental results also reveal that aileron oscillations became divergent near Mach 1 for all configurations tested.

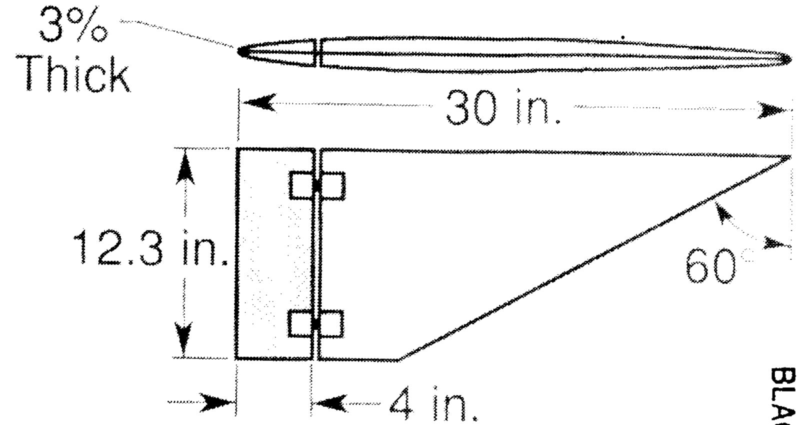
Future Plans: Testing of four composite models is currently underway. A parameter study has been initiated in order to determine what effects leading edge sweep, airfoil thickness, aileron spring-rate, aileron mass and aileron inertia have on aileron buzz characteristics. The results of this study will be published in a later report.

Figure 11 (a).

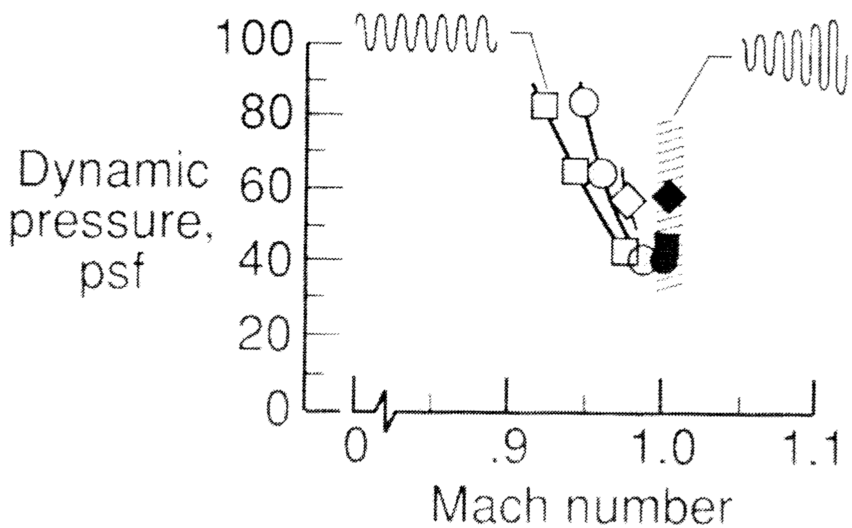
AILERON BUZZ MEASURED ON DELTA WING CONFIGURATION

MODEL IN TDT





BUZZ BOUNDARIES



BUZZ FREQUENCIES

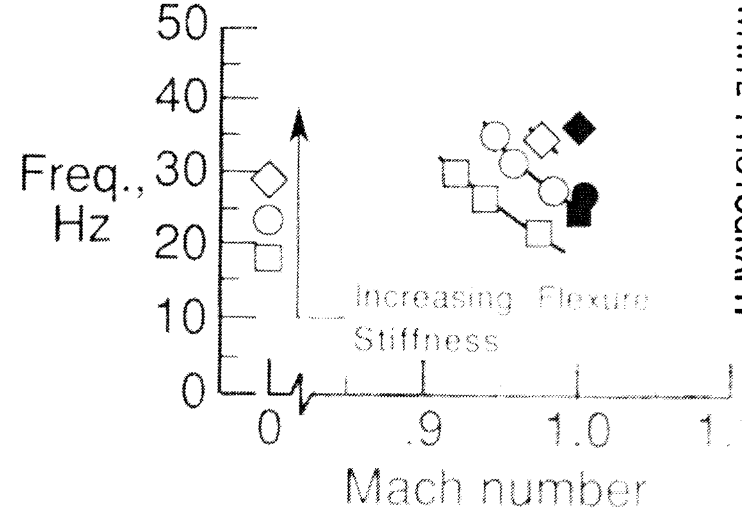


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Figure 11 (b).

OPEN- AND CLOSED-LOOP FLUTTER CHARACTERISTICS OF MODIFIED AFW MODEL DETERMINED IN TDT

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and
Boyd Perry III
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective: The Active Flexible Wing (AFW) Program is a joint NASA LaRC/Rockwell effort to design and implement multi-input/multi-output digital control of aeroelastically-influenced phenomena. The long-term goal of the project is to operate the AFW wind-tunnel model above the flutter boundary using active flutter suppression (FS) while performing rolling maneuvers in which the wing loads are actively controlled. A major objective of this first wind-tunnel test was to determine flutter boundaries and model behavior both open-loop (no FS commands sent to model) and closed-loop (utilizing active FS).

Approach: The AFW model, shown in figure 12b, required physical modifications in order to obtain wing flutter within the operating envelope of the Langley Transonic Dynamics Tunnel (TDT). The insert drawing on the accompanying figure shows a tip boom which was added to the original AFW wing tips so that flutter was attainable in the TDT. The boom utilized a hydraulic brake mechanism for the nominal "stiff" configuration and a supporting pivot and pitch spring arrangement for a decoupler-concept "soft" configuration. By decoupling the tip boom, flutter of the stiff configuration could be alleviated. The variable-stiffness pitch spring allowed model changes during the wind-tunnel test should flutter occur at undesirable conditions for the soft configuration. Three FS control laws and one rolling maneuver load alleviation (RMLA) control law were developed for this test. A future TDT test will demonstrate simultaneous FS and RMLA at conditions beyond the passive flutter boundary.

Accomplishment Description: The first wind-tunnel test has been completed. Stiffness changes to the "soft" pitch spring were found to be necessary in order for the decoupler concept to be utilized during the active FS control law work. Following model modifications, flutter boundaries were determined both open- and closed-loop for the three FS control laws. One of the FS control laws resulted in an increase of over 20 percent in the flutter dynamic pressure. The RMLA control law was tested open-loop and data was gathered which will assist in preparing for future closed-loop testing.

Significance: The modifications made to the AFW tip boom as a result of wind-tunnel findings allowed for the continuation of the program as planned. The determination of the open- and closed-loop flutter boundaries following the model modifications showed that it will be feasible in a future TDT test to attain test conditions at which FS and RMLA control laws can be tested simultaneously. Further, the demonstrated ability to test beyond the flutter boundary with one of the FS control laws indicates the program is progressing well toward this ultimate goal of multi-function control law testing.

Future Plans: Results from this test will be documented in a number of formal technical reports. Data reduction, control law maturation, and test planning will continue. The next wind-tunnel test is currently scheduled for the late summer of 1990.

Figure 12 (a).

OPEN- AND CLOSED-LOOP FLUTTER CHARACTERISTICS OF MODIFIED AFW MODEL DETERMINED IN TDT

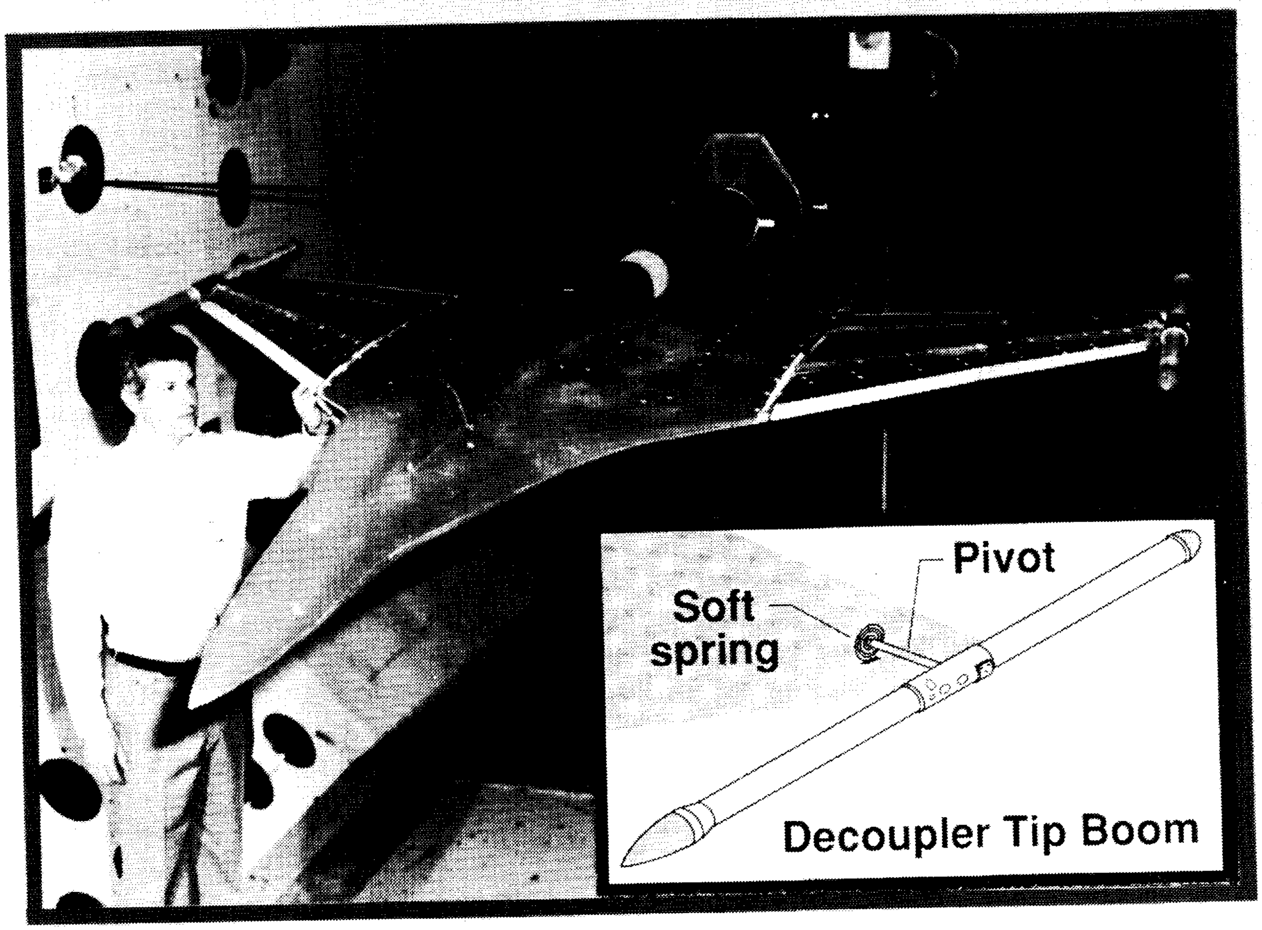

Figure 12 (b).

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FLUTTER CHARACTERISTICS OF A JOINED-WING HIGH-ALTITUDE VEHICLE DETERMINED

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RTOP 505-63-21

Objective: High-altitude vehicles have been studied for such uses as communications, weather monitoring, and military surveillance. Typically the missions require remotely-piloted long duration flights at altitudes of 80 thousand to 100 thousand feet. A unique joined-wing design has been considered for this task. The design involves two pairs of wings, one pair mounted aft and swept forward and the other mounted forward and swept aft, with tips joined to form a diamond shape from both the front and top views. This unusual configuration can provide a more efficient lightweight truss-like structure. Many applications of the joined-wing had been studied, but no aeroelastic data were available. Applying the joined-wing concept to the high-altitude mission, produces a large lightweight, structurally efficient, but flexible aircraft. Therefore, a NASA/LaRC study was conducted to determine the aeroelastic characteristics of a high-altitude vehicle utilizing this joined-wing design.

Approach: An aeroelastic model of a joined-wing high-altitude vehicle was designed and built at LaRC. A photograph of the model cable-mounted in the wind tunnel is presented in figure 13b. The model was constructed of fiberglass skins over a foam core with aluminum spars. Static calibrations were conducted to measure the stiffness, mass, and inertia properties of the model components. From these measured properties a structural finite element model was developed. Analytical vibration results were correlated with a ground vibration test of the full-span model. Using doublet-lattice unsteady aerodynamics, analytical flutter predictions were obtained. Flutter tests of the model were conducted in the Transonic Dynamics Tunnel (TDT).

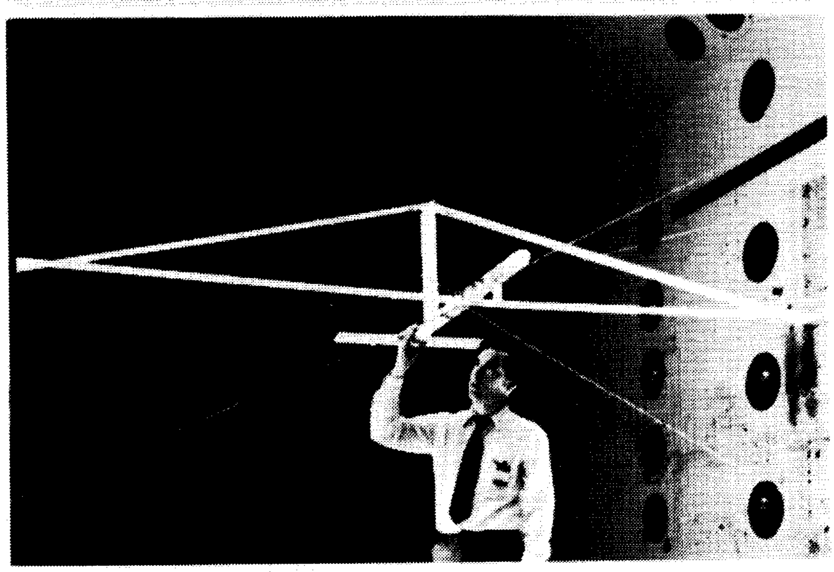
Accomplishment Description: Flutter boundaries presented in the attached figure are plotted as flutter dynamic pressure (psf) vs. Mach number. The flutter mechanism for this joined-wing design is a coupling between the symmetric wing first bending and the rigid body pitch mode. The steep subsonic boundary is a mass-ratio effect from this unusually lightweight structure. A cable interaction is shown by the effect on flutter speeds of varying rear flying cable loads. As the cable tension is increased, the experimental results are in better agreement with the analytical results which do not include cable loads.

Significance: This study provides a first insight into the aeroelastic characteristics of a joined-wing. The results show that a joined-wing can have an unusual flutter mechanism involving rigid body modes. The experimental trends of a steep subsonic boundary were predicted by linear theory.

Future Plans: These results will be presented in a NASA technical publication.

Figure 13 (a).

FLUTTER CHARACTERISTICS OF A JOINED-WING HIGH-ALTITUDE VEHICLE DETERMINED



MODEL IN TDT

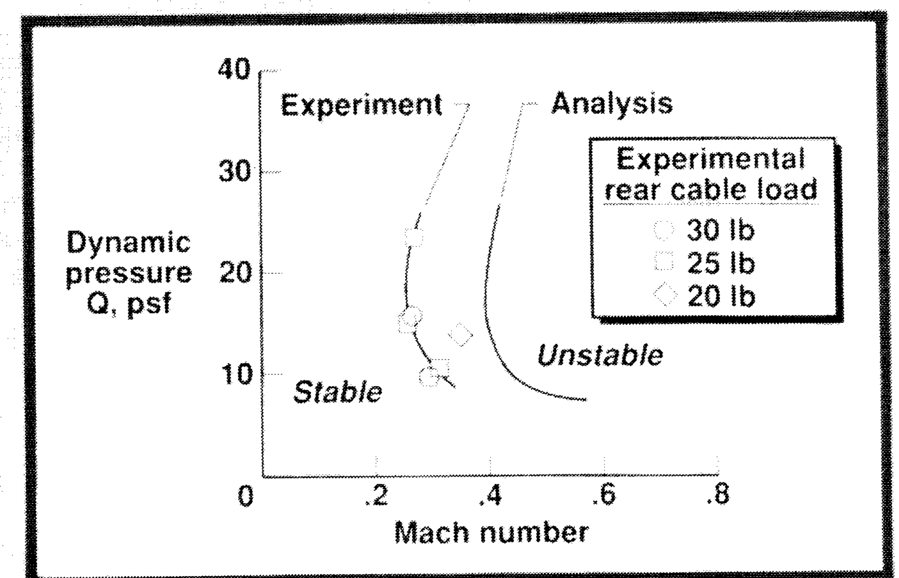


Figure 13 (b).

F-106B FLIGHT ENVELOPE FLUTTER CLEARED FOR VORTEX FLAP EXPERIMENTS

Frank W. Cazier, Jr., Michael H. Durham, and Maynard C. Sandford
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RTOP 505-63-21

Research Objective: The objective of this research was to provide flight envelope flutter clearance for the F-106B aircraft as modified with the addition of vortex flaps located on the wing leading edges in a fixed 40-degree down position.

Approach: A two-step approach was taken in establishing the flutter clearance (see figure 14b). The first step was to convert an existing finite element model of the F-106 aircraft to a NASTRAN finite element model that included the vortex flap wing leading edge extensions. NASTRAN modal results were used with a planar kernal function unsteady aerodynamic analysis program to predict the flutter boundary for the modified aircraft. The second step was to monitor onboard accelerometers using strip chart recorders and a frequency analyzer to determine frequency and response amplitude changes that would indicate modifications in damping. The flight envelope was expanded using a conservative, incremental approach requiring that the damping be greater than .03 and that flutter was not predicted at the next incremental step increase in flight speed.

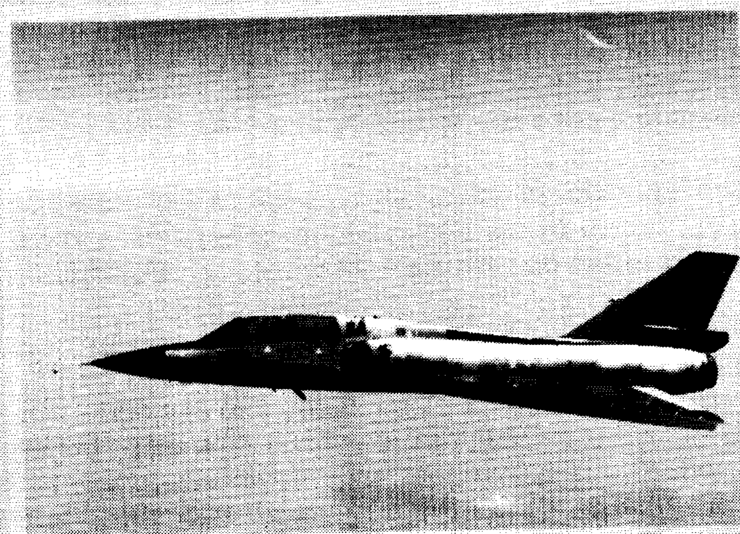
Accomplishment Description: Analysis results indicated a need to restrict flight speed to less than 310 KEAS from sea level up to 20K feet altitude. Clearance flights were performed at altitudes of 20K feet to M=0.90, at 30K feet to M=1.00, and at 40K feet to M=1.00.

Significance: Clearance to the required altitude/speed envelope was sufficient to allow the experimentors to evaluate performance of the wing leading edge vortex flap in a variety of maneuvering flight test conditions.

Future Plans: Additional flight test flutter clearance assistance will be given to the vortex flap flight test project when the F-106B resumes flight testing with a new flap setting.

Figure 14 (a).

F-106B FLIGHT ENVELOPE FLUTTER CLEARED FOR VORTEX FLAP EXPERIMENTS



WALLOPS TELEMETRY STATION

Figure 14 (b).

EFFECTS OF THREE TIP SHAPES ON WING FLUTTER DETERMINED IN TDT

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RTOP 505-63-21

Research Objective: Maximizing the aerodynamic performance and structural efficiency of new airplane designs by employing new and novel aerodynamic and structural concepts may lead to unconventional geometric configurations which could have serious aeroelastic deficiencies. Because of the important role that parametric wind-tunnel model studies have historically played in understanding flutter phenomena, particularly at transonic speeds, the present study was undertaken to examine the effects of unconventional tip geometries on wing flutter.

Approach: Six flat-plate flutter models, subdivided into two series of three models each, were tested at zero lift in the Langley Transonic Dynamics Tunnel (TDT). A photograph of one model mounted in the wind tunnel is shown in figure 15b. Sketches of the models are also shown in the figure. All six models had 60 degree-leading-edge sweep, 49 degree-trailing-edge sweep, and the same root chord length. In each series of three, there was a model with the tip chord parallel to the root chord (conventional tip aligned with the free stream velocity), a second model with the tip chord perpendicular to the mid-chord line, and a third model with the tip chord perpendicular to the root chord (perpendicular to the free stream velocity). The differences between the corresponding models in the two series was in planform area and aspect ratio. All the models in one series had a planform area of 4.35 sq. ft. The aspect ratio of these models ranged from 2.4 to 3.1, for a nominal value of 2.8. All the models in the other series had a planform area of 3.52 sq. ft. The aspect ratio of the models in this series ranged from 1.4 to 2.5, for a nominal value of 2.0.

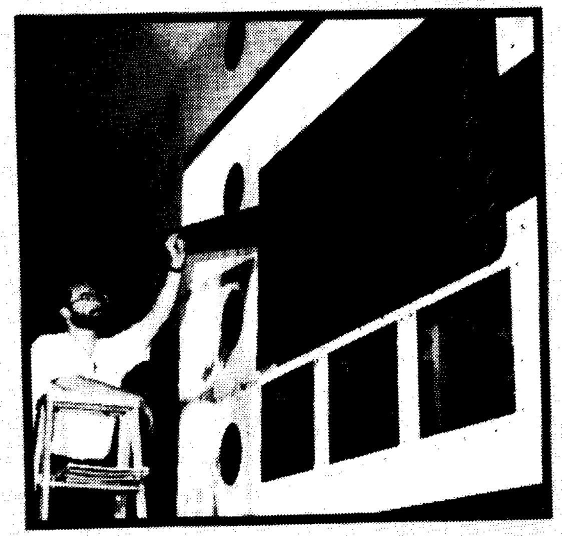
Accomplishment Description: Flutter results for the two series of models are presented in the figure as the variation of the Flutter Speed Index V_I with Mach number. The stable region, no flutter, is below the curves; the unstable region is above the curves. The parameter V_I, which is a function of dynamic pressure, wing geometry, and structural stiffness, is used by aeroelasticians to correlate flutter results for different configurations, thus providing a means for assessing parametric changes. The data in the figure for the higher aspect ratio series of models show that the flutter speed is reduced as the tip is swept from aligned to the free stream to perpendicular to free stream. The reduction in flutter speed is, however, relatively small. The data in the figure for the lower aspect ratio series of models also show that the flutter speed decreases for similar changes in tip orientation. The effect in this case is considerably larger suggesting that the effects on flutter of tip geometry increase as aspect ratio decreases.

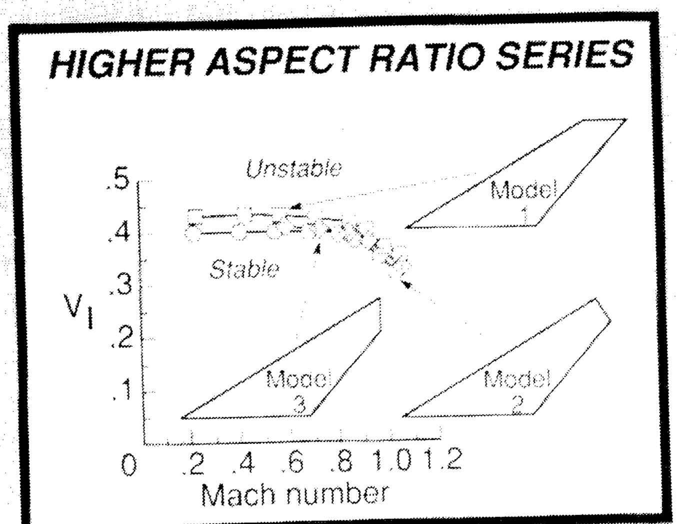
Significance: This work marks a significant addition to the parametric flutter data base that has been historically used by airplane designers to assess the flutter implications of new and novel configurations.

Future Plans: The results of this study will be documented in a formal NASA publication.

Figure 15 (a).

EFFECTS OF THREE TIP SHAPES ON WING FLUTTER DETERMINED IN TDT





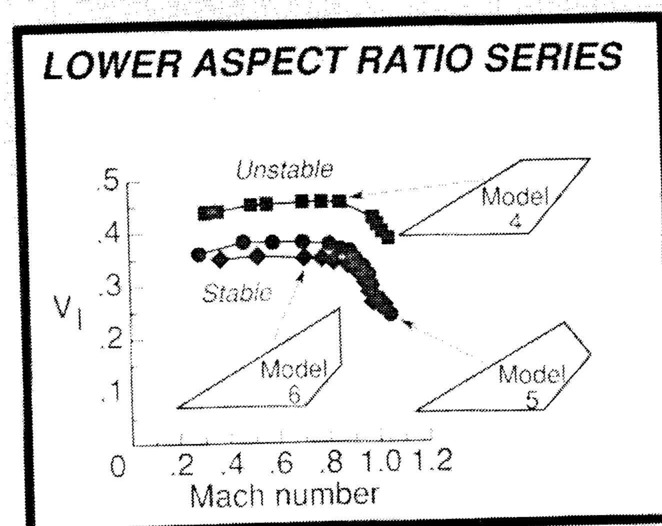


Figure 15 (b).

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TDT TESTS SHOW THAT GROUND WIND LOADS ARE NO PROBLEM FOR ATLAS II

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RTOP 505-63-21

Research Objective: The ATLAS II Expendable Booster is being developed by General Dynamics to launch both military and commercial payloads into orbit. The ATLAS II will be longer than the ATLAS I and will be able to launch heavier payloads. The objective of these tests was to demonstrate that the ATLAS II can withstand ground winds of up to 30 knots on the launch pad. The booster will be exposed to ground winds for several hours as it is fueled prior to launch. The total ground-wind loads include both steady drag loads and dynamic loads resulting from oscillatory flow around the vehicle.

Approach: A photograph of the test apparatus in the Langley Transonic Dynamics Tunnel (TDT) is shown in figure 16b. The approximately 1/12 scaled model booster and umbilical tower were bolted to a large turntable attached to the floor of the test section. The turntable could be rotated 360 degrees to simulate ground winds from any compass bearing. The booster was aeroelastically scaled to simulate the first and second bending modes of the full scale booster. During the test weights were added inside the booster to simulate four different fuel conditions. The payload fairing shown in the figure simulated a 14 foot diameter full-scale fairing that will be used for large payloads. An 11 foot diameter fairing was also simulated during the tests. The model umbilical tower, however, was not aeroelastically scaled; it was used to approximately simulate the aerodynamic effects of the tower on the vehicle. The full scale tower will have a damper that extends out to the booster to greatly suppress booster dynamic loads. This damper was simulated in the wind-tunnel tests. During the tests both steady and dynamic bending moments were measured on the booster using strain gages.

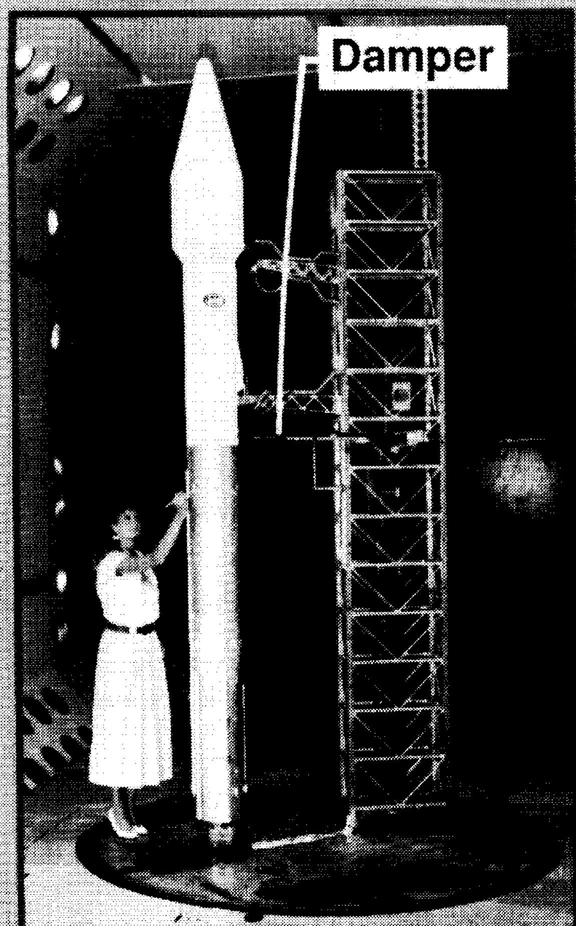
Accomplishment Description: The total steady and dynamic bending moment near the base of the model booster for one fuel condition and at a typical wind compass bearing is shown in the figure. The data have been scaled to the full scale booster. The critical limit bending moment for the full-scale booster is also shown. With the damper installed the total bending moment is almost entirely due to the steady bending moment. With the damper installed the bending moment at 30 knots is less than the critical limit. Data obtained without the damper show that the dynamic moment is so great that the critical limit would be exceeded at about 23 knots.

Significance: These tests have shown that, if final launch procedures are begun only when the weather forecast is good, launches will rarely if ever have to be scrubbed because of weather after the fueling process has begun.

Future Plans: No further work is planned.

Figure 16 (a).

TDT TESTS SHOW THAT GROUND WIND LOADS ARE NO PROBLEM FOR ATLAS II



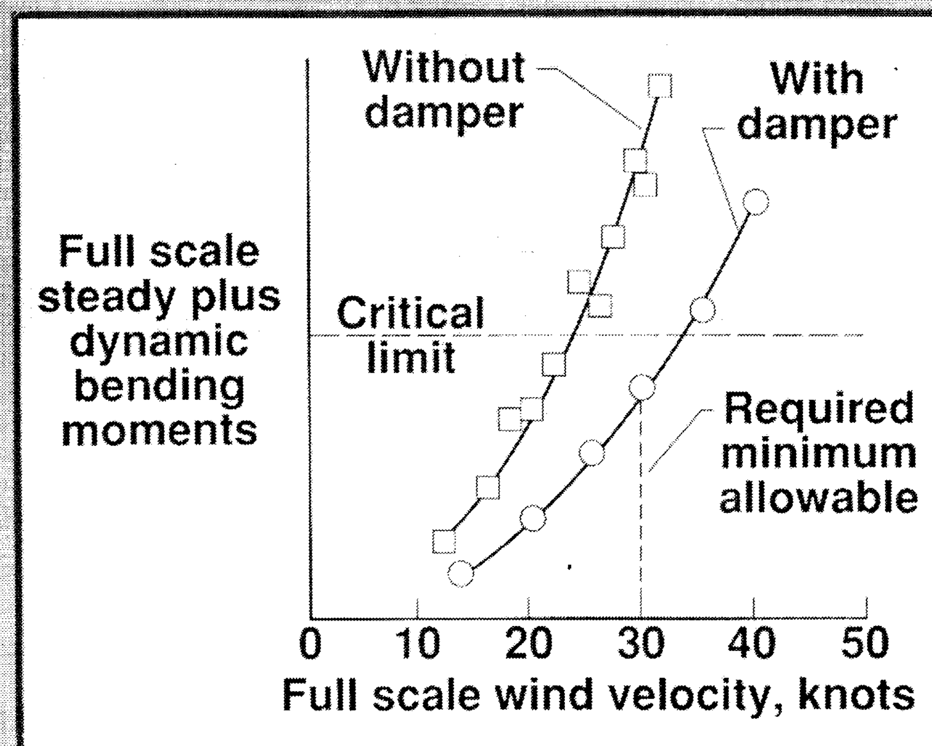


Figure 16 (b).

USE OF BLADE NON-STRUCTURAL MASS REDUCES ROTORCRAFT VIBRATIONS

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RTOP 505-63-21

Research Objective: Historically, rotorcraft has experienced substantial vibration problems. One means of reducing these vibrations is through the addition of non-structural mass to the rotor blades. These masses are intended to "tune" the blades so that the vibrations are reduced to an acceptable level. Generally, these problems have been addressed in the post-design phase by modifying the rotor systems on prototype or production aircraft. However, recent rotor blade design techniques have attempted to provide low vibration characteristics without later requiring design modifications. This design method requires the use of highly accurate analytical prediction techniques. Many of the early attempts to design low vibration rotors have been unsuccessful due to the inadequacy of these prediction methods. The objectives of this research program are to systematically investigate helicopter vibration reduction through the use of non-structural blade masses and to obtain an experimental data base for correlation, validation, and development of analytical prediction methods.

Approach: Aeroelastically scaled model rotor blade hardware was fabricated for testing on the Aeroelastic Rotor Experimental System (ARES) in the Transonic Dynamics Tunnel (TDT). These blades, as shown in figure 17b, were comprised of two major components: an airfoil glove, which had an internal channel centered about its quarter chord, and a steel spar which could be inserted in the channel. The steel spar had 13 cutouts located every 5% of span beginning at the 30% radial station. A tungsten or steel mass could be mounted in any of the cutouts on the steel spar which was then mated with the airfoil glove for the complete blade assembly. Testing was conducted in the TDT to provide a parametric study of the effects of mass radial placement on rotorcraft vibrations throughout a representative forward flight speed range.

Accomplishment Description: A sample of the results of the wind tunnel test is shown in the lower left of the figure. This plot shows 4 per revolution vertical fixed-system loads for a single tungsten mass located at several different radial locations. Reductions of up to 30% may be achieved by the appropriate selection and placement of a non-structural mass. Results were obtained for three different thrust conditions at representative propulsion forces throughout the forward flight speed range. Parameters varied included the amount and location of the added mass and the rotor trim condition.

Significance: This research effort has provided a parametric study of the effects of non-structural blade masses on rotorcraft vibration reduction. The data base developed will be significantly useful in the verification of existing rotorcraft analytical design methods and in the development of new, low vibration rotor design techniques.

Future Plans: A report will be released documenting the results of the wind tunnel test. The results will also be used in a correlation effort with CAMRAD, a government owned and developed rotorcraft analysis.

Figure 17 (a).

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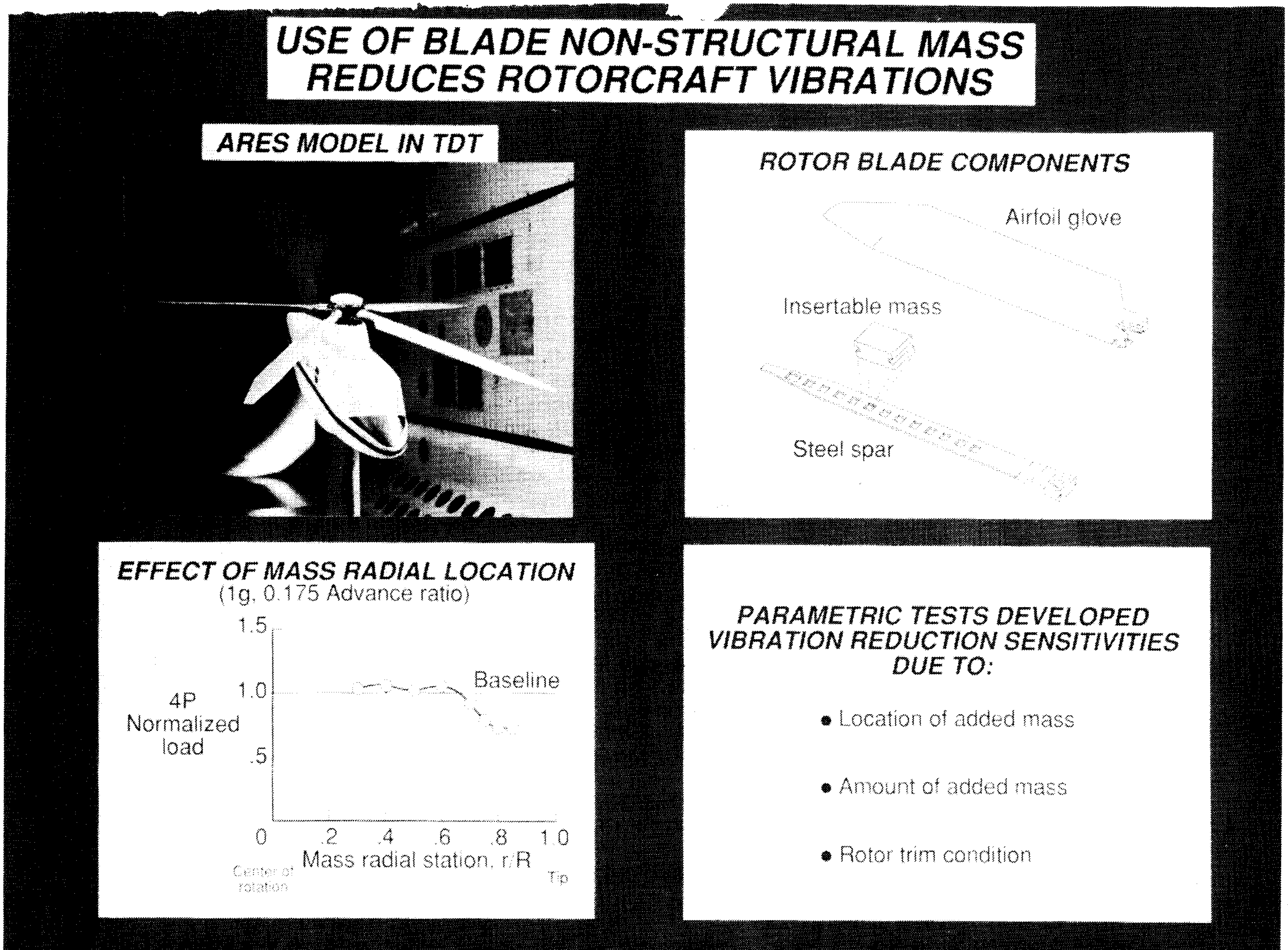

Figure 17 (b).

BVI NOISE REDUCED USING HIGHER-HARMONIC PITCH CONTROL

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Configuration Aeroelasticity Branch, SDyD

RTOP 505-63-51

Research Objective: Blade-vortex interaction (BVI) is the source of one of the most objectionable types of helicopter noise. The objective is to evaluate the use of higher-harmonic pitch (HHP) control of rotor blades to reduce BVI noise.

Approach: Impulsive BVI noise, due to blade interaction with shed vortices of preceding blades, has been a major topic of rotorcraft acoustics research for several years. One noise reduction concept is that decreases in blade lift and/or vortex strength at the vortex encounters should reduce the intensity of the interactions and thus the noise. This idea is illustrated in figure 18b showing the rotor blades undergoing a specific 4-per-rev HHP control mode. To evaluate the concept, a unique interdisciplinary study was conducted of rotor acoustics and vibration. The figure shows the Aeroelastic Rotor Experimental System (ARES) in the Transonic Dynamics Tunnel (TDT). Specially calibrated microphones are shown positioned upstream & downstream of the model. A unique sound power measurement technique was developed for the test because of TDT reverberance & Freon environment.

Accomplishment: The ARES model had a fully articulated hub design and used a four-bladed, 110 inch diameter rotor with untwisted NACA 0012 blades. Using a specially developed open-loop control system, prescribed HHP control modes were superimposed on the basic cyclic trim pitch for a broad range of rotor operating conditions. Rotor operation included (1) the normal one-per-rev (1-p) cyclic control for zero flapping flight trim, (2) a 4-p collective pitch control for the same trim conditions, and (3) a special pitch control to simulate individual blade control, again for the same trim condition. It was found that HHP could significantly increase or decrease the noise depending on the amplitude and azimuth for the pitch control. A 4-per-rev HHP of -1° amplitude at 60° azimuth was found to be of particular interest. The figure shows spatially averaged noise levels which were band pass filtered to emphasize the impulsive BVI noise. Noise levels shown are roughly equivalent to dB_A annoyance levels, offset by a constant. Helicopter descent angle was calculated using model test conditions and typical full-scale fuselage drag data. Advance ratio is the forward speed divided by rotor tip speed. For normal flight conditions without HHP, descent angles between 4° to 10° are seen to produce high BVI noise levels, especially at low speeds. Using HHP, noise levels for matched flight conditions are significantly altered, with the highest BVI noise levels being reduced 4.7 dB.

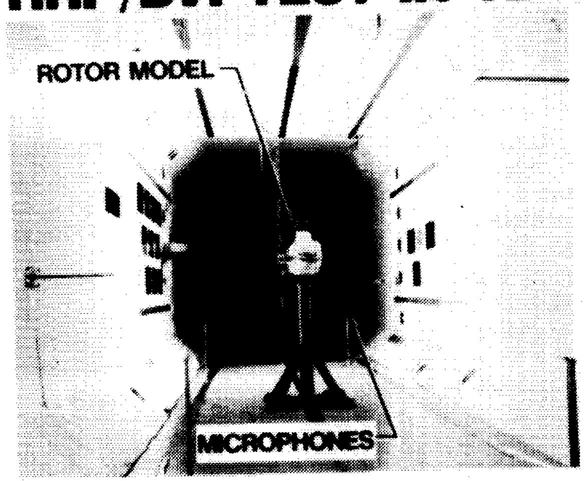
Significance: The results demonstrate HHP to be an effective noise reduction technique for descending flight where BVI noise dominates. Selective use of HHP, such as during landing approach, for full scale helicopters should prove practical. Its use could be a specific application of higher-harmonic control capability, which could otherwise be used for vibration reduction during cruise.

Future Plans: Detailed data processing has begun which will produce a complete parametric mapping for this rotor of the effect of HHP on BVI noise. Corresponding blade loading and vibration data for matched cases will provide the key to evaluate the practicality of particular pitch controls for noise reduction. The results will be compared to predictions from a newly modified version of the CAMRAD rotor performance program and from a HHP/BVI noise prediction program currently under development.

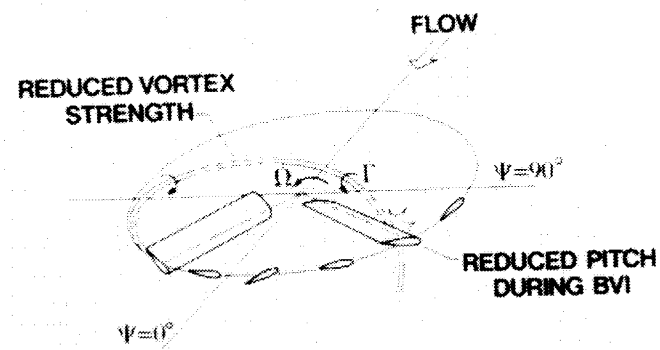
Figure 18 (a).

BVI NOISE REDUCED USING HIGHER-HARMONIC PITCH CONTROL

HHP/BVI TEST IN TDT



HHP CONCEPT



NOISE LEVEL (dB) WITH HELICOPTER FLIGHT CONDITION

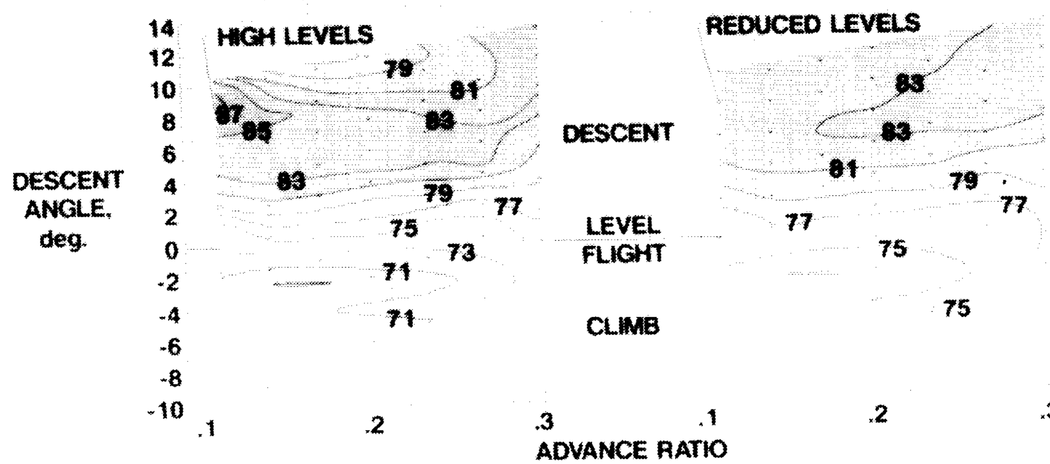


Figure 18 (b).

SIGNIFICANCE OF WARPING DETERMINED ANALYTICALLY FOR COMPOSITE ROTOR BLADES WITH EXTENSION-TWIST COUPLING

Mark W. Nixon
Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective: The primary load carrying members in composite rotor blades are configured as thin-walled beams. When loaded, the overall bending and twisting deformations are usually accompanied by additional deformations which are associated with the warping of the cross sections. Typical practice in preliminary design work is to use analyses based on classical beam theory which does not include the effects of warping on the bending and twisting behavior. For composite blades which incorporate extension-twist coupling through an off-axis arrangement of the plies in the laminates which comprise the walls of the beam, the effects of warping may not be negligible. The object of this research was to determine the significance of warping on the predicted torsional and bending behavior by comparing deformations computed using a classical analysis which neglects warping & more complete analysis which includes all warping effects.

Approach: As illustrated in figure 19b, there are two types of warping. The warping associated with torsion of a beam involves displacements normal to the plane of the cross section; the warping associated with bending involves distortions of the cross sections in their own planes. The analysis of torsion is usually based on a St. Venant approach which assumes that cross sections are free to warp out of their plane. However, rotor blades are constrained so that they are not completely free to warp, and this effect is not accounted for in a St.Venant solution. In the case of bending, classical beam analyses take no account of warping effects. The errors associated with neglecting the effects of warping on both the bending and torsional deformations of a composite beam with extension-twist structural coupling were determined by comparing the results calculated using classical beam theories with results calculated using a finite element model consisting of 2-D shell elements which includes all warping effects.

Accomplishment Description: The error associated with not accounting fully for the effects of torsion-related warping is indicated in the graph in the upper right of the figure where the relative twist at the tip of a cantilevered beam is shown as a function of a warping parameter (a function of length, torsional stiffness, and warping stiffness). The reference twist is that given by the classical St.Venant torsion solution. It is seen that the error is significant only when the warping parameter (read length) is small. Similar results were obtained when the ply angle was varied. Thus the effects of warping on torsion are strongly dependent on the length of the beam but essentially independent of ply angle (i.e., extension-twist coupling). Because rotor blades typically have warping parameters greater than 100, warping effects are not significant for such structures. The error associated with neglecting bending-related warping is indicated in the graph in the lower right of the figure which shows the relative displacement at the tip of the beam as a function of the material off-axis ply angle. The reference tip deflection is from an analysis based on classical beam theory. The ply angle is an indicator of the amount of extension-twist coupling in the laminates comprising the walls of the beam. For off-axis angles of 0 and 45 degrees there is no extension-twist coupling, while an angle of about 22.5 degrees yields the maximum amount of coupling. The results indicate that the effects of warping are significant over the same range of ply angles which would be employed in a blade design intended to take advantage of extension-twist coupling. Similar results were obtained when the length of the beam was varied. Thus, the effects of warping on bending behavior are strongly dependent on the ply lay-up angle but essentially independent of length.

Significance: The work shows that classical beam theories must be used with caution in the analysis of composite rotor blades with extension-twist coupling. In particular, classical theory is adequate for torsion but is wholly inadequate for bending where the effects of warping are critical to the prediction of overall bending behavior.

Future Plans: As a result of this investigation, attention will be focused on developing a hybrid method of analysis which combines the accuracy of a 2-D analysis to define the warping behavior of a cross section with the simplicity of a 1-D beam analysis to define the overall bending & torsional behavior.

Figure 19 (a).

SIGNIFICANCE OF WARPING DETERMINED ANALYTICALLY FOR COMPOSITE ROTOR BLADES WITH EXTENSION-TWIST COUPLING

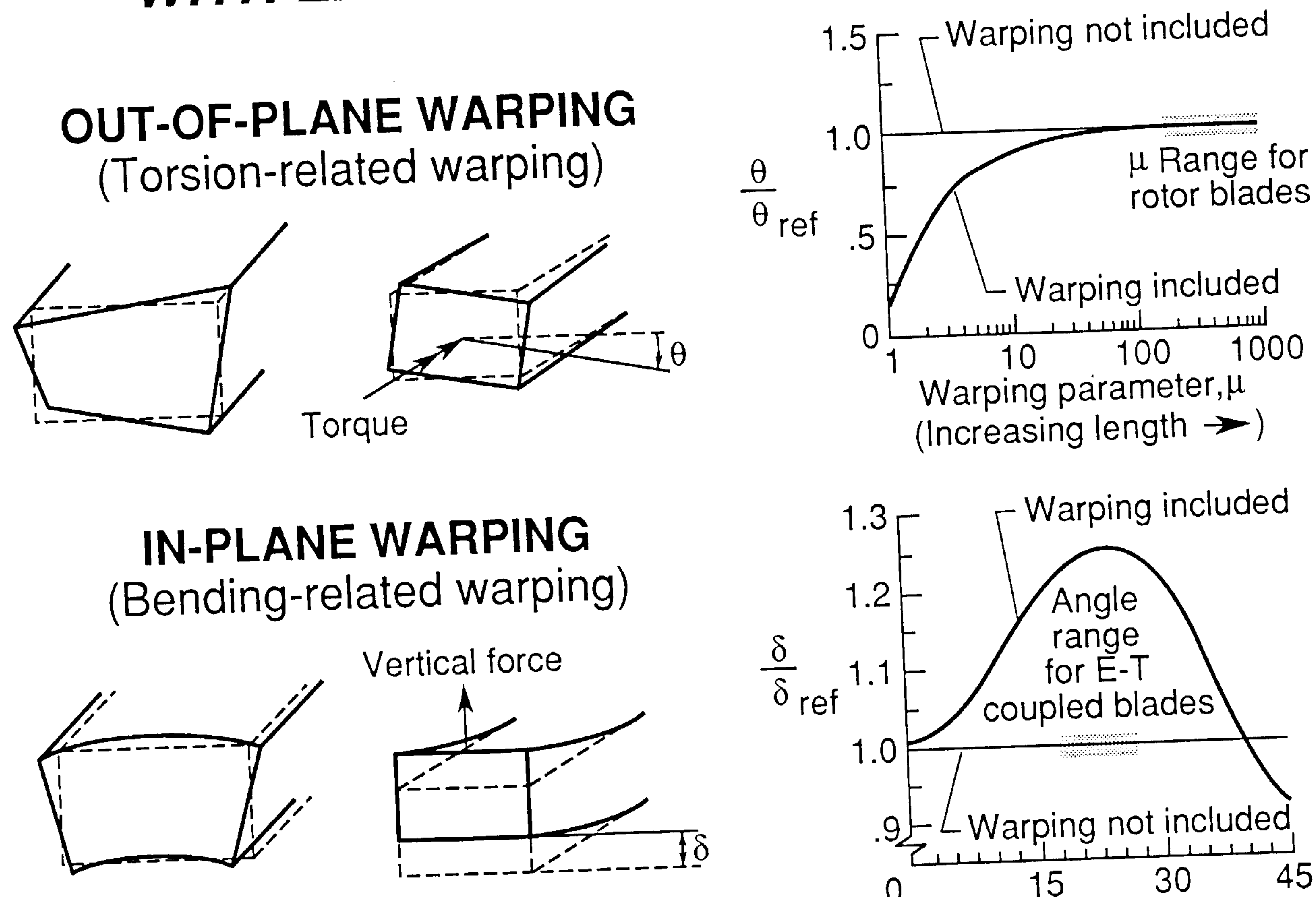


Figure 19 (b).

P-VERSION TAPERED BEAM FINITE ELEMENT DEVELOPED TO IMPROVE VIBRATIONS ANALYSES

Howard E. Hinnant
Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective: The reduction of rotorcraft vibrations is one of the most pressing and difficult problems facing the structural dynamicist today. To have a significant impact on reducing vibration, without attendent weight or performance penalties, it is becoming increasingly important for vibration reduction to be considered during design. However, current finite element based dynamic analyses are not accurate enough to be used with confidence during design. The objective of this research effort is to develop finite element dynamic analysis methodologies which are more accurate, easier to use, and computationally more efficient.

Approach: Although there are several possible reasons for the lack of correlation with experiment, one that has recently attracted attention is the accuracy of the elements themselves. Most finite element codes in use today employ what are commonly referred to as h-version elements. An h-version element uses fixed-order shape functions to relate the discrete nodal displacements of the element to the continuous displacements within the element. The p-version element is different from the h-version element in that the user may select the order of the shape functions defining the displacement behavior of the element. Increasing the order of the shape functions is analogous to increasing the number of elements for a comparable h-version model.

Accomplishment Description: A p-version tapered beam finite element has been derived. Three finite element models of a truncated, conical beam were then formed: one using a single p-version tapered beam element, the second using h-version tapered beam elements, and the third using h-version uniform beam elements. Convergence studies were then performed by varying the number of elements used in the h-version models and by increasing the order of the shape functions in the p-version model. As shown on the right side of figure 20b, the model using p-version tapered beam element produces more accurate frequencies for a given number of degrees of freedom than the models using tapered h-version elements or uniform h-version elements. This is a general indication of the computational efficiency of the finite element. The most valuable characteristic of p-version elements is the relative ease of increasing the accuracy of the finite element model. This is difficult with large h-version models because of the need to generate a new, refined mesh. With p-version finite elements, however, the existing mesh doesn't need to be altered. Instead the order of the shape functions used for the elements is increased, a relatively trivial task.

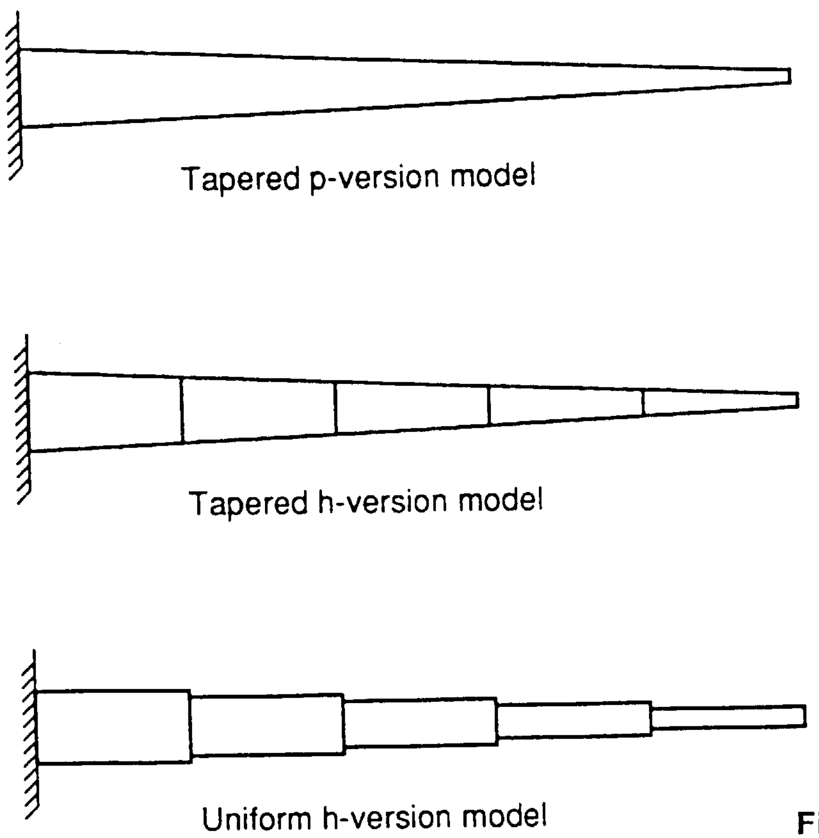
Significance: The development of an efficient, well-behaved, p-version tapered beam element is an important demonstration of the type of technology that might be used to improve the effectiveness of current finite element dynamic analyses.

Future Plans: A small, research oriented, finite element code is under development in which the beam element will be installed. A p-version plate-type element will be implemented in the code as well. With these tools, a series of successively more complex analysis/experiment correlations will be performed.

Figure 20 (a).

P-VERSION TAPERED BEAM FINITE ELEMENT DEVELOPED TO IMPROVE VIBRATIONS ANALYSES

- Simplifies Convergence Checks
- Superior Convergence Rate



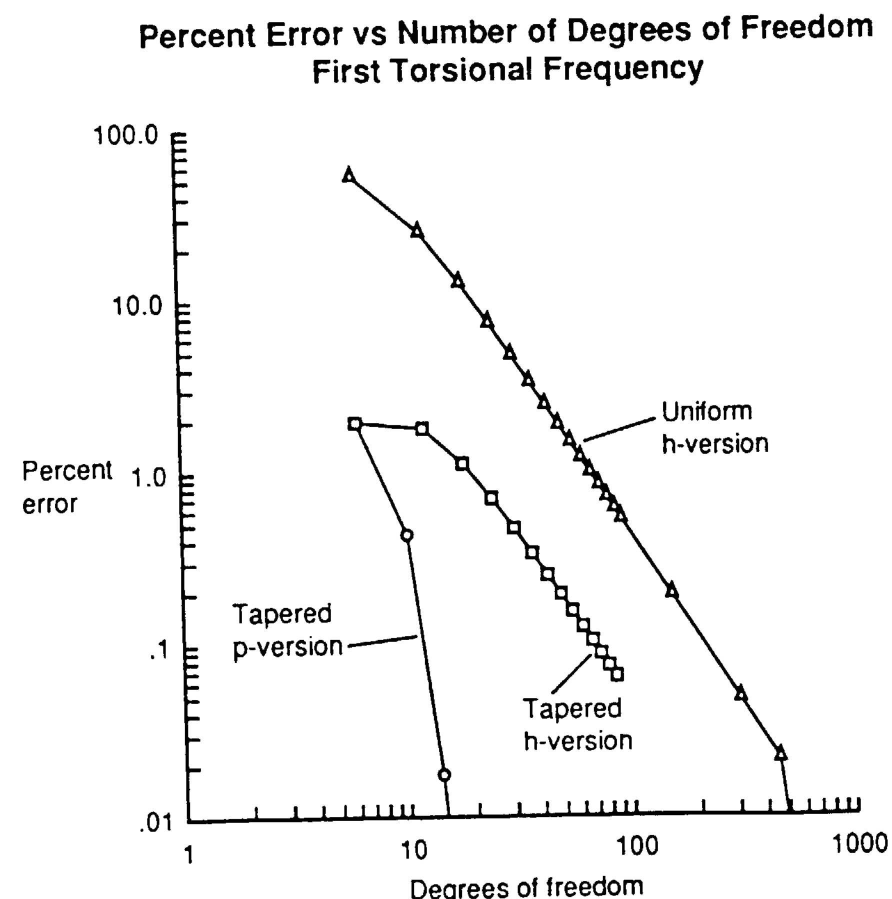


Figure 20 (b).

GROUND VIBRATION TEST OF BELL ALL-COMPOSITE AIRFRAME (ACAP) CONDUCTED AS PART OF DIFFICULT COMPONENTS STUDIES

Raymond G. Kvaternik
Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective: Excessive vibration is the most common technical problem to arise as a "show stopper" in the development of a new rotorcraft. With only a few exceptions, vibration problems have not been identified until flight test. Solutions at that stage of development are usually add-on fixes which adversely impact cost, schedule, and vehicle performance. Vibration predictions have not been relied on by the industry during design because of deficiencies in current vibration analysis methods. With a view toward establishing a capability in the industry to fully utilize vibration analysis during design, the NASA Langley Research Center has underway a program, designated DAMVIBS (Design Analysis Methods for VIBrationS), with the overall objective of establishing the foundations for developing a superior design analysis capability for vibrations. One of the many being conducted under the DAMVIBS program is aimed at identifying those "difficult components" which are the important contributors to airframe vibratory response and which require more detailed finite element representation.

Approach: Typically, only the primary (major load carrying) structure is represented fully (stiffness and mass) when forming the finite element model (FEM) of an airframe. There are many components (e.g., transmission, engines, and stores) and secondary structure (e.g., fairings, doors, and access panels) which are represented only as lumped masses. This may be a major contributing factor to the poor agreement which has been obtained between test and analysis at the higher frequencies of interest. To isolate the effects of such components on overall vibratory response, multiple ground vibration tests will be conducted with each test representing a progressive removal of a component until only the primary airframe structure remains. At each stage, analyses will be performed using an existing FEM of the airframe modified as necessary to reflect the specific configuration tested. Both full-scale airframes and their components and small-scale generic models of both metal and composite construction would be studied.

Accomplishment Description: A ground vibration test of the Bell all-composite (ACAP) helicopter airframe has been conducted by the Army at the Aviation Applied Technology Directorate, Fort Eustis. A total of 11 airframe configurations were tested ranging from full up to fully stripped down. Figure 21b shows the airframe in a partially stripped-down configuration. Components which were removed to isolate their effects included: cockpit and cargo doors, transparencies, horizontal and vertical stabilizers, landing gear, engines, and various panels and fairings. Detail analysis and study of the data are underway. Finite element models of all configurations tested are being formed by Bell using the finite element model of the ACAP developed earlier under the DAMVIBS program.

Significance: The subject test on the Bell ACAP helicopter airframe which was conducted as part of the difficult components studies represents the first such test conducted on a composite airframe structure. The results of this investigation, when completed, will identify other important contributors to airframe vibratory response, extending the results obtained in an earlier difficult components study conducted on a (metal) AH-1G helicopter airframe.

Future Plans: The difficult components studies are to continue through a combination of tests and analyses utilizing both full-scale composite airframes and their components as well as small-scale generic models of both metal and composite construction.

Figure 21 (a).

ORIGINAL PAGE
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GROUND VIBRATION TEST OF BELL ALL-COMPOSITE AIRFRAME (ACAP) CONDUCTED AS PART OF DIFFICULT COMPONENTS STUDIES

Figure 21 (b).

ANOMALY IDENTIFIED IN USE OF DAMPING TREATMENTS FOR REDUCING STRUCTURAL VIBRATIONS

Dr. Maurice I. Young (Vigyan)
Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective: To achieve up to an order of magnitude reduction in resonant or near resonant structural forced vibrations by the addition of one or more localized and concentrated damping treatments and devices. This is possible since aircraft and similar types of structures are inherently lightly damped and small additional amounts of damping will yield large reductions in forced vibratory responses when the structure is responding at or near resonance.

Approach: The aim is to develop a simplified analytical preliminary design method which will predict accurately the effect of localized, concentrated and other types of damping treatments in increasing the various modal vibration fractions of equivalent viscous critical damping. An energy quotient analogous to Raleigh's quotient for predicting undamped, natural frequencies of vibration for the various modes of complex structures is employed. This energy quotient utilizes the known structural mass, stiffness and undamped modes of vibration and permits a straightforward evaluation of the effect of damping treatment types, locations and magnitudes on the modal damping fractions. Application of this energy quotient for z_i is followed by an interaction analysis to account for a modal resonance effect when two modes have the same or nearly the same natural frequency.

Accomplishment Description: Rigorous engineering analysis has demonstrated that the damped modal vectors differ from the undamped modal vectors by terms of the order of the modal damping fraction, except in the case where there are degenerate structural modes which have identical or nearly identical natural frequencies. Accordingly, the simplified analytical preliminary design method followed by the interaction analysis yields predictions of z_i within one percent or less of the exact magnitudes when the damping fractions are less than ten percent of critical. In effect, the undamped modes are utilized to design the damped modes. An illustrative example of the method is given in figure 22b for a uniform simply supported beam with a tuning spring at the center. A viscous damper is employed with a view towards damping the second (anti-symmetrical) bending mode. The first (symmetrical) bending mode is tuned so that its natural frequency matches that of the second mode. As the frequency ratio (w_1/w_2) increases from its normal ratio of one fourth with increasing tuning spring stiffness, the effective damping fraction of the second mode progressively decreases from its design objective to zero. As the frequency ratio increases beyond unity, the effective damping fraction of the second mode recovers to its design objective. It is seen that the first mode steals damping intended for the second mode when their natural frequencies match or nearly match.

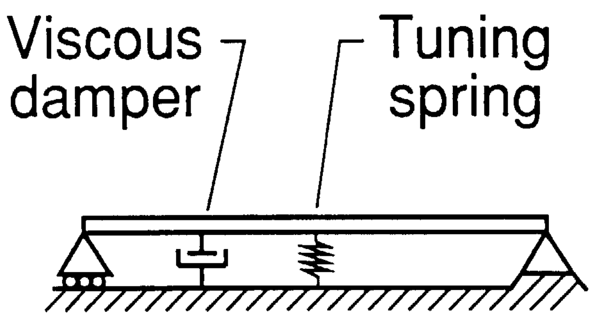
Significance: The anomaly of a resonating structure not benefiting from a damping treatment to reduce its vibratory response has been identified. An analytical method has been developed which computes effectiveness of the damping treatment when two different modes of vibration have matching or nearly matching natural frequencies. This permits the structural designer to better employ damping treatments in conjunction with structural mass and stiffness considerations which influence natural frequency placement.

Future Plans: To carry out analysis and laboratory testing on a structural model to validate the identification of this anomaly.

Figure 22 (a).

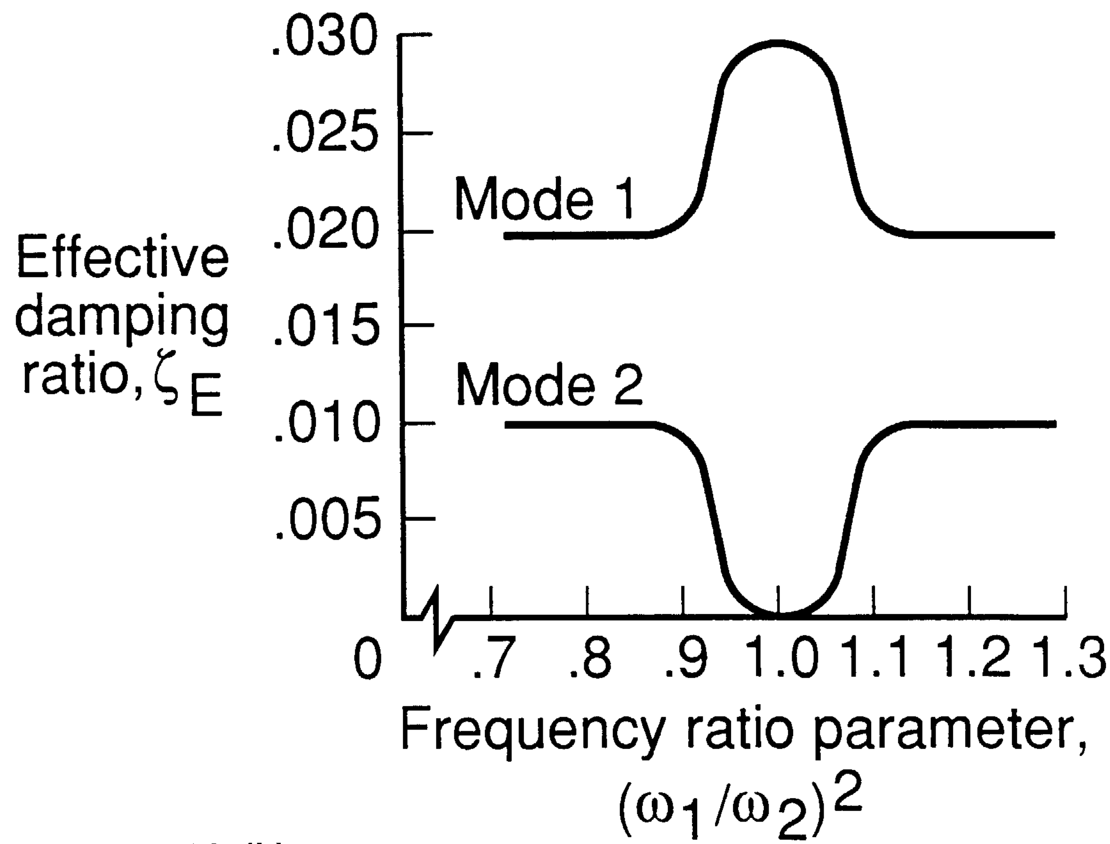
ANOMALY IDENTIFIED IN USE OF DAMPING TREATMENTS FOR REDUCING STRUCTURAL VIBRATIONS

SIMPLY SUPPORTED UNIFORM BEAM



DESIGN OBJECTIVE:

Second mode damping ratio, $\zeta_2 = .010$



D87-14

Figure 22 (b).

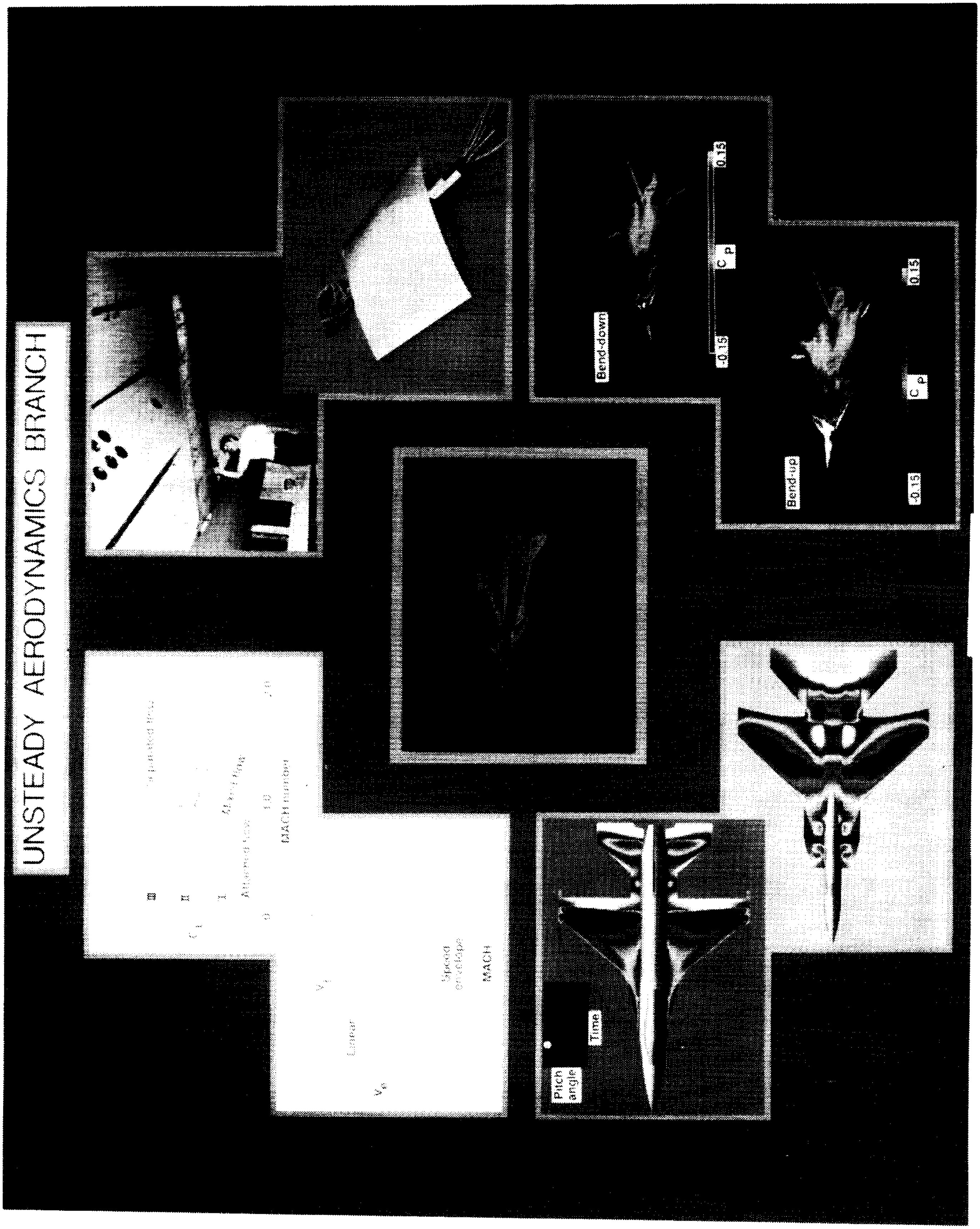


Figure 23.

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UNSTEADY AERODYNAMICS BRANCH

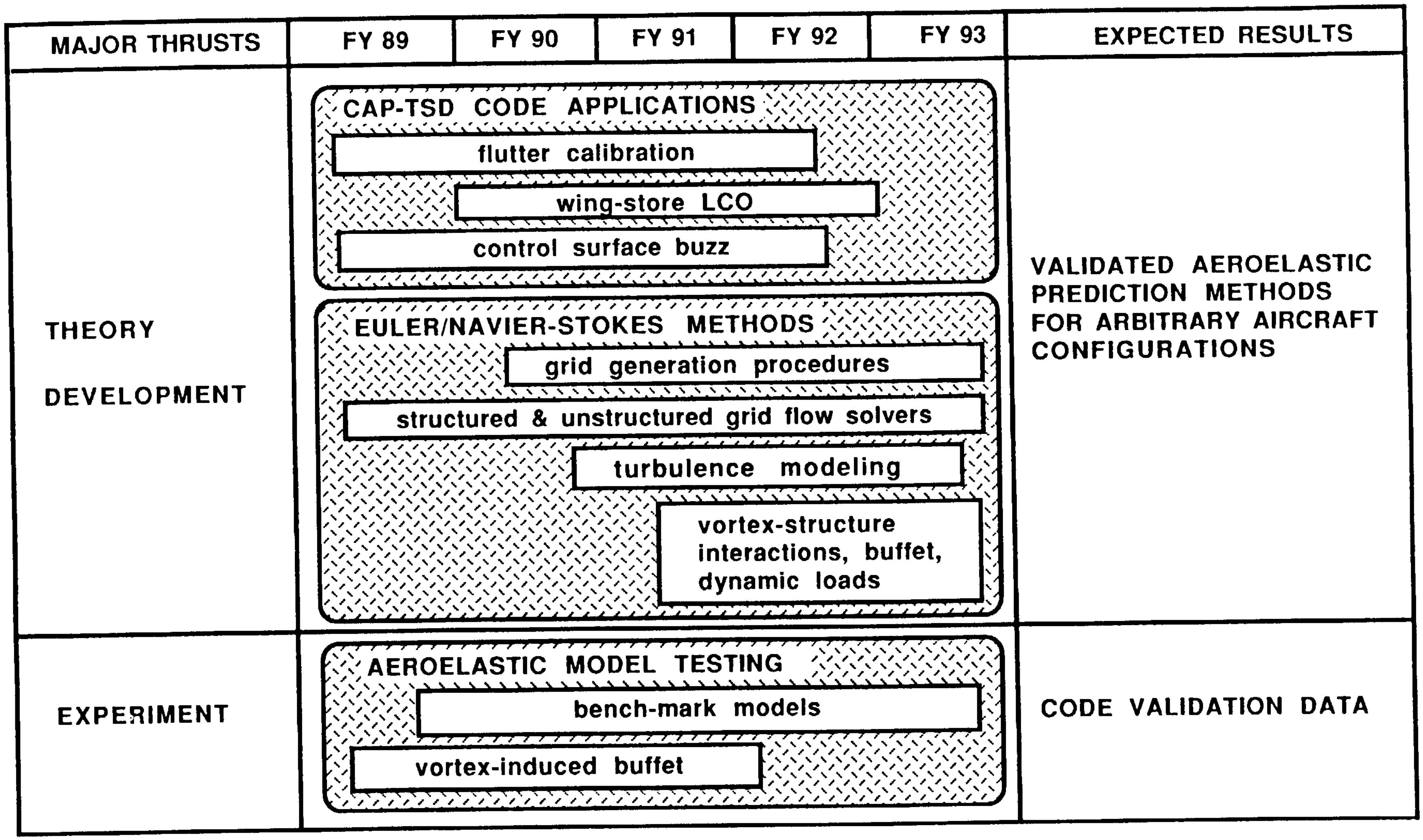


Figure 24.

FINITE-DIFFERENCE MESH DESIGN FOR THE TRANSONIC SMALL DISTURBANCE EQUATION

Samuel R. Bland
Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective: This research aims to provide finite-difference meshes with characteristics suitable for the time-accurate unsteady flow calculations needed in aeroelastic analysis.

Approach: A one-dimensional equation is used to model the unsteady wave propagation of the complete Transonic Small Disturbance (TSD) equation. Analysis of the finite-difference scheme and spatial discretization used confirm the numerical results obtained in solving the difference equations.

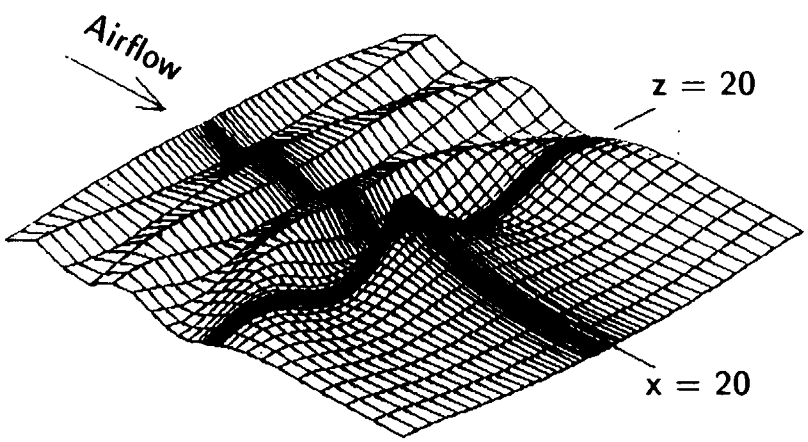
Accomplishment Description: The exact solution of the linearized version of TSD theory for an oscillating source is shown on the left in figure 25b. Waves propagate outward from an airfoil located in the center of a typical two-dimensional stretched finite-difference mesh. Experience with an early TSD code identified inaccuracies associated with the wave propagation in the z-direction. This effect is simulated in the present study by solving the equation for wave motion in this vertical direction using the same algorithm as is used in the state-of-the-art CAP-TSD code. Analysis of the finite-difference equation predicts that a reflection of the outgoing wave may occur at the simulated far-field boundary and will also be produced by the change in spacing needed to stretch the mesh away from the airfoil. The right of figure 25b illustrates these problems. Shown in each case is the response at the airfoil to a single pulse in downwash at time $t = 2$ using a mesh of 35 points with a 16 chord-length extent. In the top figure, a highly stretched mesh is used in which the ratio of the mesh spacing is constant. The oscillations beginning near $t = 24$ are produced by reflections that originate from distances about 2/3 of the way across the mesh. In the middle figure, a quadratically stretched mesh is used in which the second difference of the spacings is constant. This mesh is typical of those which we have used for unsteady flow calculations. The calculation is much improved, with only a small disturbance caused by the reflection from the far-field boundary observed. The result shown in the bottom figure was obtained on a mesh in which a more gradual stretching was used and in which the spacing was reduced at the far-field boundary. This type of mesh, which is a product of the present study, is currently being used in the CAP-TSD code for aeroelastic analysis.

Significance: This study provides guidelines for the properties of finite-difference meshes and methods needed to obtain accurate solution of the unsteady flow equations in aeroelastic analysis.

Future Plans: The techniques developed for TSD theory using the simple one-dimensional model equation will be extended to the more exact Euler and Navier-Stokes equations.

Figure 25 (a).

FINITE-DIFFERENCE MESH DESIGN FOR THE TRANSONIC SMALL DISTURBANCE EQUATION



Wave propagation in two-dimensional linear theory

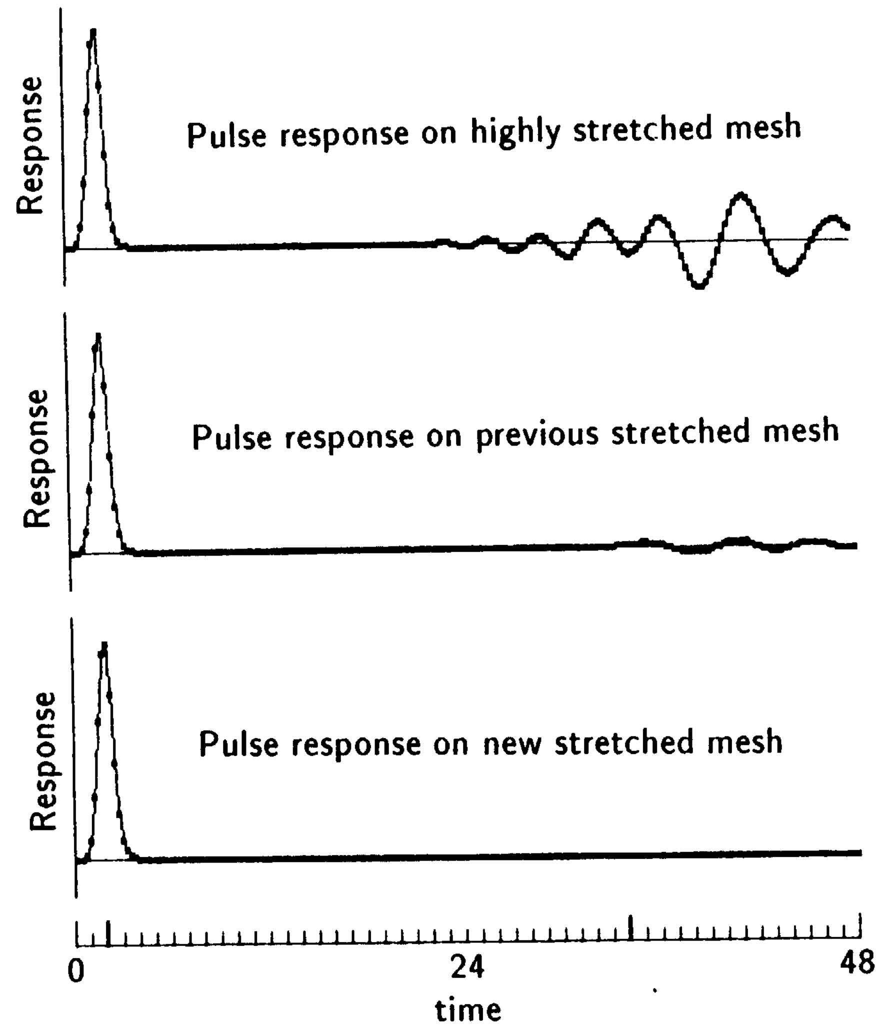


Figure 25 (b).

UNSTEADY VORTEX-DOMINATED FLOWS COMPUTED USING A STRUCTURED MESH, NAVIER-STOKES SOLVER

Professor Osama A. Kandil and Dr. Andrew Chuang
Old Dominion University

RTOP 505-63-21

Research Objective: The major objective of the present research work is to develop a consistent and systematic formulation and computational scheme for accurate and efficient prediction of unsteady, vortex-dominated flows. The flow unsteadiness could be due to rigid-body motions, aeroelastic deformations and/or relative rigid-body motions.

Approach: The formulation of the problem consists of two sets of equations; the conservative, unsteady, compressible Navier-Stokes equations written in a general moving frame of reference and the non-conservative, unsteady Navier displacement equations written for fluid flows. The first set is solved for the conservative components of the flow field vector using an implicit, approximately-factored, central-difference finite-volume scheme. The second set uses the solution of the first set along with prescribed displacement boundary conditions to solve for the grid displacements at any time step. The alternating-direction-implicit (ADI) scheme is used for the solution of the second set. These schemes are coded in the well known program "ICF3D".

Accomplishment Description: A sample computation for a locally-conical, unsteady flow around a sharp-edged delta wing undergoing bending-mode oscillations is presented in figure 26b. The grid is initially generated for the steady initial conditions using a modified Joukowski transformation over three cross-flow planes. The wing is then deformed using a forced bending-mode oscillation. With a time step of 0.5×10^{-3}, each cycle of oscillation requires 4000 time steps. The grid deformation is computed every 50 time steps. Two snapshots of the solution at the time steps of 14,000 and 15,000 are given in the figure which shows the grid, cross-flow velocities for the left and right leading-edges and surface-pressure coefficient. The details of primary and secondary vortices are clearly captured. The vortex on the left is decreasing due to the wing concaving motion. The corresponding change in the surface pressure is clearly evident.

Significance: The present novel formulation and efficient computational code can treat unsteady vortex-dominated flow problems in which flow unsteadiness is due to rigid-body motion, aeroelastic deformation and/or relative rigid-body motion. The formulation and solution are ready to be coupled with the dynamics and aeroelastic problems.

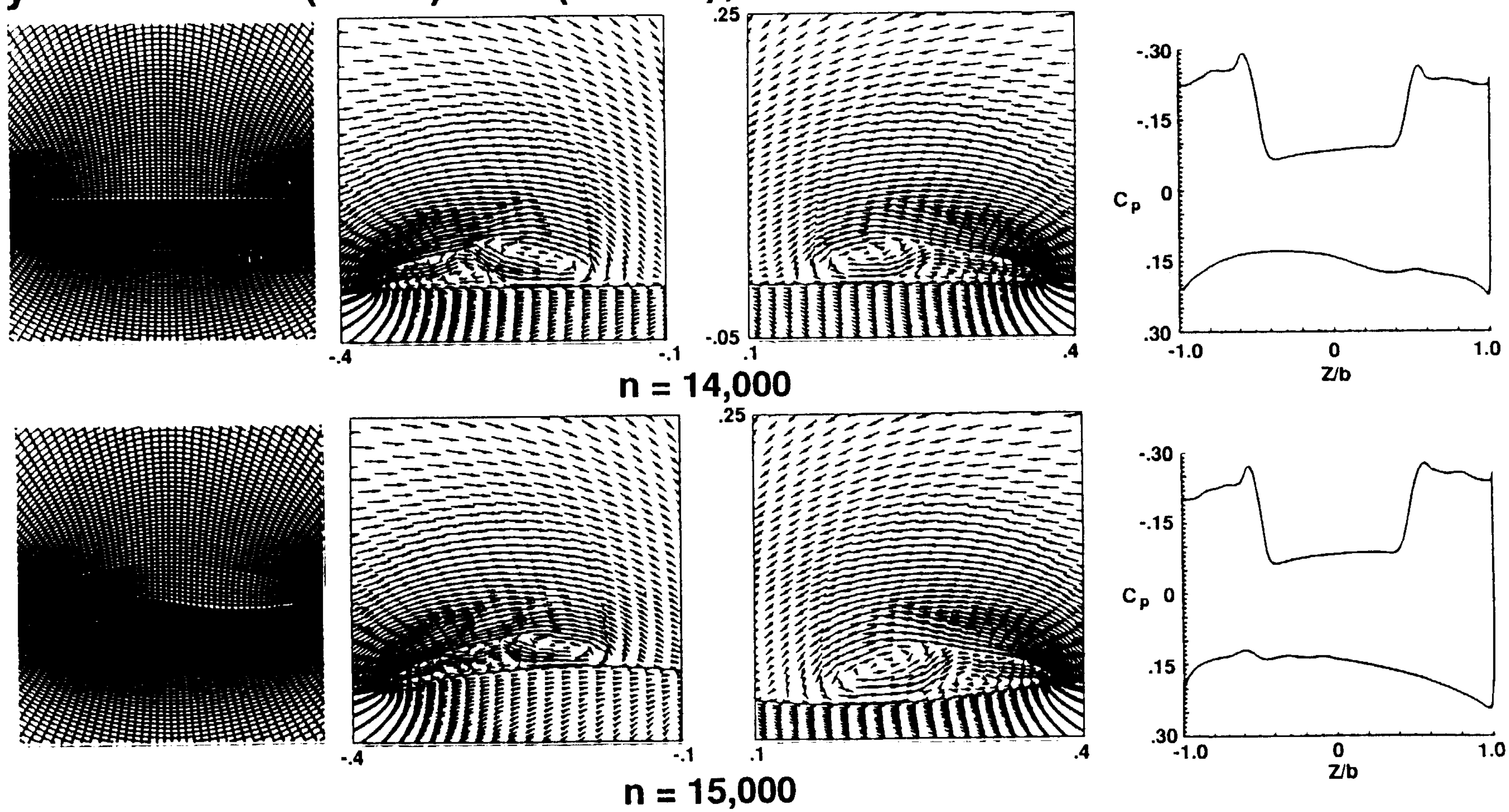
Future Plans: Large-amplitude oscillations including the vortex breakdown critical range will be studied. Oscillations of leading-edge flaps are also investigated. The fluid dynamics problem will be coupled with the dynamics and aeroelastic problems. Moreover, several turbulence models will also be investigated.

Figure 26 (a).

BENDING-MODE OSCILLATION OF A SHARP-EDGED DELTA WING NAVIER/STOKES NAVIER-DISPLACEMENT SOLVERS

$\alpha = 10°$, $M_\infty = 2$, $Re = 0.5 \times 10^6$, 264 x 90 grid points

$y = 0.05b \sin(\Pi Z/b) \sin(2\Pi t/\tau)$; $b = \tan 20°$, $\tau = 2$, $\Delta t = 0.5 \times 10^{-3}$



ADAPTIVE MESH REFINEMENT CAPABILITY DEVELOPED FOR ACCURATE VORTICAL FLOW COMPUTATION

John T. Batina
Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective: Many of the vortex-dominated flow solutions that have been reported in the literature have been obtained without the grid resolution that is required for high spatial accuracy. The objective of this research was to develop an adaptive mesh refinement capability to enrich the mesh locally, as a part of the solution procedure, to more accurately resolve the vortical flow features.

Approach: An adaptive refinement procedure was developed within a conical Euler/Navier-Stokes flow solver based on the use of unstructured grids of triangles. In this procedure, total pressure loss is used as an indicator to determine where to refine the mesh, since total pressure loss is a good indicator of the vortical flow structure. If the total pressure loss within a given triangle exceeds a preset tolerance, the triangle is divided into smaller triangles. Each time the mesh is refined, the flow variables at the new nodes of the mesh are determined by interpolation from neighboring nodes, and the calculation proceeds. Refinement is typically performed two or three times during the course of the calculation to produce a locally fine mesh in the vortical regions.

Accomplishment Description: To test the new adaptive refinement procedure, conical Euler calculations were performed for a 75° flat plate delta wing at a freestream Mach number of $M_\infty = 1.4$, an angle of attack of $\alpha = 20°$, and a sideslip angle of $\beta = 10°$. The calculations were performed by starting with a coarse mesh, as shown in the left part of figure 27b, and running the flow solver for 4000 iterations. An intermediate solution at iteration 500, also shown in the figure, indicates that the vortices are very diffuse. The mesh was then adapted at iterations 500, 1000, and 1500, which produced the final mesh and solution that are shown in the right part of the figure. This solution now shows clearly that there is a strong flat vortex produced by the separated flow from the windward (left) leading edge with a crossflow shock wave beneath the vortex. There is also a weaker more circular vortex produced by the separated flow from the leeward (right) leading edge.

Significance: The results indicate that the vortical flow features are more clearly defined when the mesh is adaptively refined. A spatially accurate solution is thus obtained for this case using an order of magnitude fewer grid points (2,801 nodes) than if a globally fine mesh were used (32,768 nodes).

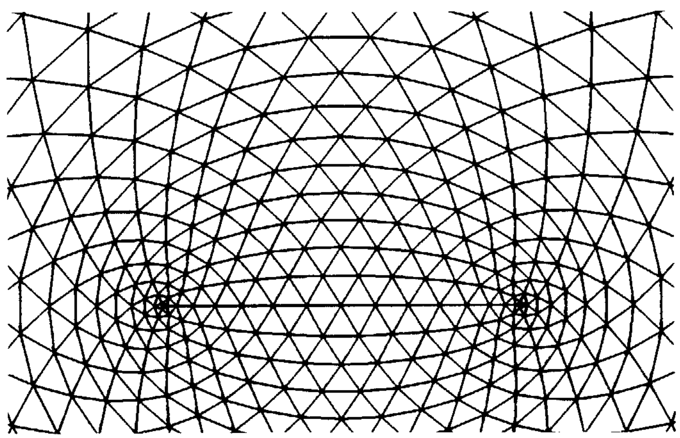
Future Plans: The capability will be used to study unsteady vortical flows about rolling delta wings to gain insight into the flow mechanisms responsible for the so-called wing rock phenomenon.

Figure 27 (a).

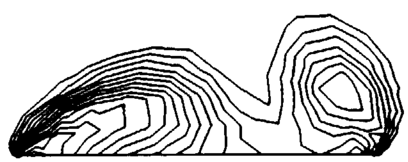
ADAPTIVE MESH REFINEMENT CAPABILITY DEVELOPED FOR ACCURATE VORTICAL FLOW COMPUTATION

- 75° swept flat-plate delta wing at $M_\infty = 1.4$, $\alpha = 20°$, and $\beta = 10°$

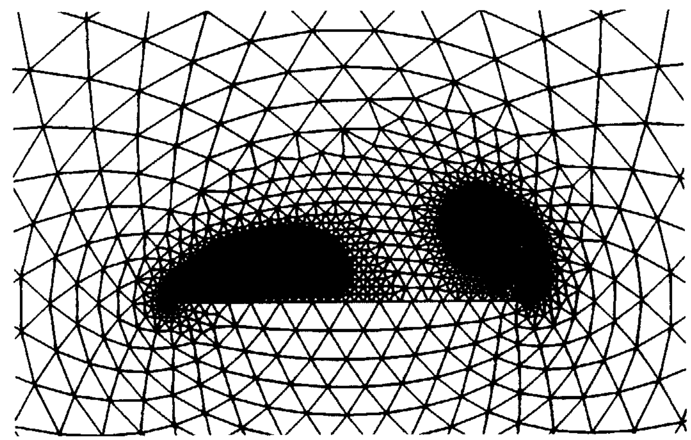
- Starting solution
  - coarse mesh (512 node)



  - total pressure loss



- Adapted solution
  - adapted mesh (2801 nodes)



  - total pressure loss

Figure 27 (b)

NEW UNSTEADY EULER ALGORITHM IMPROVES ACCURACY AND EFFICIENCY OF UNSTRUCTURED GRID METHODOLOGY

John T. Batina
Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective: Previous methods of solving the unsteady Euler equations on unstructured grids are inefficient because of the explicit time-marching involved and are less accurate than upwind schemes on structured meshes because of the central-difference-type spatial discretization involved. Therefore the objective of this research was to develop a new solution algorithm for the unsteady Euler equations on an unstructured grid of triangles that is more accurate and efficient than previous methods.

Approach: A new algorithm has been developed which involves improved spatial and temporal discretizations of the Euler equations. The spatial discretization now involves a so-called flux-split (upwind) approach which accounts for the local wave-propagation characteristics of the flow and captures shock waves sharply with at most one grid point within the shock structure. The flux-split discretization is naturally dissipative and consequently does not require additional artificial dissipation terms or the adjustment of free parameters to control the dissipation. Furthermore, the temporal discretization is now an implicit time-integration scheme involving a Gauss-Seidel relaxation procedure. The scheme allows the selection of the step size based on the temporal accuracy dictated by the problem being considered, rather than on the numerical stability of the algorithm. Consequently, very large time steps may be used for rapid convergence to steady state, and an appropriate step size may be selected for unsteady cases, independent of numerical stability issues.

Accomplishment Description: To test the new algorithm, steady flow results were obtained for the NACA 0012 airfoil at $M_\infty = 0.8$ and $\alpha = 1.25°$, using both implicit and explicit time-marching. The explicit time-marching results were obtained using a CFL number of 2.5 and the implicit time-marching results were obtained using a CFL number of 100,000. A comparison of the convergence histories is shown in the left part of figure 28b and the resulting steady pressure distributions are shown in the right part. The "error" in the solution was taken to be the L_2-norm of the density residual. As shown, the explicit solution is very slow to converge whereas the implicit solution is converged to four orders of magnitude in only approximately 500 steps. Also, the pressure distributions indicate that there is only one grid point within the shock structure, on either the upper or lower surface of the airfoil, due to the sharp shock capturing ability of the flux splitting in comparison with central-differencing.

Significance: Converged steady solutions are obtained with the implicit algorithm with an order of magnitude less CPU time than the explicit algorithm, and the shock waves are more sharply captured with the flux-split spatial discretization than the central-difference-type discretization. These improvements in accuracy and efficiency are also realized for unsteady applications.

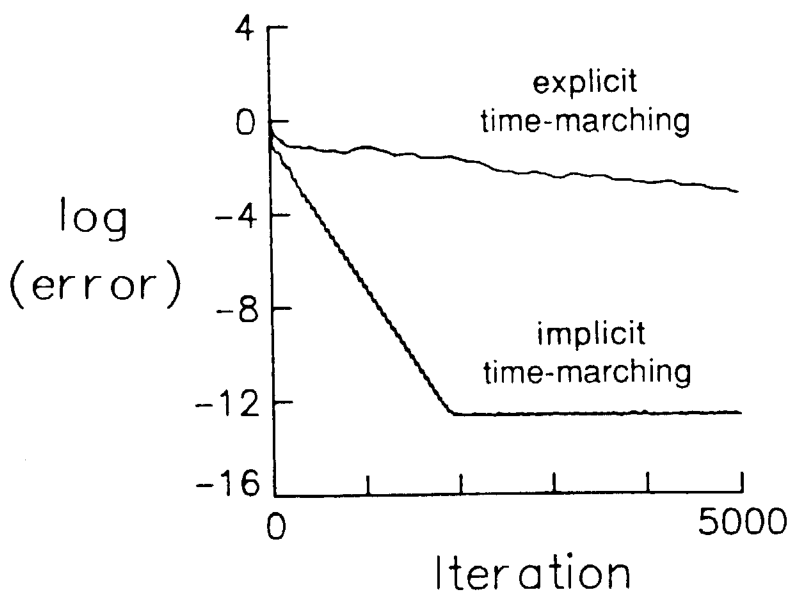
Future Plans: The three-dimensional version of the implicit flux-split algorithm will be developed and validation calculations will be performed.

Figure 28 (a).

NEW UNSTEADY EULER ALGORITHM IMPROVES ACCURACY AND EFFICIENCY OF UNSTRUCTURED GRID METHODOLOGY

- NACA 0012 airfoil at $M_\infty = 0.8$ and $\alpha = 1.25°$

- Efficiency improvement due to implicit temporal discretization - convergence history comparison



- Accuracy improvement due to flux-split spatial discretization - steady pressure comparison

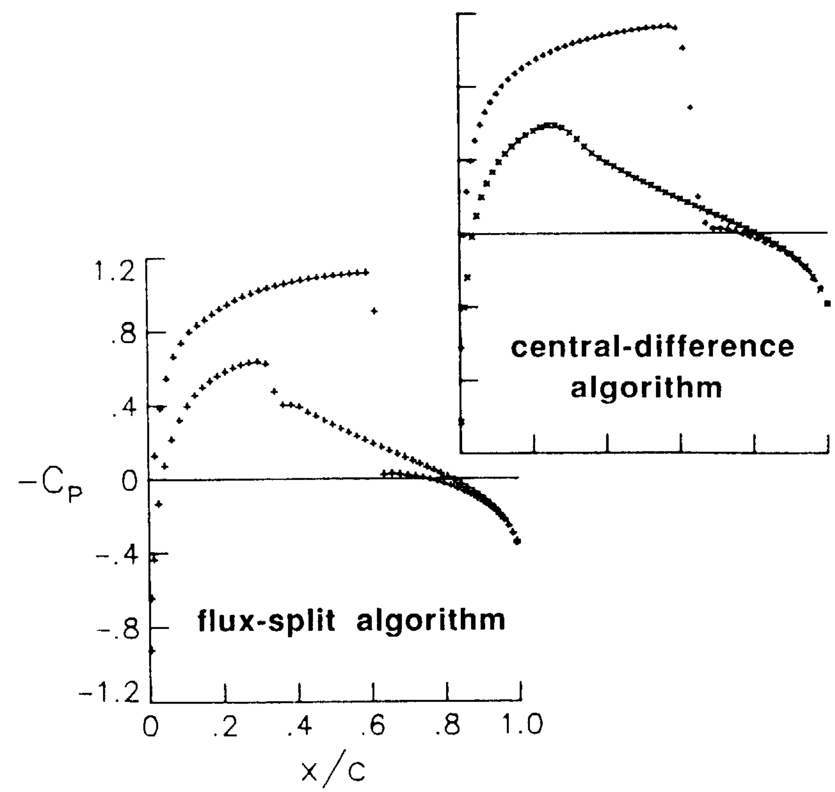


Figure 28 (b).

TEMPORAL ADAPTIVE EULER METHOD DEVELOPED FOR UNSTEADY AERODYNAMIC ANALYSIS

William L. Kleb and John T. Batina
Purdue University and Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective: The application of current explicit time-marching schemes for integrating the Euler equations for unsteady problems is computationally expensive because of the very small global time step required for numerical stability. The objective of this research was to improve the efficiency of such schemes through the use of temporal adaptation.

Approach: A temporal adaptation algorithm has been developed together with an explicit time-marching approach to solving the unsteady Euler equations. Temporal adaptation is a method of time integration whereby each cell of the grid is integrated using a different time step. Shown in the left part of figure 29b, for example, is a temporal stencil for a one-dimensional model problem which diagrams the integration from time level n to time level n+1. The diagram shows that the small cells are integrated with small time steps and large cells are integrated with large time steps. If the value needed for the integration is unknown at a particular temporal node, it is determined from a linear interpolation between two known values. Time accuracy is maintained by bringing all of the cells to the same time level (n+1) as determined by the step size of the largest cell.

Accomplishment Description: Calculations were performed for a one-dimensional shock-tube problem to test the speed and accuracy of the temporal adaptive algorithm. The problem is a good test case since it contains all of the major types of flow features expected to occur in transient problems including a moving shock wave, an expansion fan, and a contact surface. The test case involved a density ratio of five run on a mesh of random spacing with 100 points, which is a challenging test for the adaptive algorithm since the local time step will vary considerably from grid cell to grid cell. The right part of the figure shows the resulting density profiles at three times during the calculation.

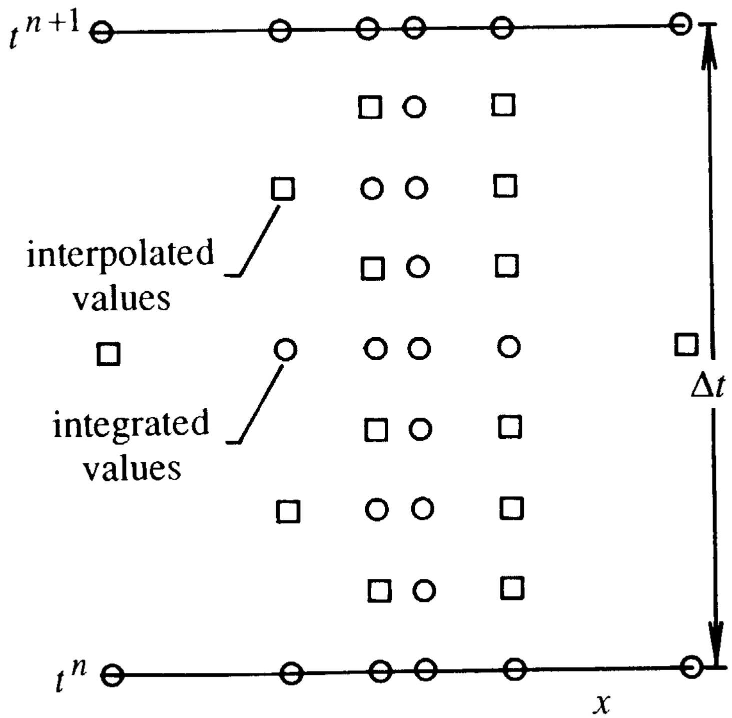
Significance: These profiles show good agreement between results obtained using the temporal adaptive time-stepping and global time-stepping which tends to verify the one-dimensional temporal adaptive method. Both calculated solutions also agree well with the exact solution. Furthermore, the solution obtained using temporal adaptation was five times less expensive than the solution obtained using global time-stepping.

Future Plans: The temporal adaptive algorithm is being applied currently to a two-dimensional unsteady Euler/Navier-Stokes algorithm based on the use of unstructured grids and applications will be performed for oscillating airfoils.

Figure 29 (a).

TEMPORAL ADAPTIVE EULER METHOD DEVELOPED FOR UNSTEADY AERODYNAMIC ANALYSIS

- Method integrates small cells with small time steps and large cells with large time steps
- Time accuracy is maintained by bringing all cells to same time level as dictated by step size of largest cell
- Typical temporal stencil



- 1D shock tube example

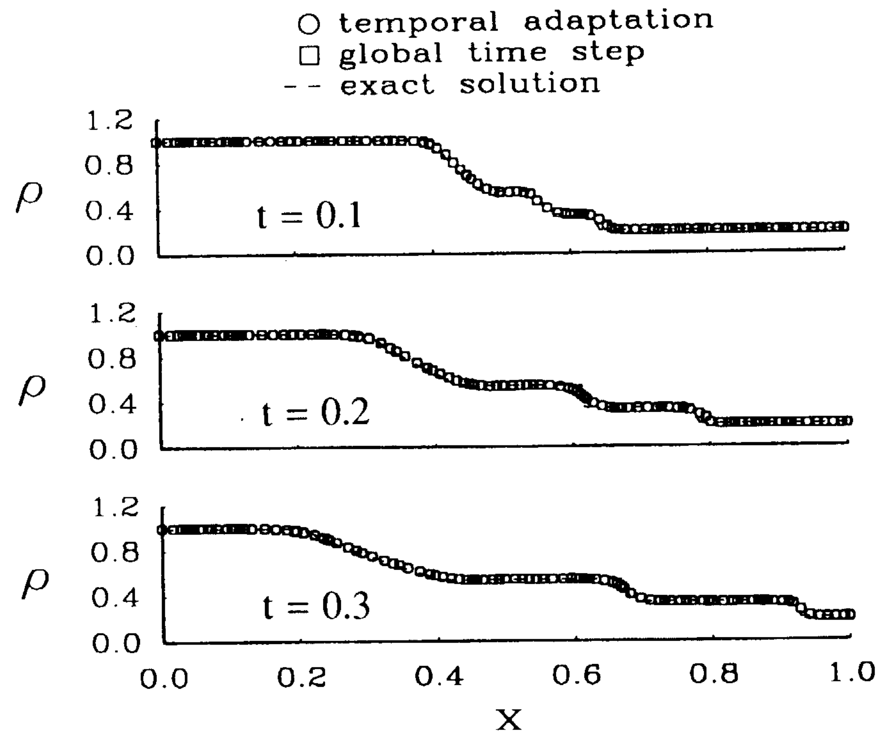


Figure 29 (b).

GRAPHICS CODE DEVELOPED TO PERMIT RAPID VISUALIZATION OF CFD RESULTS FOR 2-D UNSTRUCTURED MESHES

Robert W. Neely
Lockheed

RTOP 506-80-31

Research Objective: Computational Fluid Dynamics (CFD) methods are used routinely to predict spatial and temporal distributions of flowfield velocity components, density and total energy per unit volume. From these primitive variables, a number of other important properties are computed, such as surface pressure coefficient, local Mach number, and flowfield static pressure. The objective of this research was the identification of an appropriate workstation environment and the development of needed graphical software for use in displaying and analyzing the large variety of numerical results obtained from CFD solution procedures based on unstructured computational-grid topologies.

Approach: The graphical display of CFD results obtained from unstructured grids required the development of fast and efficient data search algorithms, coupled with a suitable graphics workstation on which to run the software and display the computed results. The approach taken was to develop the needed software using a robust plotting language (DI-3000) together with graphics software capabilities available with IRIS graphics workstations.

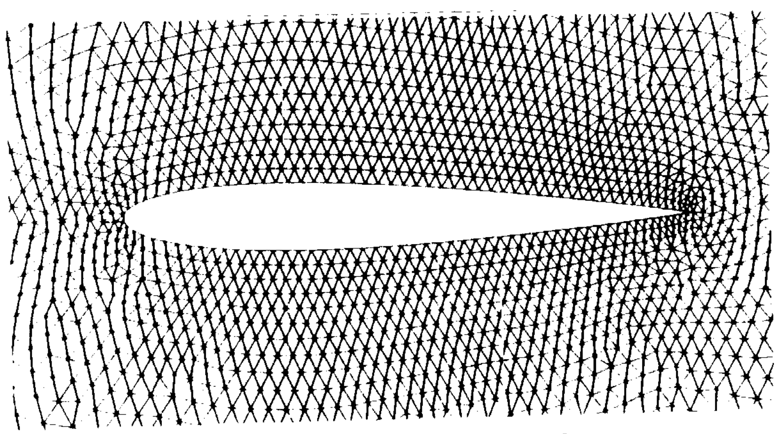
Accomplishment Description: A complete 2-D plotting code was developed to display data generated from a CFD code using unstructured grids. The plotting code permits the display of the computational grid used in the calculation, together with Mach number contours, pressure contours, velocity vectors and particle traces. Figure 30b shows sample results displayed for a typical CFD calculation. Using the hardware features inherent in the graphics workstation, the analyst can translate and/or magnify the display as desired.

Significance: The graphics program developed permits the rapid selection and plotting of CFD-generated flowfield data, thus enabling the aerodynamicist to better understand steady and unsteady fluid dynamic phenomena under investigation.

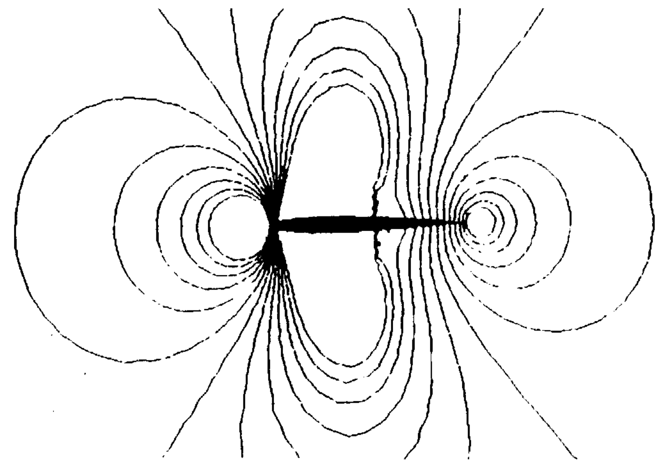
Future Plans: To develop a similar graphics capability for 3-D CFD codes using unstructured computational meshes.

Figure 30 (a).

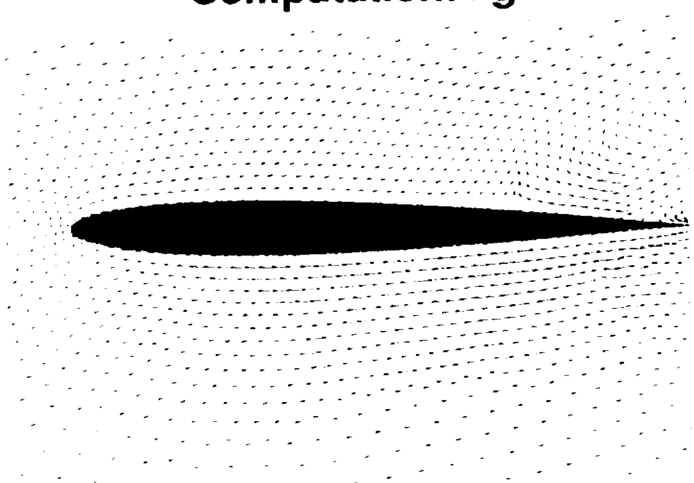
CFD FLOW VISUALIZATION FOR UNSTRUCTURED MESHES



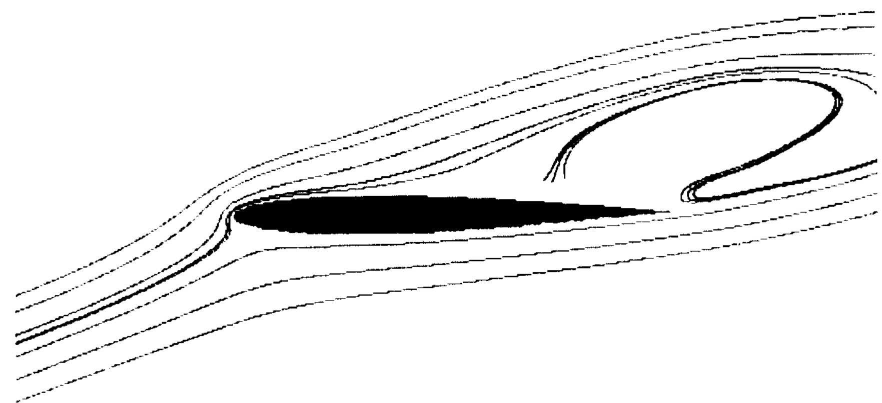
Computational grid



Pressure contours ($\alpha = 0.0°$)



Velocity vectors ($\alpha = 20.0°$)



Particle traces ($\alpha = 20.0°$)

NACA0012 Airfoil, $M_\infty = 0.84$

Figure 30 (b).

CAP-TSD VISCOUS CODE CONFIRMS SIMPLE CRITERIA FOR AILERON BUZZ

James T. Howlett
Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective: The objective of this research is to develop accurate analytical techniques for predicting the onset of aileron buzz.

Approach: An integral boundary layer method is interacted with the inviscid CAP-TSD computer code to calculate the effects of viscosity on the unsteady flow field surrounding an airfoil with an oscillating flap. The boundary layer calculation includes an inverse method for separated flows as well as a direct method for attached flows. The aeroelastic capabilities of the CAP-TSD computer code are used to determine the limit cycle oscillations typical of aileron buzz.

Accomplishment Description: A simple criteria for aileron buzz based upon steady experimental results was discussed by Lambourne in 1960 (Lambourne, N. C.: Some Instabilities Arising from the Interactions Between Shock Waves and Boundary Layers, A. R. C. Technical Report C. P. No. 473, 1960). A plot from this reference is reproduced in the upper left side of figure 31b with recently calculated results for the P-80 airfoil added. As the figure indicates, unsteady buzz oscillations are encountered at Mach Number 0.771, slightly above the sudden reduction in trailing edge pressure coefficient. The boundary layer shape factors for steady flow shown on the upper right of the figure indicate that steady flow separation over the entire flap is coincident with the reduction in trailing edge pressure coefficient. A plot of unsteady flap angle versus time, shown on the lower left side of the figure, confirms that the limit cycle oscillations typical of aileron buzz occur at a Mach Number of 0.771. The plot of unsteady boundary layer shape factor on the lower right of the figure clearly indicates the phase lag between the minimum downward deflection of the flap and massive boundary layer separation which occurs at a flap angle of 0 degrees and moving upwards.

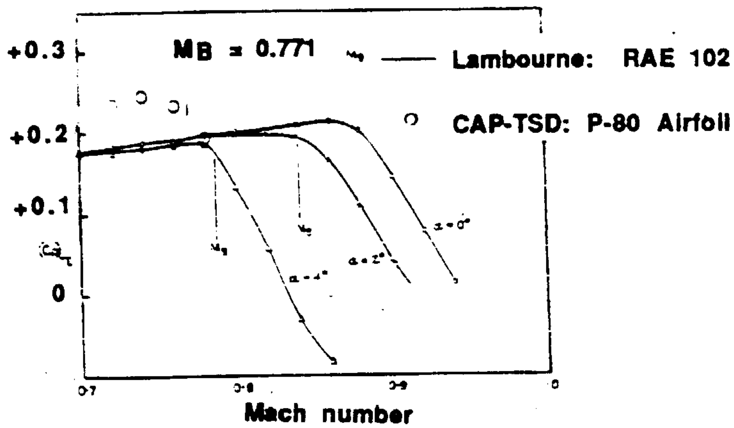
Significance: The results indicate that significant savings in computer time and wind tunnel testing time may be obtained by using steady results to determine the Mach number ranges where the unsteady buzz phenomena may be expected to occur.

Future Plans: Further calculations for a variety of conditions and airfoils are required to confirm the general validity of this criteria. Studies of 3-dimensional wings with trailing edge flaps also needed to extend the criteria to practical configurations.

Figure 31 (a).

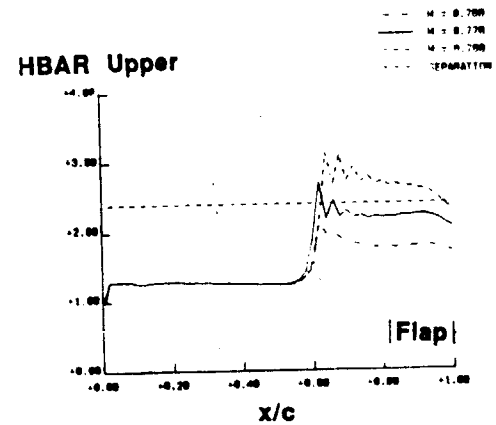
CAP-TSD VISCOUS CODE CONFIRMS SIMPLE CRITERIA FOR AILERON BUZZ

Steady



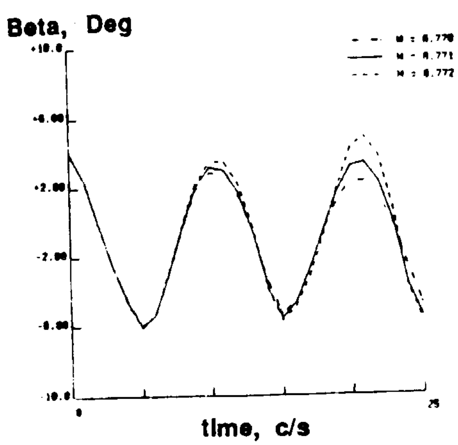
- Buzz onset indicated by drop in steady Cp at trailing-edge

Steady



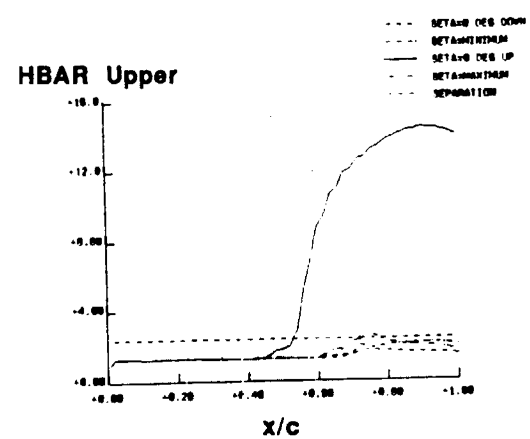
- Steady boundary layer separates over entire flap at buzz Mach no.

Unsteady



- Variation of flap angle with time for 3 Mach No.

Unsteady



- Variation of boundary layer separation during a cycle of buzz oscillation at M = 0.771.

Figure 31 (b).

FLUTTER ANALYSIS OF A HIGHLY SWEPT DELTA WING WITH THICKNESS

Michael D. Gibbons
Lockheed

RTOP 506-80-31

Research Objective: The objective of this research was to determine the accuracy of a transonic aeroelastic analysis code in predicting flutter for a highly swept delta wing with thickness.

Approach: Comparisons of flutter conditions are made between analytical results and experimental flutter values. The small disturbance code used to predict flutter is known as CAP-TSD which is an acronym for Computational Aeroelasticity Program - Transonic Small Disturbance.

Accomplishment Description: The flutter calculations were performed on a delta and a clipped delta wing with a sweep of 72° and a three percent thick circular arc airfoil. Flutter calculations were made over a range of Mach numbers $M_\infty = 0.6$ to 1.2. A finite element program was used to calculate the required mode shapes which were tuned to match the experimental frequencies. Comparisons of theory and experiment are shown in figure 32b for the 72° delta wing. region. The CAP-TSD results compare well with experiment in the subsonic-transonic region.

Significance: The results show that fairly good agreement between calculated flutter boundaries and experiment can be achieved for highly swept delta wings with moderate thickness for transonic flows.

Future Plans: Further calculations involving more complicated flows and geometries are planned using CAP-TSD and other codes based on higher equation levels.

Figure 32 (a).

FLUTTER ANALYSIS OF A HIGHLY SWEPT DELTA WING WITH THICKNESS

- Leading edge swept 72° with a 3% thick circular arc airfoil
- Comparison of calculated flutter boundary using CAP-TSD and experimental results

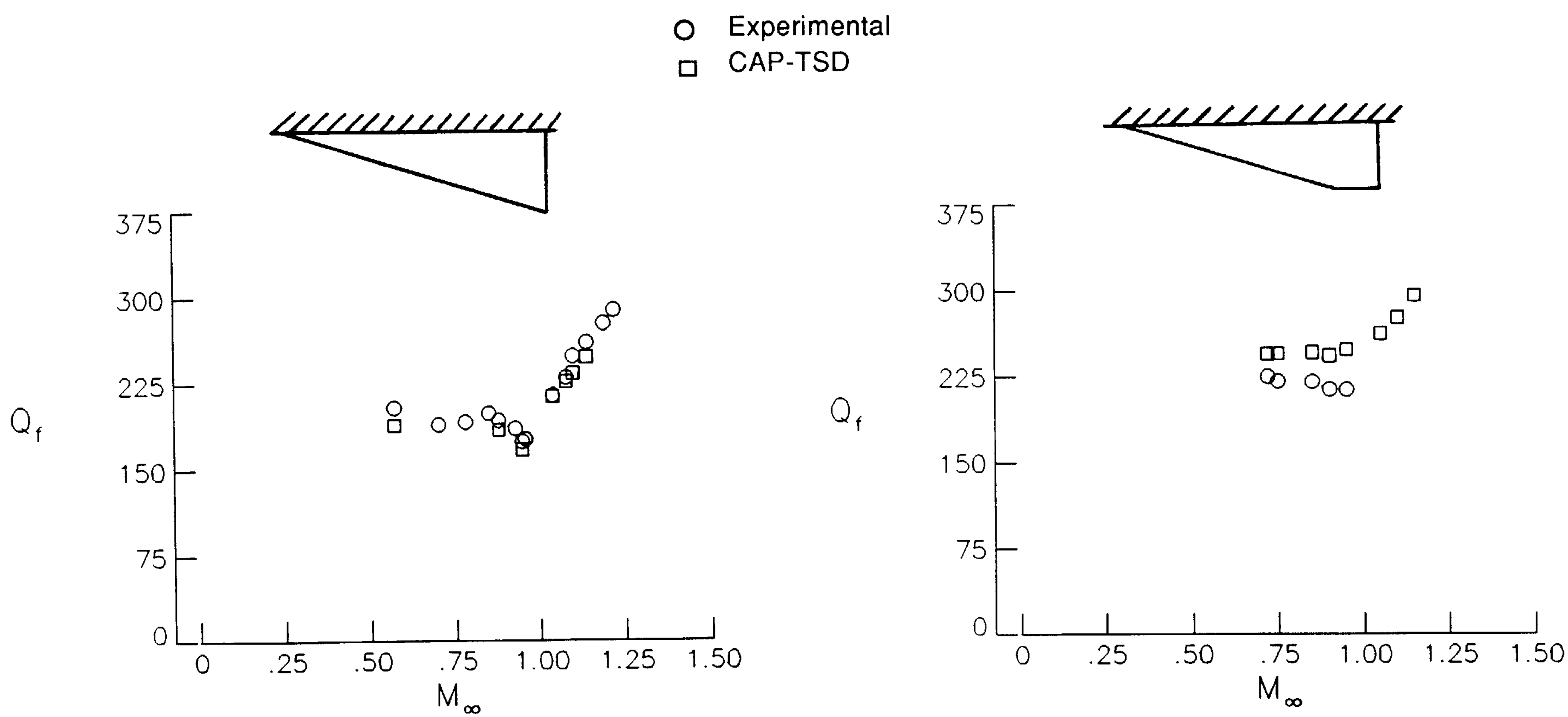


Figure 32 (b).

CFL3D CODE MODIFIED FOR AEROELASTIC ANALYSIS

Brian A. Robinson and John T. Batina
Purdue University and Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective: Computational aeroelasticity codes based on transonic small-disturbance potential theory are limited to applications involving thin lifting surfaces at small angles of attack undergoing small deformations. To remove these limitations, the higher governing fluid flow equations such as the Euler or Navier-Stokes equations are generally required. Therefore, the objective of this research was to develop an aeroelastic analysis capability within an existing three-dimensional Euler/Navier-Stokes aerodynamics code.

Approach: Modifications were made to the CFL3D unsteady Euler/Navier-Stokes code (developed by the Computational Aerodynamics Branch, FldMD) to allow the aeroelastic analysis of wings. The CFL3D code is a three-factor, implicit algorithm based on upwind-biased spatial differencing. The modifications involve including a deforming mesh capability which can move the mesh to continuously conform to the instantaneous shape of the aeroelastically deforming wing, and including the structural equations of motion for their simultaneous time-integration with the governing flow equations.

Accomplishment Description: Results were obtained using the Euler equations to verify the modifications to the code. To test the deforming mesh capability, calculations were performed for a NACA 0012 airfoil pitching harmonically at $M_\infty = 0.8$ and $\alpha = 0°$. Results were obtained using a rigidly rotating mesh as well as using the deforming mesh. As shown in the left part of figure 33b, the lift coefficient due to pitch obtained using the two ways of moving the mesh agree well with each other and with similar results obtained independently by Rausch. This good agreement, for the entire range of reduced frequency k that was considered, tends to verify the deforming mesh capability that was implemented within CFL3D. Calculations were also performed for an AGARD standard aeroelastic wing to assess the code for aeroelastic analysis. The wing has a quarter-chord sweep of 45°, a panel aspect ratio of 1.65, and a taper ratio of 0.66. The right part of the figure shows computed aeroelastic transients for 0.9 and 1.1 times the experimental flutter dynamic pressure, Q_{exp}, at $M_\infty = 0.9$ and $\alpha = 0°$. These transients show stable & unstable motions which bracket the flutter point.

Significance: From these aeroelastic transients, the flutter speed was determined by interpolation of the damping. This flutter speed was found to be within 3% of the experimental value which tends to verify the aeroelasticity modifications to the CFL3D code.

Future Plans: The CFL3D aeroelasticity capability will be applied to additional cases to further validate the methodology.

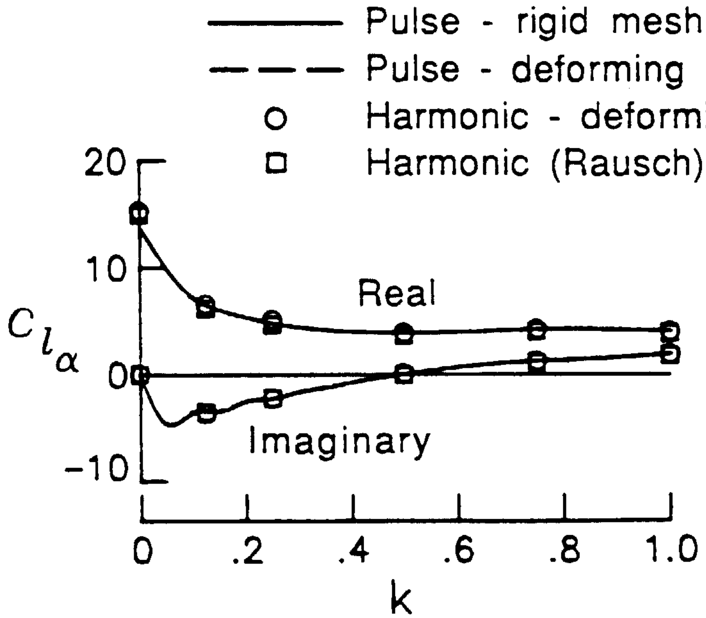
Figure 33 (a).

CFL3D CODE MODIFIED FOR AEROELASTIC ANALYSIS

- **Modifications to CFL3D Euler/Navier-Stokes code**
  - **deforming mesh capability to move mesh to conform to instantaneous shape of aeroelastically deforming wing**
  - **structural equations of motion for simultaneous time-integration**

- **Deforming mesh verification - NACA 0012 airfoil at $M_\infty = 0.8$,**

  $\alpha = 0°$



- **Aeroelastic application - AGARD wing at $M_\infty = 0.9$,**

  $\alpha = 0°$

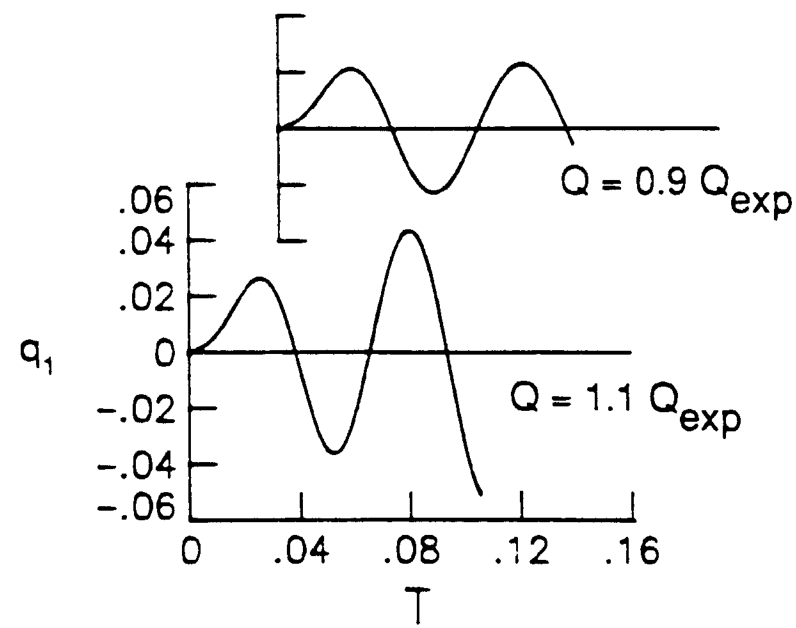


Figure 33 (b).

EULER FLUTTER ANALYSIS OF AIRFOILS DEMONSTRATED USING UNSTRUCTURED DYNAMIC MESHES

Russ D. Rausch and John T. Batina
Purdue University and Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective: Flow solvers based on unstructured grids have advantages over structured grid flow solvers such as the ability to treat very complicated geometries and the ability to adaptively refine the grid for high spatial accuracy. The methodology may also have advantages for unsteady aerodynamic and aeroelastic analysis. Therefore, the objective of the research was to assess the accuracy of the unstructured grid methodology for flutter analysis of airfoils as a first step toward aeroelastic analysis of complete aircraft configurations.

Approach: Modifications were made to a two-dimensional unsteady Euler code involving unstructured grids of triangles, to include the structural equations of motion for their simultaneous time-integration with the governing fluid flow equations. Detailed comparisons of results obtained using a structured grid Euler code (CFL3D run in a 2D mode) with the results obtained using the present Euler code are used to determine the accuracy of the unstructured grid methodology for flutter applications.

Accomplishment Description: To assess the accuracy, generalized force computations and aeroelastic analyses were performed for the NACA 0012 airfoil. Shown in the upper part of figure 34b is a partial view of the unstructured grid that was used. Results were first obtained for the airfoil pitching harmonically about the quarter chord at $M_\infty = 0.8$ and $\alpha = 0°$. As shown in the lower left part of the figure, the lift coefficient due to pitching, C_{ℓ_α} agrees well between the different methods for the entire range of reduced frequency k that was considered, which verifies the unstructured grid flow solver for generalized force computation. Aeroelastic results were also obtained for a two degree-of-freedom (pitch and plunge) NACA 0012 airfoil at $M_\infty = 0.8$ and $\alpha = 0°$. Comparisons of second mode generalized displacement (q_2) are shown in the lower right part of the figure for two values of non-dimensional dynamic pressure ($\bar{Q}$) that bracket the flutter point.

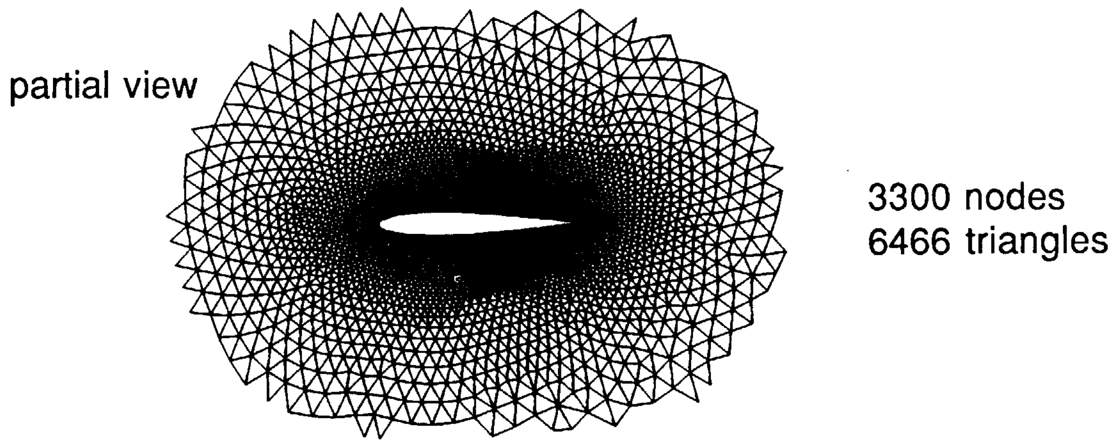
Significance: The generalized displacements agree well with the structured grid (CFL3D) solution which verifies the unstructured grid methodology for aeroelastic analysis. The flutter speed for this case, determined by interpolation of the dominant damping of these generalized displacements, also agrees to within 2% of the CFL3D value.

Future Plans: The unstructured grid Euler aeroelasticity capability will be applied to additional cases to further validate the methodology. Extensions to treat three-dimensional configurations will be developed and aeroelastic analyses will be performed for complete aircraft geometries.

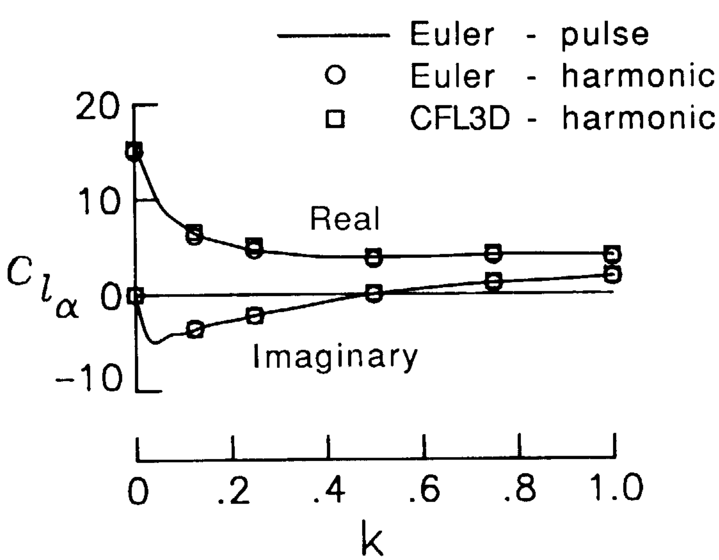
Figure 34 (a).

EULER FLUTTER ANALYSIS OF AIRFOILS DEMONSTRATED USING UNSTRUCTURED DYNAMIC MESHES

- **Unstructured grid of triangles about NACA 0012 airfoil**



- **Lift coefficient comparison $M_\infty = 0.8$ and $\alpha = 0°$**



- **Aeroelastic transient comparison $M_\infty = 0.8$ and $\alpha = 0°$**

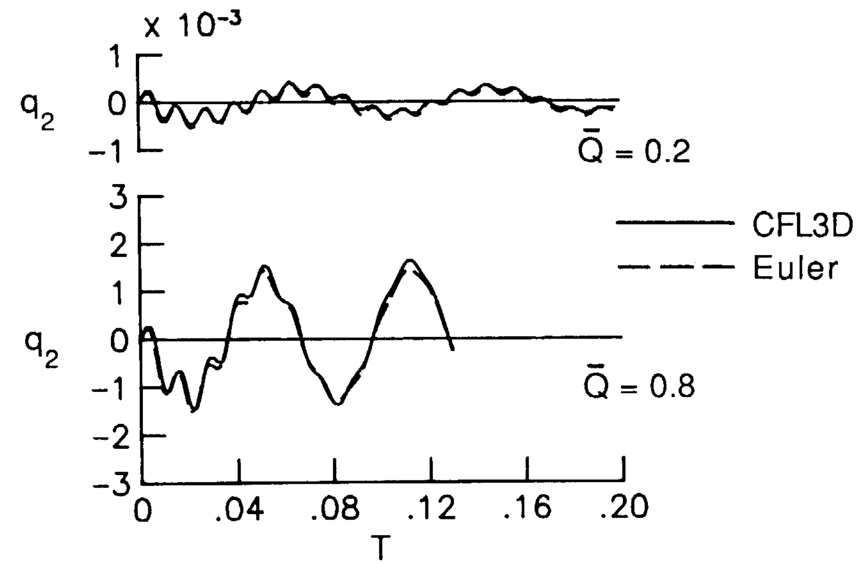


Figure 34 (b).

CLIPPED DELTA CANARD TRANSONIC FLOWFIELD VISUALIZED

David A. Seidel, Bryan E. Dansberry and Clinton V. Eckstrom
Unsteady Aerodynamics and Configuration Aeroelasticity Branches

RTOP 505-63-21

Research Objective: The objective of this research is to define the steady and unsteady subsonic and transonic flow characteristics on a clipped delta canard for use in validating analytical methods now under development. Flow characteristics are to be described by both flowfield visualization techniques and by the measurement of steady and unsteady pressures on the canard.

Approach: A two-step approach was taken to define the steady and unsteady flow characteristics on a clipped delta canard. The first step was to fabricate an uninstrumented canard, splitter plate and actuator hardware to oscillate the canard in pitch. A wind tunnel test in the NASA Langley Transonic Dynamics Tunnel (TDT) would be conducted to develop the techniques required to visualize the steady and unsteady flowfields at subsonic and transonic conditions. The second step would be to fabricate an instrumented canard. This canard would then be tested in the TDT and steady and unsteady pressures on the canard would be measured in addition to the steady and unsteady flowfields about the canard.

Accomplishment Description: An uninstrumented canard, a splitter plate and actuator hardware were fabricated and installed in the TDT. The canard had a leading edge sweep of 50.5° and a panel aspect ratio of 1.24. The canard airfoil section was a 6% thick circular arc. A picture of the canard and splitter plate installed in the TDT is found on figure 35b. A wind tunnel test was conducted to develop techniques to visualize the steady and unsteady flowfields about the canard at subsonic and transonic conditions. Smoke particles were injected into the flowfield directly underneath the canard leading edge to be entrained in the canard's leading edge vortex and were illuminated using a laser light sheet. The flowfield was visualized for a wide range of conditions, with Mach number varying from 0.4 to 0.92, steady angle of attack from 5° to 25°, oscillation amplitude up to ±9.5° and frequencies up to 6.67 Hz. The next figure contains examples of the steady and unsteady flowfield visualization accomplished during the test. A multiple exposure photograph shows the steady vortex on the canard at a Mach number of 0.92 and an angle of attack of 15°. The light sheet is perpendicular to the freestream flow and located at 25%, 50%, 75% and 100% root chord. Also shown in the figure are four photographs taken while the canard was oscillating in pitch at a Mach number of 0.92. The canard was oscillating ±5° about a mean angle of attack of 15° at a frequency of 3.35 Hz. The laser light sheet was perpendicular to the freestream flow and located at 50% root chord. The light sheet was strobed to illuminate the flowfield when the canard angle of attack was at 15° (angle increasing), 20°, 15° (angle decreasing) and 10°.

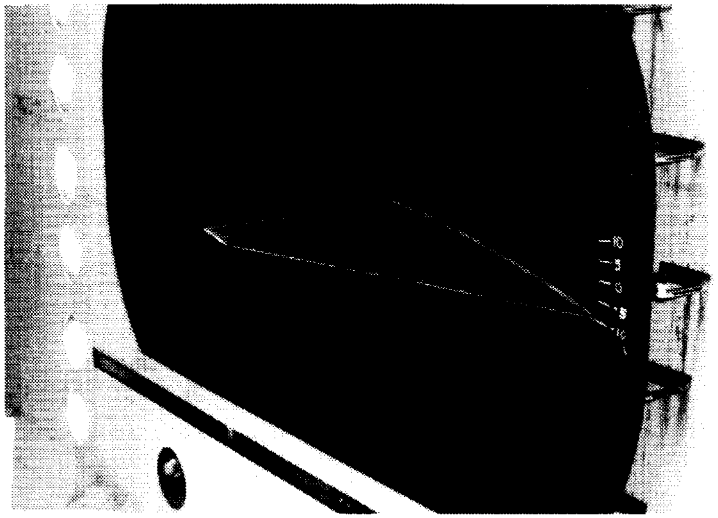
Significance: Procedures have been developed to visualize the steady and unsteady flowfield about a clipped delta canard for subsonic to transonic conditions. Future tests will be able to correlate measured surface pressures with changes in the flowfield about a lifting surface.

Future Plans: The instrumented canard is being fabricated. A test will be run in the TDT where steady and unsteady pressures will be measured on the instrumented canard in addition to the steady and unsteady flowfields about the canard.

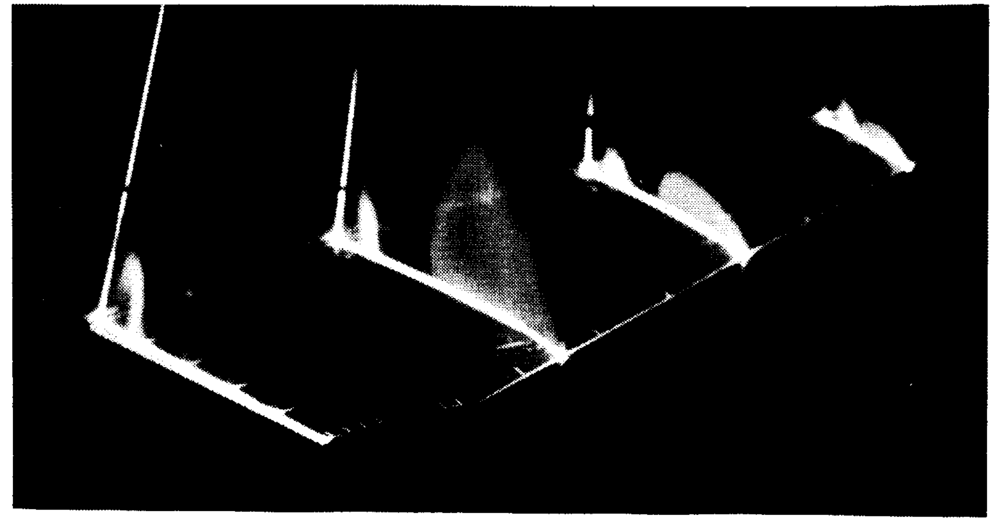
Figure 35 (a).

CLIPPED DELTA CANARD TRANSONIC FLOWFIELD VISUALIZED

Canard in TDT

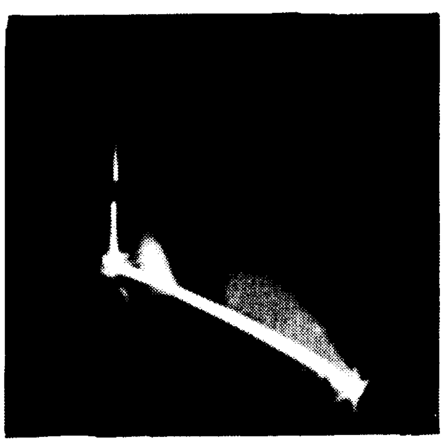


$M=0.92$ $\alpha=15^\circ$

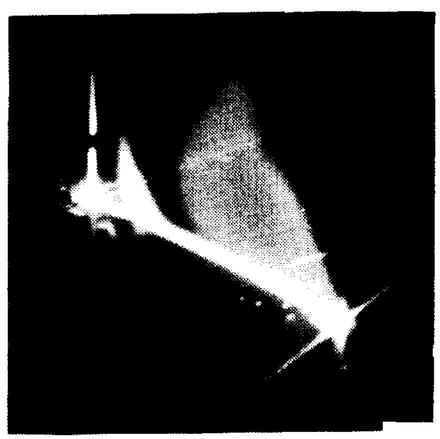


$M=0.92$ $a=15^\circ \pm 5^\circ$ $f=3.35Hz$

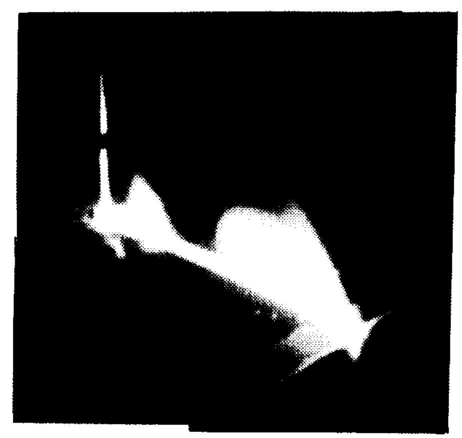
$\alpha=15^\circ$ ($\uparrow$)



$\alpha=20^\circ$



$\alpha=15^\circ$ ($\downarrow$)



$\alpha=10^\circ$

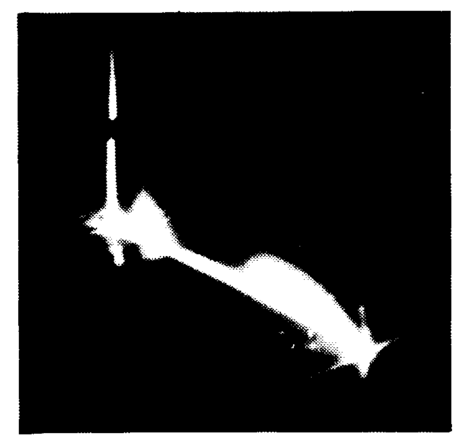


Figure 35 (b).

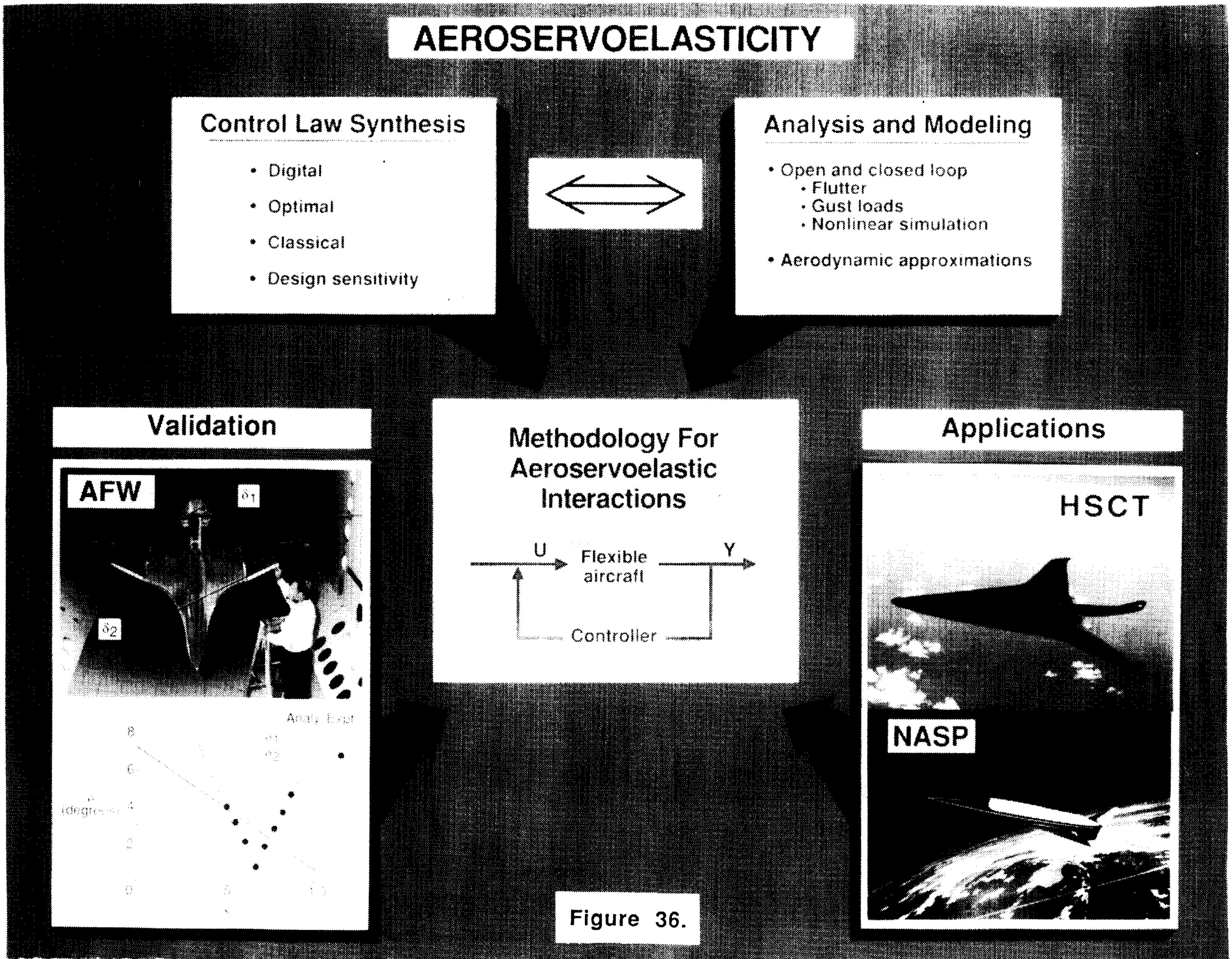

AEROSERVOELASTICITY
Control Law Synthesis
• Digital
• Optimal
• Classical
• Design sensitivity
Analysis and Modeling
• Open and closed loop
• Flutter
• Gust loads
• Nonlinear simulation
• Aerodynamic approximations
Methodology For Aeroservoelastic Interactions
U
Flexible aircraft
Y
Controller
Validation
AFW
Applications
HSCT
NASP

Figure 36.

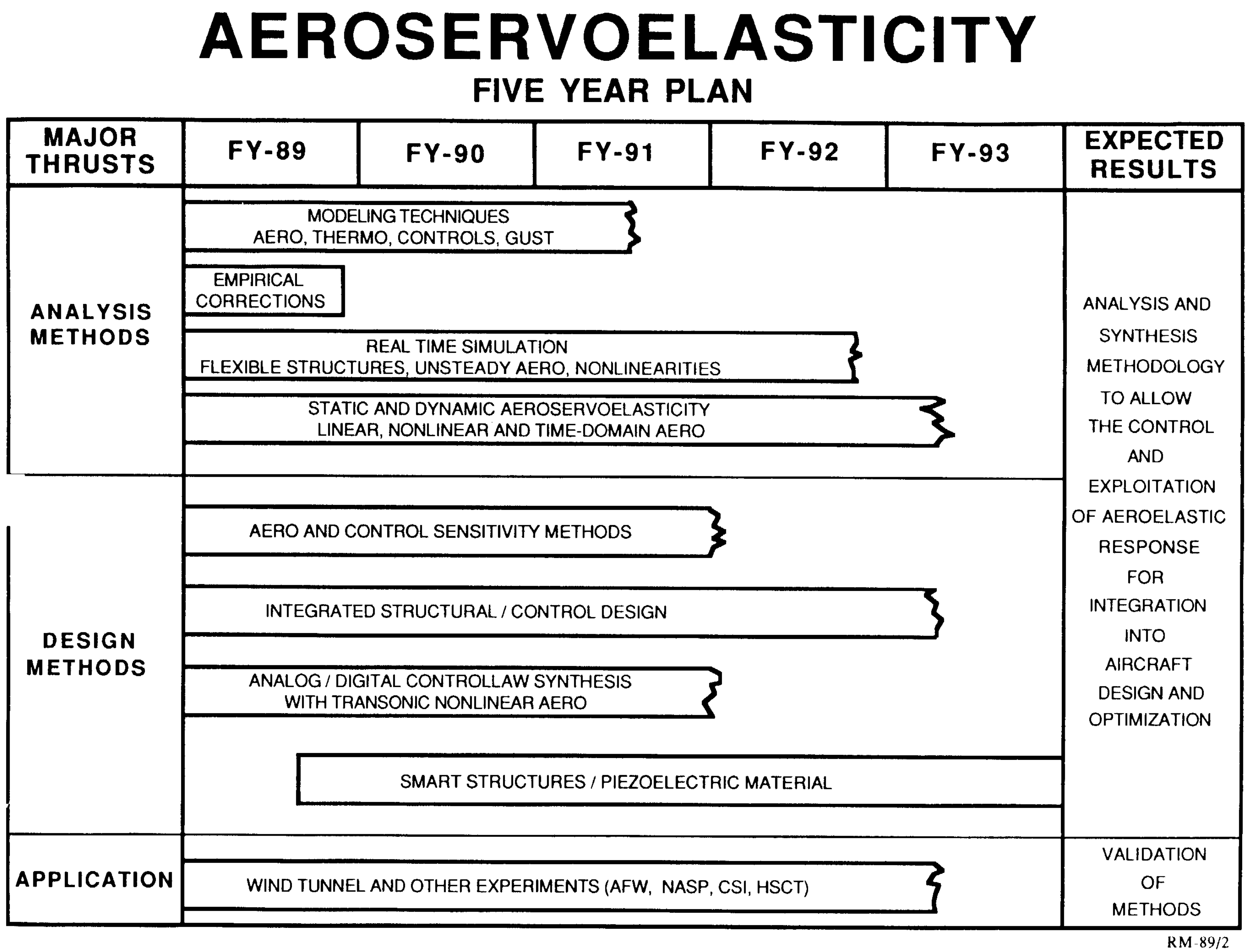

AEROSERVOELASTICITY
FIVE YEAR PLAN
MAJOR THRUSTS
FY-89
FY-90
FY-91
FY-92
FY-93
EXPECTED RESULTS
ANALYSIS METHODS
MODELING TECHNIQUES
AERO, THERMO, CONTROLS, GUST
EMPIRICAL CORRECTIONS
REAL TIME SIMULATION
FLEXIBLE STRUCTURES, UNSTEADY AERO, NONLINEARITIES
STATIC AND DYNAMIC AEROSERVOELASTICITY
LINEAR, NONLINEAR AND TIME-DOMAIN AERO
DESIGN METHODS
AERO AND CONTROL SENSITIVITY METHODS
INTEGRATED STRUCTURAL / CONTROL DESIGN
ANALOG / DIGITAL CONTROLLAW SYNTHESIS
WITH TRANSONIC NONLINEAR AERO
SMART STRUCTURES / PIEZOELECTRIC MATERIAL
ANALYSIS AND SYNTHESIS METHODOLOGY TO ALLOW THE CONTROL AND EXPLOITATION OF AEROELASTIC RESPONSE FOR INTEGRATION INTO AIRCRAFT DESIGN AND OPTIMIZATION
APPLICATION
WIND TUNNEL AND OTHER EXPERIMENTS (AFW, NASP, CSI, HSCT)
VALIDATION OF METHODS
RM-89/2

Figure 37.

NOVEL APPROACH DEVELOPED FOR DETERMINING MAXIMIZED AND TIME-CORRELATED GUST LOADS

Anthony S. Pototzky (Lockheed), Dr. Thomas A. Zeiler (Lockheed), and Boyd Perry III
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective: The objective of this project is to apply Matched Filter Theory (MFT) to linear aeroelastic systems to compute maximized and time-correlated gust loads.

Approach: MFT was originally applied in the field of signal processing for the design of an electrical filter that maximized the detection of a returning radar signal. In the current application to linear aeroelastic systems, MFT is used to "design" a critical gust pattern (a time history of vertical gust velocity) that produces (1) the worst-case deterministic response of a chosen load quantity and (2) time-correlated responses of other load quantities. The accompanying figure 38b shows, in the upper left, a critical gust pattern produced by MFT. It has a finite probability of occurring in turbulence described by the von Karman power spectral density function. The critical gust pattern is then applied analytically to the flexible airplane depicted in the center of the figure. Dynamic loads at locations 1 and 2 on the wing are computed in response to the critical gust pattern and are shown on the right side of the figure. MFT guarantees that, in response to all other equiprobable gust patterns, the value of Load 1 at any value of time will never exceed the maximum value shown in the figure.

Accomplishment Description: This approach has been presented at a variety of national and international technical meetings and conferences. Two NASA Technical Memoranda and an AIAA paper have been written, and oral presentations have been made at the TTCP HAG-6 Workshop on Active Controls and Structural Integrity, the Fall 1988 and the Spring 1989 meetings of the Aerospace Flutter and Dynamics Council, and the 1989 AIAA Structures, Structural Dynamics, and Materials Conference.

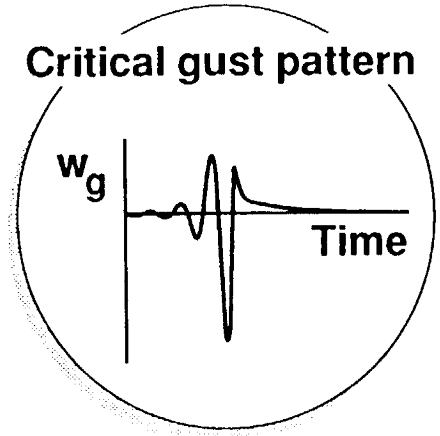
Significance: Aircraft manufacturers have recently encouraged the FAA to investigate alternate means of compliance with Federal Aviation Regulation 25.305(d), which relates to aircraft gust loading. The time responses calculated with MFT are proportional to correlation functions of Random Process Theory obtainable directly from response spectra. Both of these approaches provide new analysis options for computing time-correlated dynamic loads in aircraft that are also computationally fast. In addition to computing time-correlated gust loads, the two theories may be employed to compute time-correlated taxi, landing, and/or maneuver loads.

Future Plans: A means has been identified and plans exist to extend the current approach to aeroelastic systems with embedded nonlinearities.

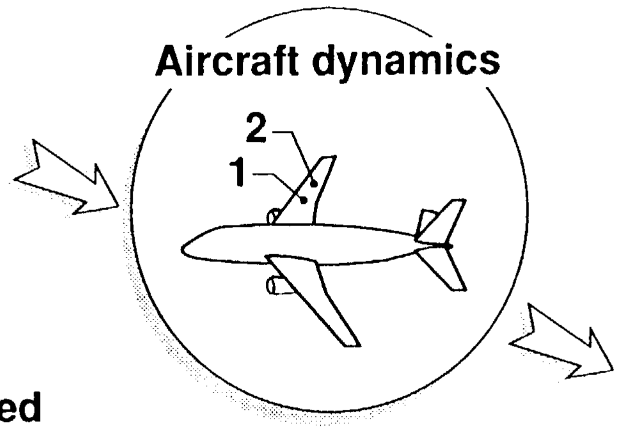
Figure 38 (a).

C-2

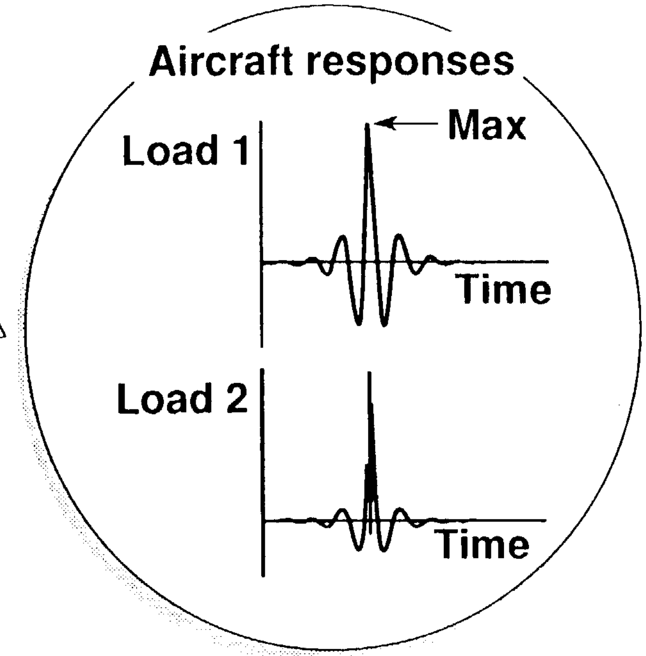
NOVEL APPROACH DEVELOPED FOR DETERMINING MAXIMIZED AND TIME-CORRELATED GUST LOADS



- Obtained from matched filter theory
- Consistent with Von Karman turbulence
- Maximizes Load 1 (worst-case gust for Load 1)



- Aeroelastic equations of motion
- Linear system



- Load 1 maximized
- Load 2 time-correlated
- Computationally efficient

Figure 38 (b).

NONLINEAR METHOD PREDICTS STATIC AND RESULTANT DYNAMIC AEROELASTIC BEHAVIOR

Walter A. Silva (Lockheed)
Aeroservoelasticity Branch
and
Robert M. Bennett
Unsteady Aerodynamics Branch

RTOP 505-63-21

Research Objective: The inability to accurately predict the transonic flutter boundary of realistic aircraft configurations is well known. The objective of the present work is to develop a methodology for determining the effects of nonlinear transonic aerodynamics on the combined static & dynamic aeroelastic behavior of complex flight vehicles.

Approach: A computational model (lower left of figure 39b) for the complete Active Flexible Wing (AFW) wind-tunnel model was developed for the CAP-TSD code. The analysis procedure first determines the static aeroelastic shape of the AFW configuration resulting from certain flight (Mach number and dynamic pressure) and model (angle-of-attack and control surface angles) conditions (upper left sketch). Dynamic perturbation analyses are then performed about the deformed shape to obtain the flutter characteristics (upper right sketch).

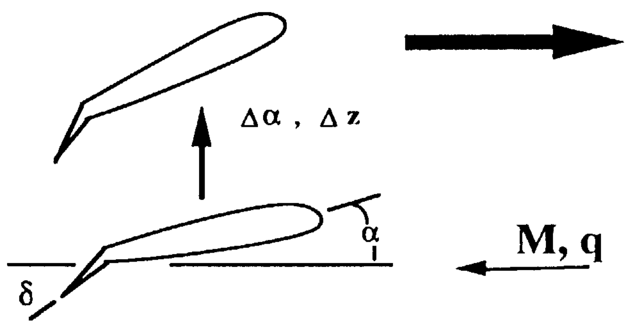
Accomplishment Description: A procedure was developed to analyze situations where static aeroelasticity is important (nonsymmetric airfoils or symmetric airfoils at non-zero angles-of-attack) in determining the dynamic stability at transonic flight conditions. For example, the steady-state aerodynamics of a nonsymmetric airfoil produces a transient response that if not handled properly can yield misleading dynamic response results. As part of the analysis procedure artificially large values of damping are used to accelerate the convergence to the static aeroelastic shape. The results of transonic flutter calculations about this static aeroelastic shape for the AFW model at 1.5 degrees angle-of-attack (previously tested condition which minimizes wing loads) are shown at the lower right in the figure. The results indicate the presence of a transonic flutter "dip" near $M = 0.95$. A comparison of linear and nonlinear CAP-TSD results at $M = 0.9$ reveals a difference in flutter dynamic pressure due to the effects of nonlinear aerodynamics. The existence of a shock at $M = 0.95$ limits the use of linear aerodynamic theories to Mach numbers not much greater than $M = 0.9$.

Significance: This study identified the importance of static aeroelasticity on the dynamic aeroelastic behavior of flight vehicles at transonic speeds. The study also addressed the important nonlinear aerodynamic effects of vehicle components, such as the fuselage and wing-tip stores, on aeroelastic characteristics.

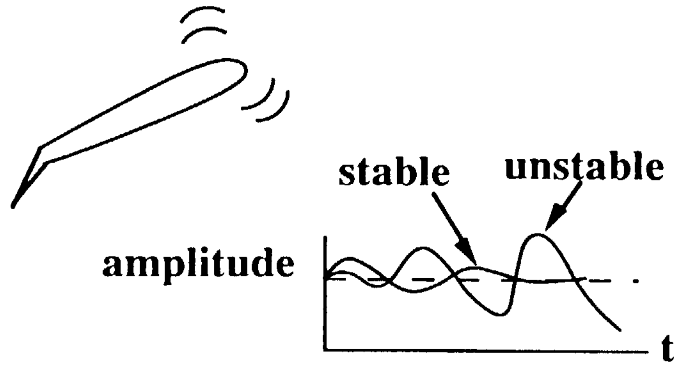
Future Plans: These analytical results will be compared to wind-tunnel data obtained in 1989. The results of these investigations will be presented at the AIAA Structures, Structural Dynamics, and Materials Conference in April 1990 and at the International Congress of the Aeronautical Sciences Meeting in September 1990.

Figure 39 (a).

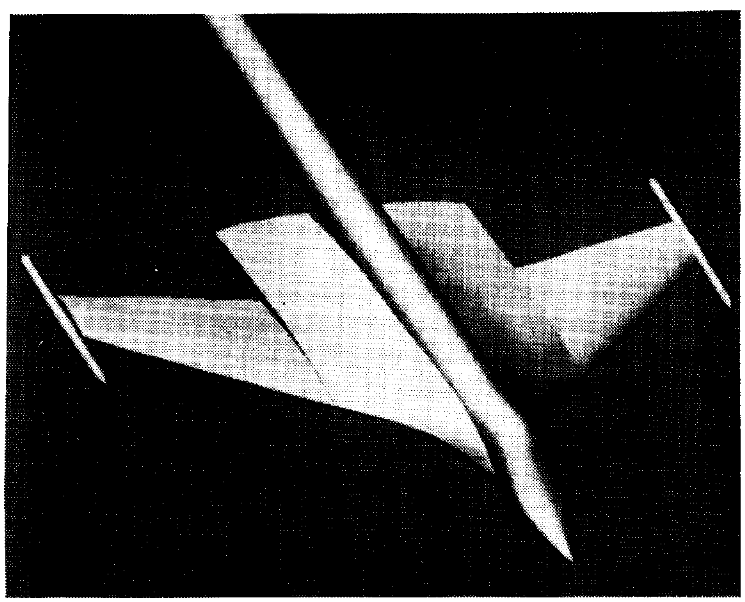
NONLINEAR METHOD PREDICTS STATIC AND RESULTANT DYNAMIC AEROELASTIC STABILITY



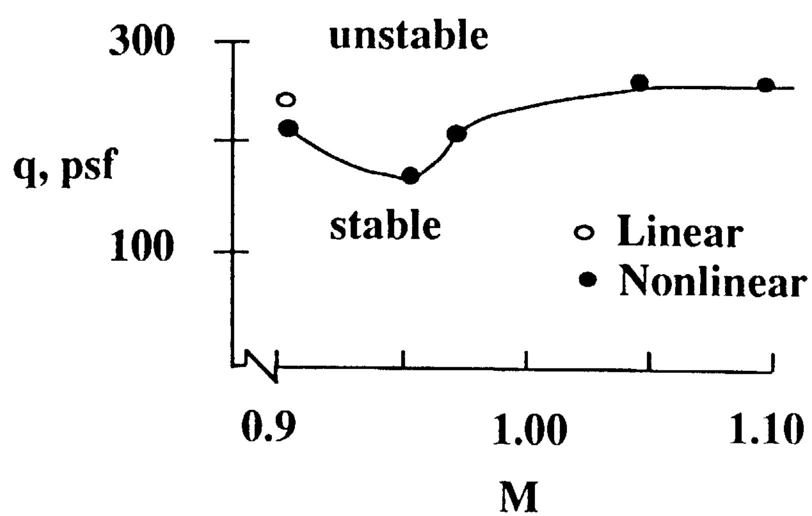
Static Aeroelastic Solution



Dynamic Perturbation



CAP-TSD Computational Model



Flutter Boundary

Figure 39 (b).

HYPERSONIC AEROSERVOELASTICITY METHOD REDUCES THERMAL HEATING EFFECTS ON FLUTTER

Michael G. Gilbert, Jennifer Heeg, and Anthony S. Pototzky (Lockheed)
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective: The objectives of this research are to develop appropriate aerothermoelastic (ATE) analysis methods and apply aeroservoelastic (ASE) technology to reduce adverse aeroelastic changes caused by the aerodynamic heating.

Approach: Atmospheric flight at high speeds causes large thermal loads due to aerodynamic heating. These large thermal loads can soften (destiffen) the structure through changes in structural material properties with temperature and through material stress level changes caused by thermal gradients in built-up structural components. The more flexible structure can significantly affect the vehicle flutter characteristics and aeroelastic responses. As shown in figure 40b, an analysis procedure has been developed to evaluate these concerns. For a given set of thermal loads (radiation equilibrium temperatures) determined for a particular aircraft geometry, the hot and cold vehicle structural stiffnesses are obtained using a finite element structural model. Structural vibration modes for the hot and cold structure are computed and used with Piston Theory unsteady aerodynamics. Detailed flutter analyses are performed for both the hot and cold structures, and compared to determine the changes in flutter dynamic pressure due to the thermal loads. Then, existing ASE design methods are used to recover the lost flutter dynamic pressure.

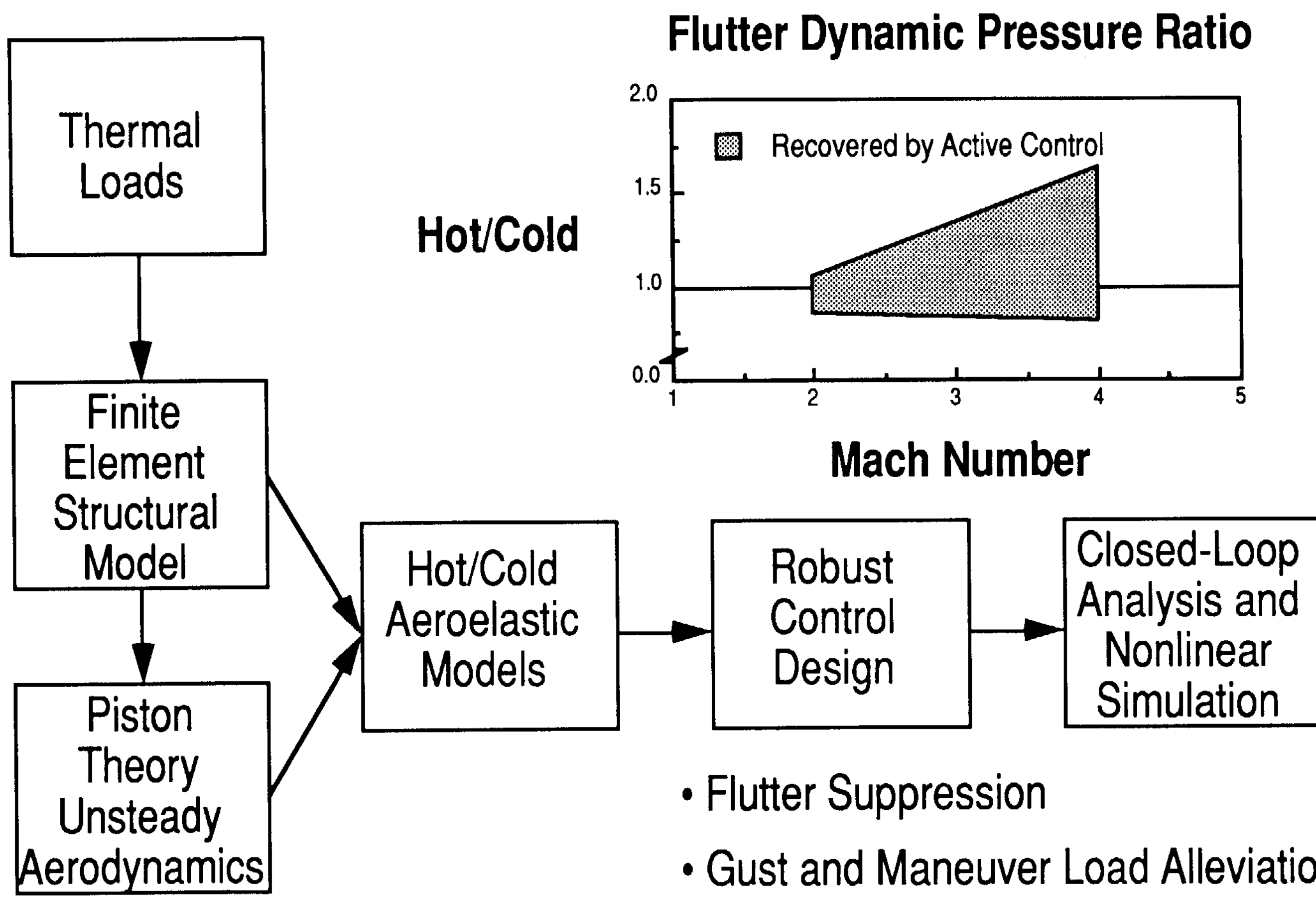
Accomplishment Description: The ATE analysis methodology has been applied to a generic hypersonic aircraft configuration. This configuration consists of an elliptical cross section body with delta wings, flaps, and ailerons. A finite element model structural representation was developed for the configuration using Titanium Aluminide structural material properties. Radiation equilibrium temperature distributions were obtained for a Mach 4.0 flight condition and applied to the finite element model to determine hot structure vibration mode shapes. Four sets of unsteady generalized aerodynamic forces were computed, Mach 2.0 cold, Mach 4.0 cold and hot, and a Mach 2.0 hot using the Mach 4.0 temperature distribution simulating a vehicle slowing from Mach 4.0. The effects of thermal loads on flutter are a 25-30% reduction in flutter dynamic pressure. A flutter suppression active control law was designed to recover the lost flutter dynamic pressure. This control law used wing and fuselage acceleration feedback signals to move the wing flap and aileron to suppress the flutter. The improvement in flutter dynamic pressure was dramatic using the same control law at both Mach 2.0 and 4.0.

Significance: The application of active control technology to hypersonic aircraft can be useful for overcoming adverse structural response and flutter characteristics caused by aerodynamic heating of the vehicle structure. Since a single control law was found to improve flutter dynamic pressure over a relatively large Mach range, complex, highly gain-scheduled control laws may not be necessary.

Future Plans: The ATE analysis will be modified to find converged static aerothermoelastic vehicle shapes using an iterative process. Active control law functions such as ride quality improvement and gust load alleviation will be investigated.

Figure 40 (a).

HYPERSONIC AEROSERVOELASTICITY METHOD REDUCES THERMAL HEATING EFFECTS ON FLUTTER



- Flutter Suppression
- Gust and Maneuver Load Alleviation
- Ride Quality

Figure 40 (b).

MULTILEVEL METHOD INTEGRATES INDEPENDENT STRUCTURE AND CONTROL LAW DESIGNS

Michael G. Gilbert
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective: The objective is to develop an integrated structure/control law design capability applicable to both air and space vehicles. The goal is to permit the use of independent structure and control law design methods, models, and techniques, with structure/control law integration achieved by means of an interdisciplinary design coordination process.

Approach: The approach to integrated structure/control law design is based on hierarchal problem decomposition, formal optimization techniques, and design sensitivity analyses methods. As shown in figure 41b, the integrated design problem is decomposed into multiple levels. At the lower level, the independent structure and control law designs are obtained; at the top level, the design coordination process occurs. The integrated design requirements are formulated in the top level as functions of a set of design integration parameters which have assigned values. The independent structure and control law designs are obtained based on the design integration parameters. Following the lower level designs, the sensitivity of those designs to the integration parameters is determined and passed to the top level. At the top level, the performance of the interconnected structure and control law is evaluated with respect to the integrated design requirement and new values of the integration parameters are selected. The lower level design sensitivity information is used as gradient information at the top level in a formal optimization to find the new integration parameter values.

Accomplishment Description: Although sensitivity analysis methods for optimal structural designs have been available for some time, a similar technique for optimal Linear Quadratic Gaussian control law designs has been just recently developed. With this necessary development step completed, the integrated design methodology was applied to a simple two-bar truss problem from literature. The integrated design requirement was selected to be the minimization of the lateral deflection of the truss subject to a random disturbance. The control force application angle and bar 1 weight were chosen as the integration parameters. The cross-sectional area of bar 1 (a function of the weight of bar 1) was selected as the structural design variable, and the control force (a function of the application angle) was determined by the control law design. The integrated design method then determines the allowable design which simultaneously has the smallest weight, control force, and lateral deflection.

Significance: The significance of this integrated design method is that the structural and control law designs are accomplished independently using separate methods, models, and expertise. This eliminates the need for a single, large, often intractable design problem. Design integration is instead achieved through specification of design integration parameters, which are used to tailor the lower level designs to meet the integrated design requirements.

Future Plans: The methodology developments are being extended to handle more complicated problems from the large space structure controls literature so that direct comparisons of various integrated design algorithms can be made. This effort entails a more sophisticated upper level optimization problem formulation and improved data interchange between levels.

Figure 41 (a).

MULTILEVEL METHOD INTEGRATES INDEPENDENT STRUCTURE AND CONTROL LAW DESIGNS

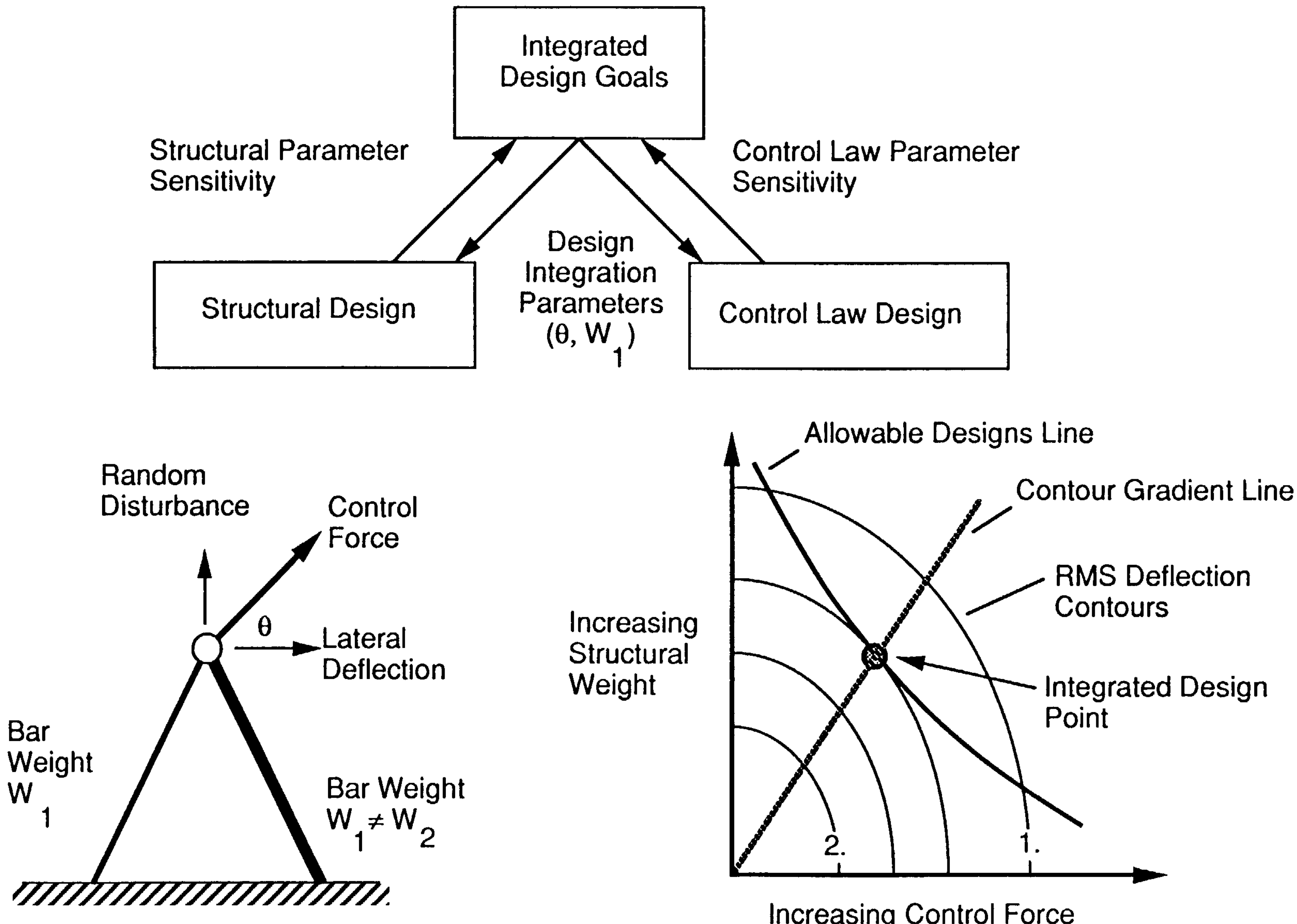


Figure 41 (b).

ADVANCED INTEGRATED STRUCTURE/CONTROL DESIGN METHOD IMPROVES SPACE STRUCTURE SYSTEM PERFORMANCE

Dr. Thomas A. Zeiler (Lockheed) and Michael G. Gilbert
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective: Existing aerospace vehicle design methodologies do not take maximum advantage of multidisciplinary interactions. The present research objective is to develop a design methodology that integrates structural design and active control design.

Approach: The integrated design methodology is based on hierarchical multilevel problem decomposition and optimization techniques. The actively controlled structure is decomposed into its structural and control subsystems as shown in figure 42b. At the topmost, integrated level, a system measure is optimized with structural and control parameters as design variables. The structural and control subsystems are optimized using design variables and methods suitable to their respective disciplines, but with those parameters specified at the top level held constant. Sensitivity derivatives, with respect to the parameters, of these optimum solutions are used in the top level optimization as gradient information.

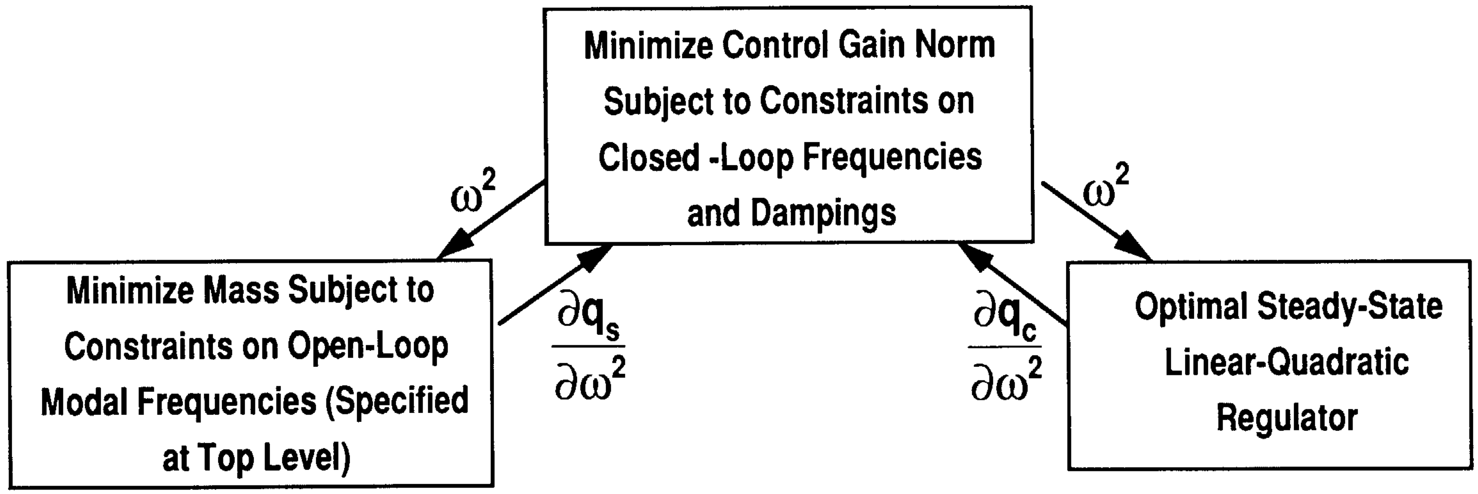
Accomplishment Description: The method was tested on an ACOSS-4 (shown in the figure) design problem existing in the open literature. Although not identified as such, that benchmark solution was a multilevel solution of sorts with cross-sectional areas of the truss members as top level design variables and a Linear Quadratic Regulator (LQR) and associated sensitivities determined for each iteration. There was, however, no decomposition of the structure. In the present approach, the design variables are a subset of the structural eigenvalues (frequencies squared). The truss member cross-sectional areas and associated sensitivities are determined at the lower level. The present method produced a value for the gain norm, which was the system level objective, similar to that obtained in the benchmark solution. However, because of the direct influence exerted upon the structural mass at the structure subsystem level, the final mass was substantially lower.

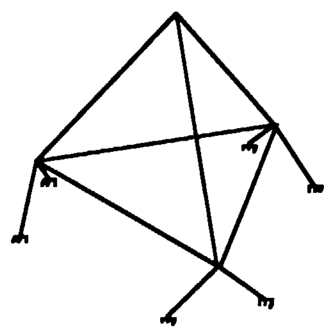
Significance: The multilevel architecture has been shown to allow greater disciplinary control over subsystem design yet permit the effects of subsystem interactions to influence the overall system design.

Future Plans: Presently, sensitivities with respect to the structural eigenvalues are being used. Plans are to prepare a paper summarizing these results and present it at the AIAA Structures, Structural Dynamics, and Materials Conference in 1990. Future work will address sensitivities of other quantities such as mode shapes and sensitivities with respect to control parameters.

Figure 42 (a).

ADVANCED INTEGRATED STRUCTURE/CONTROL DESIGN METHOD IMPROVES SPACE STRUCTURE SYSTEM PERFORMANCE





ACOSS - 4

System Measure	Grandhi et al.	Present Approach
Gain Norm	18.431	18.241
Mass	87.834	49.933

(quantities are non-dimensional)

Figure 42 (b).

LOW-ORDER DIGITAL FLUTTER SUPPRESSION CONTROL LAWS DESIGNED FOR AFW WIND-TUNNEL MODEL

Dr. Vivekananda Mukhopadhyay
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective: The objective of this effort is to design low-order digital flutter suppression system (FSS) control laws for the Active Flexible Wing (AFW) wind-tunnel model using optimal control theory and order-reduction techniques, and to validate the design through simulation and wind-tunnel tests using controller performance evaluation procedures.

Approach: The design procedure uses a continuous, time domain state-space model to initially obtain full-order, minimum energy, optimal FSS control laws based on linear-quadratic Gaussian (LQG) theory. The objective of the FSS control law is to stabilize, simultaneously, both a symmetric and an antisymmetric flutter mode of the AFW model. The effect of antialiasing filters and time delays associated with the digital controller are taken into account at this stage in the design process. The most promising control laws are selected (see figure 43b) and reduced in order using balanced realization and residualization techniques. The next step in the design process is to evaluate the reduced-order control laws from a stability robustness and performance point of view in the digital domain. Nonlinear batch and near real-time hot-bench simulations are then used to evaluate the control system, and to check the functionality of the digital controller hardware. Finally wind-tunnel test demonstrations are conducted to obtain data for validating the design methodology.

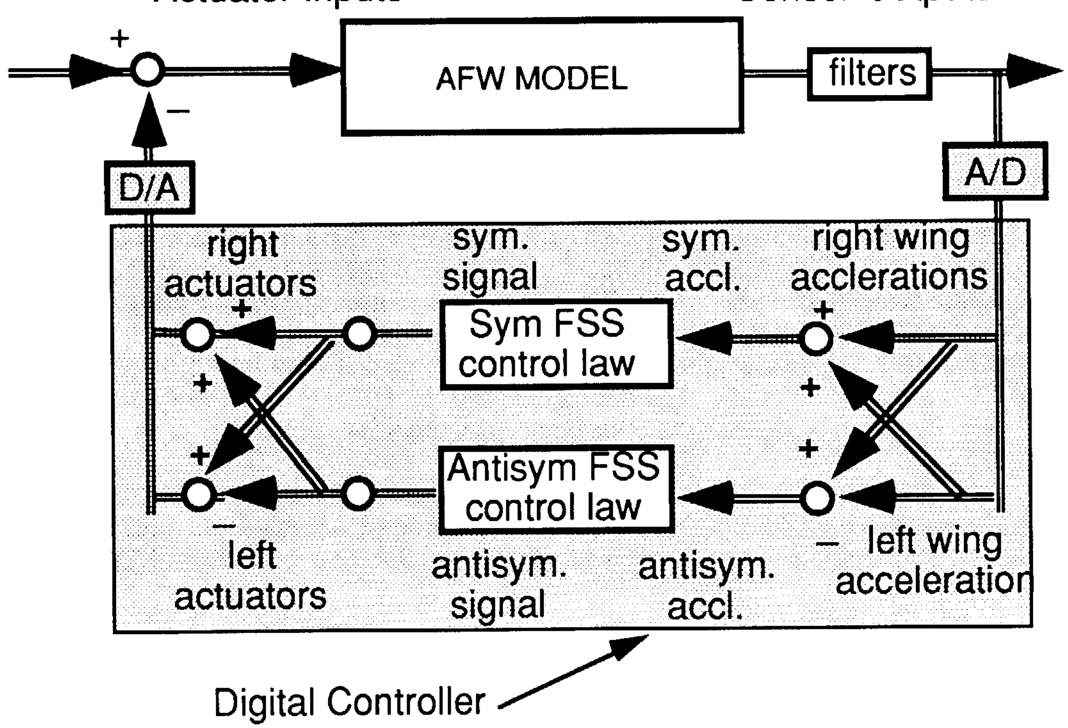
Accomplishment Description: Low-order symmetric and antisymmetric flutter suppression control laws were designed for testing in air at Mach 0.5, sea level conditions. The control laws were predicted to increase the theoretical flutter dynamic pressure by 30 percent, while maintaining established control surface deflection and rate limits and reasonable gain and phase margin requirements. A controller performance evaluation procedure based on using sine-sweep inputs to the control surfaces (symmetrically and antisymmetrically) and subsequent plant and controller responses was also developed for implementation during the wind-tunnel test investigations. Open- and closed-loop data were obtained in the wind tunnel at speeds below the flutter condition for correlation with predictions.

Significance: The testing of an FSS control law on the AFW wind-tunnel model will result in the measurement of data using an advanced, highly flexible aircraft configuration for validating and improving digital, multi-input/multi-output control law design methodologies.

Future Plans: Recent wind-tunnel test data will be reduced and evaluated with the intent of improving the state-space model of the unaugmented plant, of correcting aerodynamic stability derivatives of the various control surfaces, and of enhancing the design methodology associated with the development of multifunction active control systems to highly flexible structures. The results of these investigations will be presented at the AIAA Structures, Structural Dynamics, and Materials Conference in April 1990 and at the International Congress of the Aeronautical Sciences Meeting in September 1990.

Figure 43 (a).

LOW-ORDER DIGITAL FLUTTER SUPPRESSION CONTROL LAWS DESIGNED FOR AFW WIND-TUNNEL MODEL



- o Low order FSS designed at Mach 0.5
- o Flutter dynamic pressure increased by 20%
- o Control surface limitations and stability margin requirements satisfied
- o Controller performance evaluation procedure developed
- o Nonlinear batch and hot-bench simulation conducted

Figure 43 (b).

REAL-TIME PROCEDURES DEVELOPED FOR EVALUATING THE PERFORMANCE AND STABILITY OF DIGITAL, MULTIVARIABLE CONTROLLERS

Anthony S. Pototzky (Lockheed), Carol D. Wieseman, Vivek Mukhopadhyay,
Boyd Perry III, and Matthew E. Fox (MIT)
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective: The objective of this effort is to develop a Controller Performance Evaluation (CPE) methodology capable of providing real-time guidance while evaluating the digital control systems tested on the Active Flexible Wing (AFW) wind tunnel model, thereby improving model and wind tunnel safety.

Approach: The methodology was derived using multivariable control theory and implemented on a series of microcomputers. The CPE procedure (see figure 44b) systematically excites the wind-tunnel model at test conditions using the various control surfaces in a symmetric or antisymmetric manner. The resulting excitations and time responses at the plant output (wind-tunnel model accelerometers signals) and at the controller output (control surface command signals) locations are transformed into the frequency domain using Fast Fourier Transforms programmed on a Sun computer upgraded with a Sky Warrior board (high-speed array processor). From these frequency domain responses, auto-spectra of the excitations and cross-spectra between the excitations and the responses are also computed on the SUN/Warrior computer. Matrix procedures, written in MATLAB software, are used to analytically compute the transfer matrices of the open-loop controller and plant, and to compute the return-difference matrix of the closed-loop system. Performance and stability are determined by evaluating the singular values of the return-difference matrices at the plant input and output points. The CPE procedure can be used to directly determine closed-loop performance and stability or to predict closed-loop characteristics from responses taken while the control system is open loop.

Accomplishment Description: During recent AFW wind-tunnel tests, the CPE procedure was very instrumental in establishing the performance and stability margins of active flutter suppression and rolling maneuver load alleviation digital control systems, and was very useful in estimating plant and controller transfer matrices for comparisons with predictions.

Significance: The CPE procedure proved to be a valuable tool; it identified potentially destabilizing controllers before actually closing the loop on a control system, thereby avoiding catastrophic damage to the model or wind tunnel. The CPE procedure was also valuable in computing open-loop controller and plant transfer matrices which were used to check out and improve existing control system designs.

Future Plans: The CPE repertoire of methods will be enhanced using modern signal processing techniques to improve its capability and speed of estimating transfer functions. In addition, CPE techniques will be evaluated as a basis for developing adaptive control system design methodology for application to flexible flight vehicles. The research will be presented at a Work-in-Progress session at the AIAA Structures, Structural Dynamics, and Materials Conference in April 1990. A full paper will be presented later in the year.

Figure 44 (a).

REAL-TIME PROCEDURES DEVELOPED FOR EVALUATING THE PERFORMANCE AND STABILITY OF DIGITAL, MULTIVARIABLE CONTROLLERS

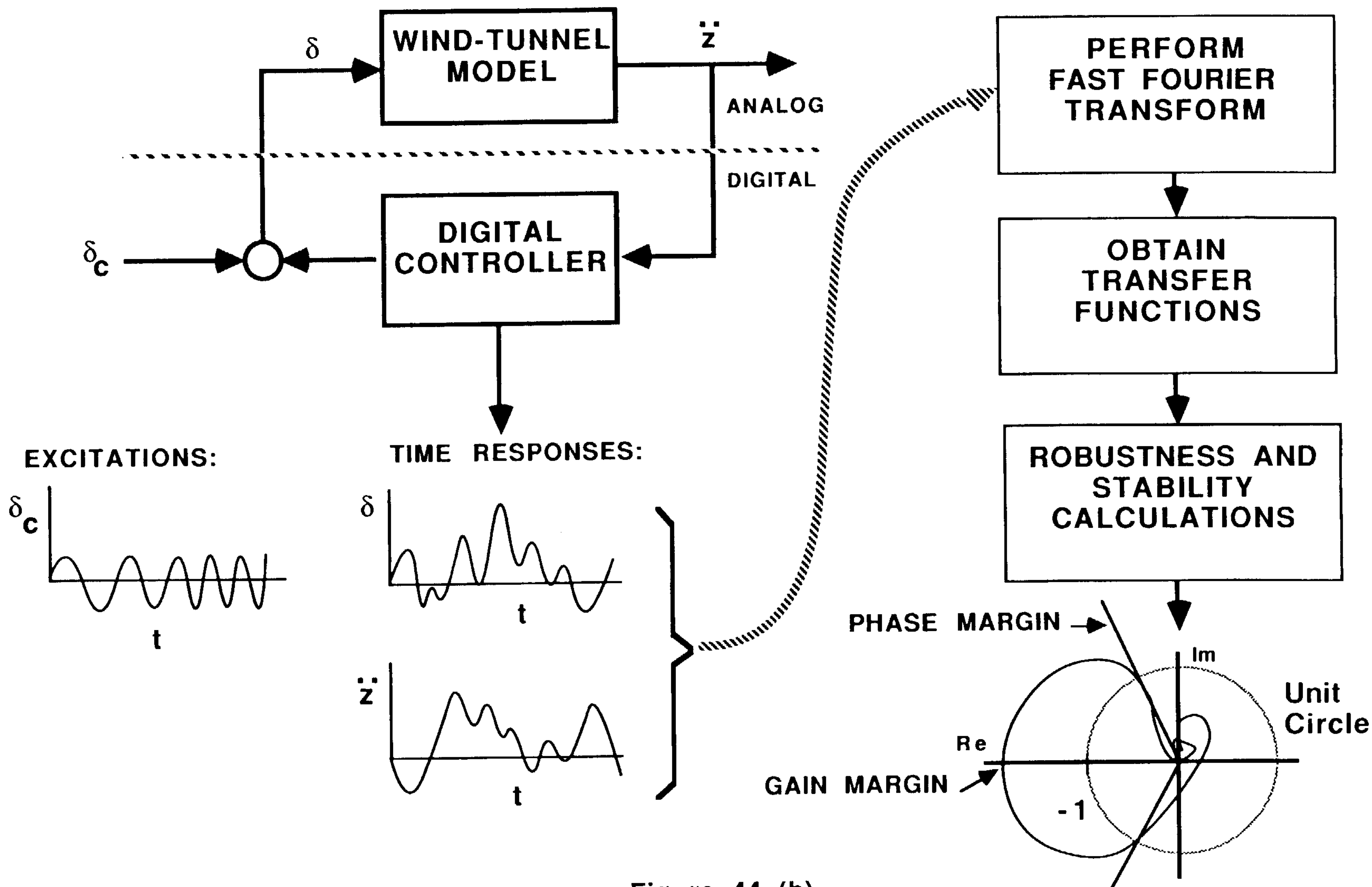


Figure 44 (b).

AFW DIGITAL CONTROLLER DESIGNED, ASSEMBLED, CODED, VALIDATED, AND TESTED

Sherwood T. Hoadley, Sandra M. McGraw (Lockheed), and Boyd Perry III
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective: The Active Flexible Wing (AFW) Program is a joint NASA Langley Research Center/Rockwell International Corporation endeavor with the overall goal of demonstrating aeroelastic, multi-input/multi-output (MIMO), digital control of a sophisticated flexible wind tunnel model. "Aeroelastic MIMO digital control", defined here as flutter suppression (FS) and rolling maneuver load alleviation (RMLA), requires an equally sophisticated real-time, MIMO digital controller.

Approach: To obtain both computational speed and system versatility, the approach was to start with a SUN 3/160 Workstation and to add analog-to-digital (ADC) and digital-to-analog (DAC) conversion boards, a digital signal processing board and a floating point array-processing board. The resulting system was programmed in a high-level language so that code could be developed and modified easily and quickly.

Accomplishment Description: The AFW digital controller was designed, assembled, coded, validated, and tested completely in-house. Figure 45b illustrates each of these accomplishments. DESIGN shows schematically how the host CPU (Central Processing Unit), the disk and tape drives, and the added boards communicate across the VME bus. During closed-loop operation the ADC boards convert analog sensor signals to digital data; the DAC boards convert digital actuator commands to analog signals; the host CPU and the user control panel provide user interface to the signal processing board; the signal processing board ("the controller") controls the real-time processing; and the array processing board performs floating-point calculations of the FS and RMLA control laws. ASSEMBLY/CODING shows an operator sitting at the SUN 3/160 Workstation. Over 22,000 lines of code in C-language were written in the last year. VALIDATION shows a comparison of predicted and actual performance of the digital controller operating at 200 samples per second. TEST shows the AFW wind tunnel model in the NASA LaRC Transonic Dynamics Tunnel during its October/November 1989 entry. The digital controller operating a FS control law took the AFW wind tunnel model over 20% (in dynamic pressure) above its open-loop flutter boundary.

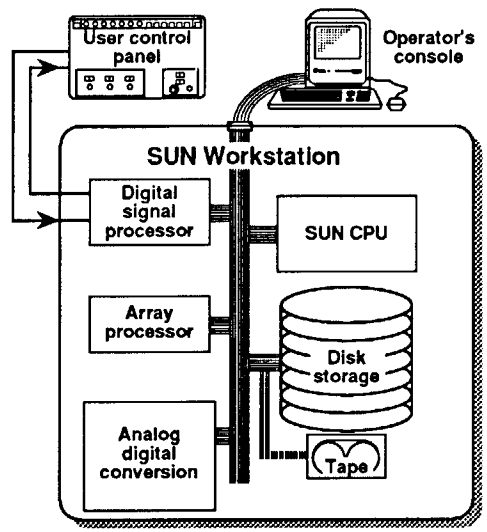
Significance: The design, assembly, coding, validation, and closed-loop testing this digital controller was performed completely in-house and represents a "first" for the Agency. The knowledge and insight gained are extremely valuable and will be applied to future projects.

Future Plans: Wind tunnel tests involving the AFW wind tunnel model and the AFW digital controller are scheduled for late summer of 1990. These tests involve additional challenges for the digital controller such as simultaneous operation of FS and RMLA and adaptive control.

Figure 45 (a).

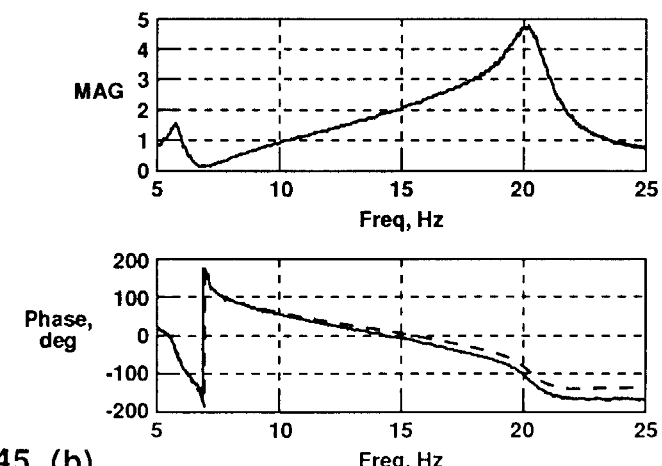
AFW DIGITAL CONTROLLER DESIGNED, ASSEMBLED, CODED, VALIDATED, AND TESTED

Designed



Assembled/Coded

Validated



Tested

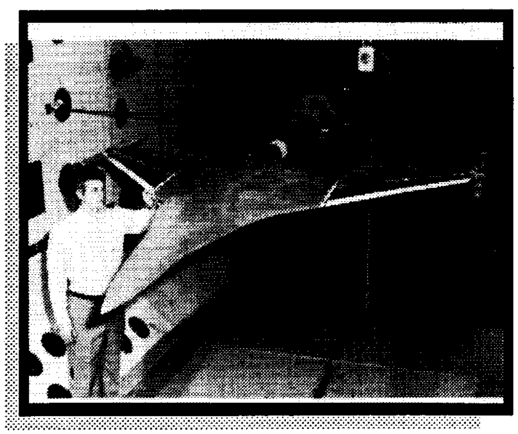


Figure 45 (b).

ACTIVE FLEXIBLE WING WIND-TUNNEL MODEL SUCCESSFULLY TESTED IN TDT

Boyd Perry III
Aeroservoelasticity Branch
and
Stanley R. Cole
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective: The Active Flexible Wing (AFW) Program is a joint NASA-LaRC / Rockwell program with the overall program goal of demonstrating multi-input/multi-output (MIMO), multifunction digital control of a sophisticated aeroelastic wind-tunnel model. The research objectives of the program are to validate the analysis, synthesis, simulation, and test methodologies necessary to perform such a demonstration.

Approach: In figure 46b, the AFW wind-tunnel model is shown mounted in the test section of the NASA-LaRC Transonic Dynamics Tunnel (TDT). The approach taken to accomplish the goal and objectives is to design, implement, and test various MIMO control concepts on the AFW wind-tunnel model. Two wind-tunnel tests are planned. During the first test, two single-function MIMO control laws are to be performed separately: flutter suppression (FS) and rolling maneuver load alleviation (RMLA). During the second test, FS and RMLA are to be combined into a multifunction control law and performed simultaneously.

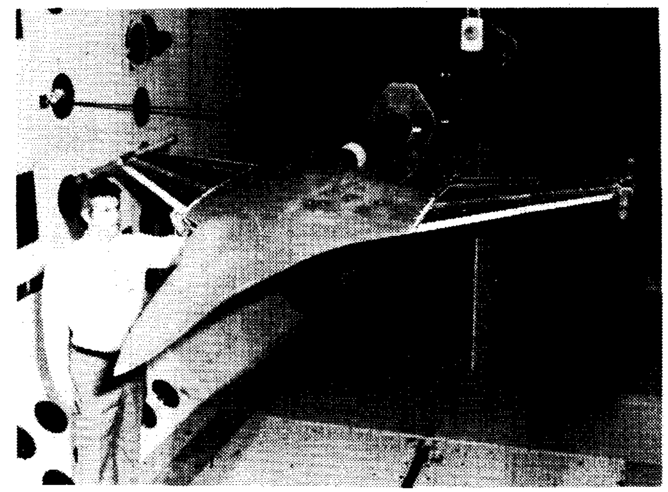
Accomplishment Description: The first of the two wind-tunnel tests was conducted in the fall of 1989 and the major accomplishments of that test are listed in the figure. Model flutter boundaries were determined; wind-tunnel model hardware, digital computer hardware and supporting software, and controller performance evaluation software worked extremely well; FS was tested both open- and closed-loop and an increase in flutter dynamic pressure of over 20% was achieved; problems encountered during the RMLA open-loop tests will be corrected during the second test.

Significance: This first wind-tunnel test affords the opportunity to validate the analysis, synthesis, simulation, and test methodologies. The knowledge and insight gained from this test, especially where experimental results and analytical predictions differ, are extremely valuable and will be applied to future projects.

Future Plans: Data reduction from the first test is currently underway, as is planning for the second test, currently scheduled for the late summer of 1990. Results from the first test will be presented at the AIAA Aerospace Engineering Conference and Exhibit to be held in Los Angeles, California in February 1990; at the 31st AIAA Structures, Structural Dynamics, and Materials Conference to be held in Long Beach, California in April 1990; and at the 17th ICAS Congress to be held in Stockholm, Sweden in September 1990.

Figure 46 (a).

ACTIVE FLEXIBLE WING WIND-TUNNEL MODEL SUCCESSFULLY TESTED IN TDT



Overall Program Goal:

- Demonstrate MIMO Multifunction Digital Control of a Sophisticated Aeroelastic Wind-Tunnel Model

Test Accomplishments:

- Subsonic Flutter Boundary and Transonic Flutter Dip Determined
- Modified Tip-Ballast Flutter-Stopper Performed Extremely Well
- Digital Controller Hardware and Software Performed Extremely Well
- Digital Rolling Maneuver Load Alleviation System (RMLA) Designed, Implemented, and Tested Open-Loop
- Three Digital Flutter Suppression Systems (FSS) Designed, Implemented, and Tested Open- and Closed-Loop
- Using FSS, 24% Increase in Flutter Dynamic Pressure Demonstrated
- On-line Controller Performance Evaluation Performed Extremely Well

Figure 46 (b).

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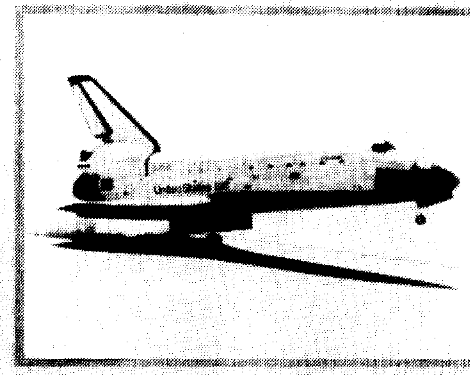
LANDING AND IMPACT DYNAMICS BRANCH



Research opportunities

- Reduce fatalities
- Improve landing gears, tires and runways
- Reduce crash loads with loading-limiting structure

Figure 47.

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LANDING AND IMPACT DYNAMICS BRANCH

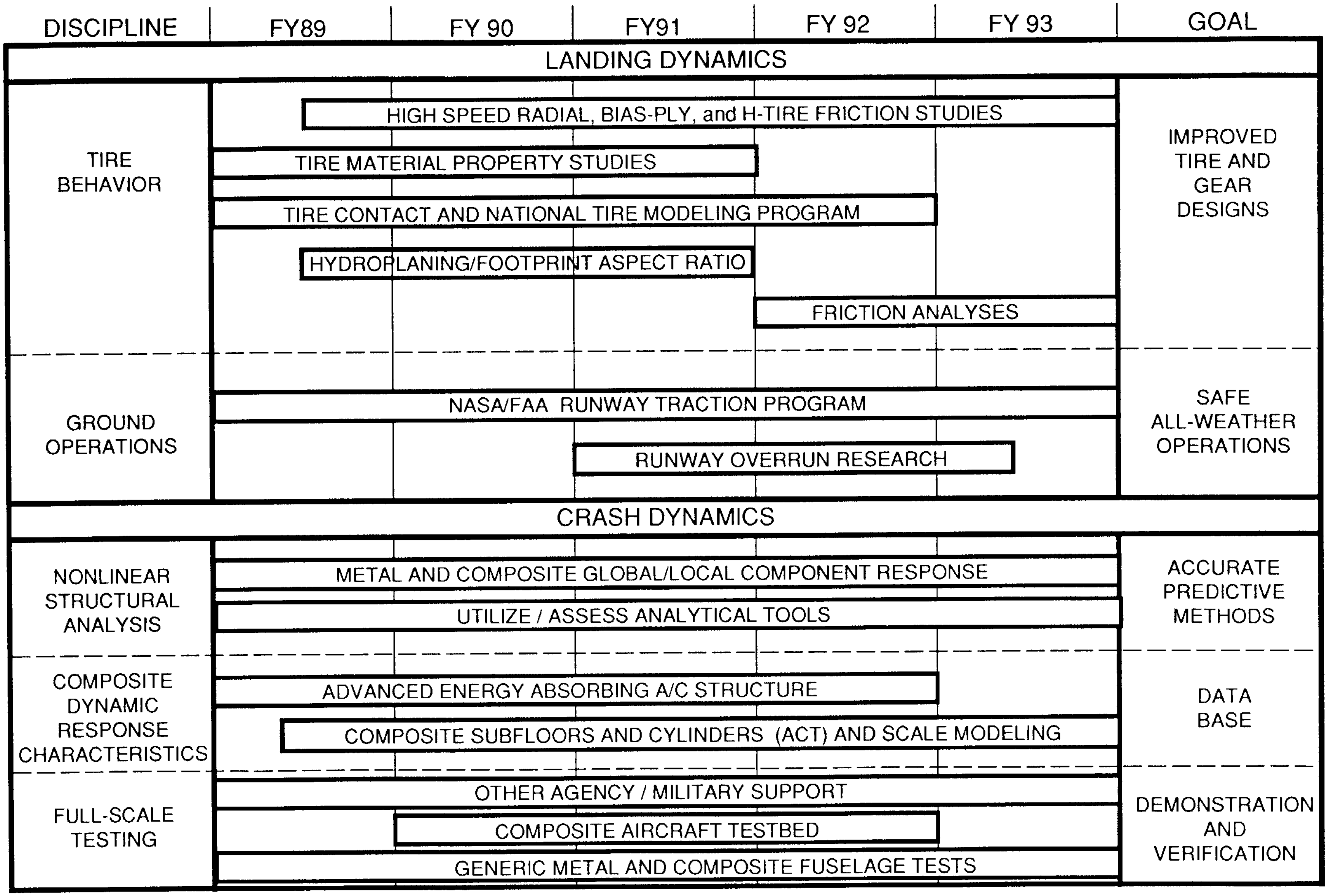

Figure 48.

WAV 12/15/1989

ROLE OF STRUCTURAL COMPONENTS EVALUATED FROM IMPACT RESPONSE OF GRAPHITE-EPOXY FLOOR SECTIONS

Richard L. Boitnott
Landing and Impact Dynamics Branch

RTOP 505-63-01

Research Objective: To understand and improve the crash behavior of graphite-epoxy aircraft structures through generic composite floor section tests.

Approach: Floor sections were fabricated by joining three graphite-epoxy semicircular Z-frames with 15 stringers which were attached to the frames with riveted aluminum shear clips. Notches were machined through the frames for intersection with the stringers. The upper floor was aluminum beams used to introduce mass loading to the specimen since interest in failure was focused on the lower graphite-epoxy structural members. The configuration was referred to as the 'skeleton floor section'. An outer skin was bonded and riveted to the frames and stringers of a second specimen to obtain a 'skinned floor section'. Both skeleton and skinned specimens were dropped at 20 fps vertical velocity onto a concrete surface to determine their dynamic response.

Accomplishment Description: As may be noted in figure 49b, the lower portion of the skeleton floor specimen lost all structural integrity during the impact. A total of 15 fractures of the frames; 5 fractures per frame, occurred at notches and most of the fractures severed the frames. On the other hand, the skinned floor specimen sustained only a few inches of permanent crushing. Although the skinned specimen had 9 fractures; 3 fractures per frame, the ends of the fractures were held in place by the intact skin and much of the original form of the floor was retained following the impact test. The importance of the presence of the skin on the floor sections are evident by comparison of magnitude of the floor decelerations. The skin stabilized the Z-frames to essentially in-plane motions thus adding greater vertical strength to the floor. The resulting floor level deceleration of 40 g's as compared to the 20 g's measured for the skeleton floor specimen is indicative of the increased strength achieved with the stabilizing of the frames. In skeleton floor, the stringers maintained frame spacing but did not prevent out-of-plane motion of the frames.

Significance: The contribution of the various parts (e.g. frames, stringers, and skin) of the floor specimens to the strength of the structure was identified and problem areas in energy absorption, structural integrity, and failure behavior have been highlighted, and insight has been gained into possible improvements which may be incorporated for better crash behavior.

Future Plans: Different cross-section frame concepts, which have a width of skin material bonded to the flange of the frame, are being tested to add to the data base and to gain additional insight into failure behavior. The tests will contribute to the goal of developing composite frame concepts which have high energy absorption and improved structural integrity under crash type loads.

Figure 49 (a).

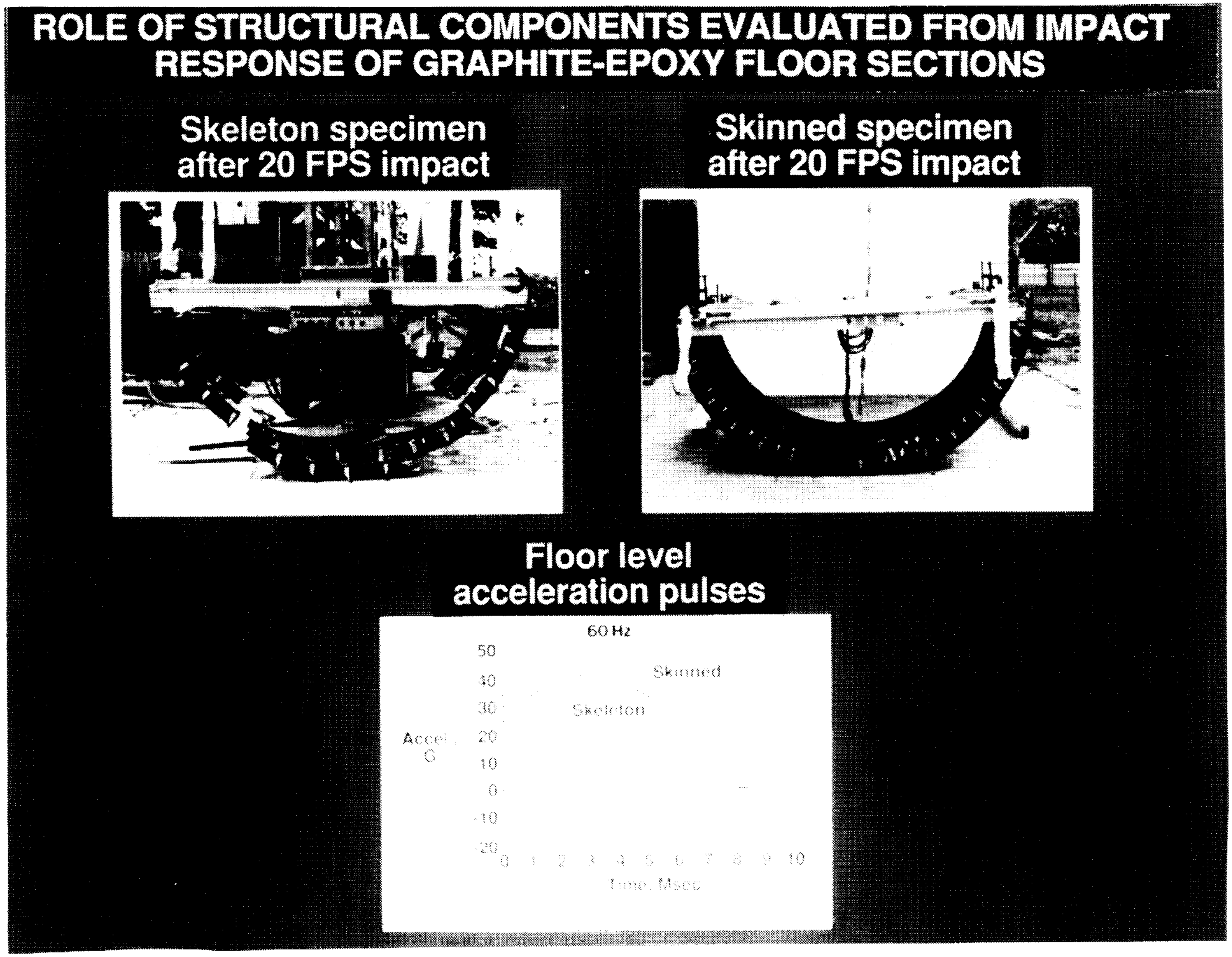

Figure 49 (b).

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IMPACT SCALING OF UNIDIRECTIONAL COMPOSITE BEAMS VERIFIED EXPERIMENTALLY

Karen E. Jackson
Landing and Impact Dynamics Branch

RTOP 505-63-01

Research Objective: An experimental program has been developed to determine the effectiveness of scale model testing for predicting the dynamic large deflection response and failure of composite beams subjected to impact. One of the main goals of the research is to characterize scaling effects in the behavior of the beams such that measurements made on subscale models will accurately predict prototype response.

Approach: Impact tests have been conducted at the Impact Dynamics Research Facility on 1/2, 2/3, 3/4, 5/6, and full scale replica model beams of AS4/3502 graphite-epoxy composite material. Beams having unidirectional, angle ply, cross ply, and quasi-isotropic lay-ups were tested under an eccentric axial compressive load to failure. The load was applied impulsively by a free falling weight using the drop tower illustrated in figure 50b. This testing configuration was chosen since it promotes large bending deformations and global beam failures which are typical of the loading environment seen by aircraft structures during a crash. Impact conditions for each test were scaled according to a scale law derived for the experiments. Test output included vertical load, end displacement of the beam, tensile and compressive surface strain at the beam midpoint, and acceleration of the drop weight. Comparisons of the test results between scale model beams and the prototype should verify the model law and indicate whether full-scale composite beam behavior can be determined through inexpensive scale model testing.

Accomplishment Description: Impact testing has been completed for all graphite-epoxy scale model beams. The figure shows a photograph of failed half and full scale unidirectional beams. Failure of the beams occurred by fiber fracture at the beam midpoint which is the location of maximum moment and by longitudinal splitting. Both the half and full scale beams exhibit this failure mode indicating that failure mechanisms are not a function of specimen size. The scaled load versus time plot shown in the figure indicates that the half scale beam accurately predicts the peak load, pulse duration, and sustained load of the full scale beam. The large scale effect in strength which was observed in previously conducted static tests is not seen in the impact response. In general, all of the scale model unidirectional beams provided an excellent comparison with the prototype when scaled by the scale law. However, comparison between the scale model and full scale load and strain time histories for the angle ply, cross ply, and quasi-isotropic laminates was inconsistent, i.e., some scale model beams showed excellent agreement while others either over or under predicted the full scale response. Thus, some laminates appear to be more suited to scale model testing than others.

Significance: It is anticipated that the results of this project will encourage the use of scale model tests in studies of the crashworthiness of transport aircraft and helicopters, particularly at the subcomponent level.

Future Plans: A dynamic beam analysis will be developed to model the composite beam response and a stress analysis will be performed to investigate beam failures. Application of the scaling techniques presented in this research project may be explored for more complicated aircraft structures such as curved frames. Further research is needed to fully understand the limitations of scale model testing of composite structures.

Figure 50 (a).

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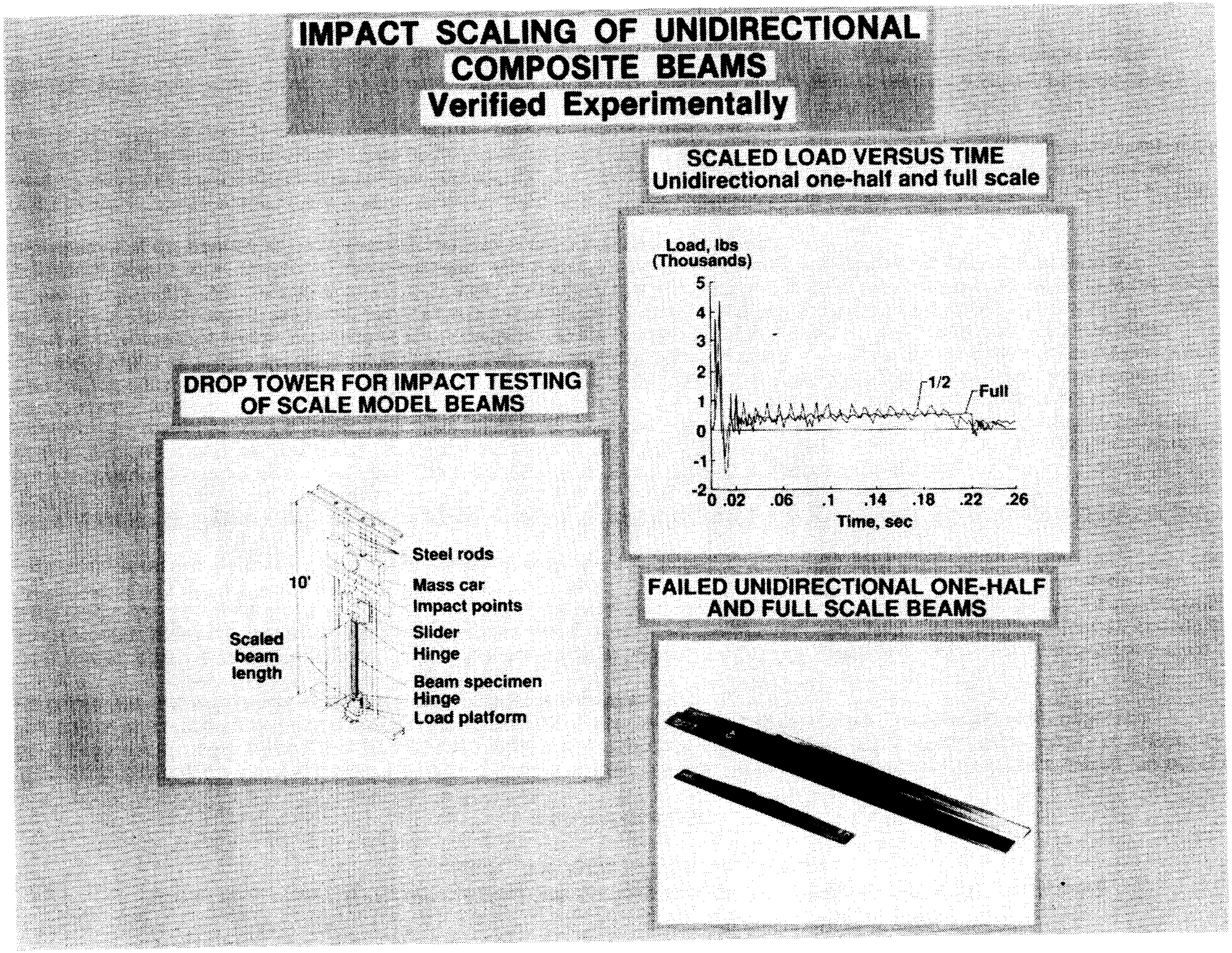


Figure 50 (b).

EXPERIMENTAL DETERMINATION OF COMPOSITE BEAM BENDING STIFFNESS CORRECTS OVERPREDICTION BY LAMINATION THEORY IN LARGE DEFLECTION BEAM ANALYSIS

Karen E. Jackson, U. S. Army Aerostructures Directorate, and Edwin L. Fasanella (Lockheed)
Landing and Impact Dynamics Branch

RTOP 505-63-01

Research Objective: The goal of the research is to demonstrate the effectiveness of scale model testing in studying the energy absorption and dynamic response capabilities of aircraft subcomponents subjected to impact or "crash" type loads. To accomplish this objective, an experimental and analytical program was developed to investigate scaling effects in the large deflection response and failure behavior of graphite-epoxy composite beams.

Approach: Static tests were conducted on several scale model beams of AS4/3502 graphite-epoxy composite material having unidirectional, angle ply, cross ply, and quasi-isotropic laminate stacking sequences. The beams were tested under an eccentric axial load to failure, as shown in figure 51b. This testing configuration was chosen since it promotes large bending deformations and global failure of the beam away from the hinged end support. Two methods of analysis were performed to compare with experimental results & to aid in fundamental understanding of the problem.

Accomplishment Description: A one dimensional large rotation "elastica" type beam solution was developed to predict the load versus end displacement response of the composite beam-columns under eccentric axial load. This analysis incorporated the exact expression relating moment and curvature thus allowing the solution to predict large rotation response. In addition, a nonlinear finite element structural analysis code was used to model the composite beam-column. The normalized load versus end displacement response for the 1/6 and full scale quasi-isotropic beams is shown in the figure with the predictions from the large rotation beam analysis and finite element model. The Euler load for the beams was determined using an equivalent beam bending stiffness derived from classical lamination theory using properties of the AS4/3502 system determined from material characterization tests. As shown in the figure, the analysis substantially overpredicts the experimental beam response and a scale effect is evident in the elastic response of the beam which is not predicted by the analyses. The normalized analytical beam response is independent of the beam stiffness and a scale effect in elastic response is unlikely which suggests that the bending stiffness calculated from lamination theory is in error. Experiments were performed to determine the bending stiffness of the scaled composite beams when loaded as beam-columns. Agreement between experiment and analysis is excellent when the experimentally determined value of bending stiffness is used to normalize the data.

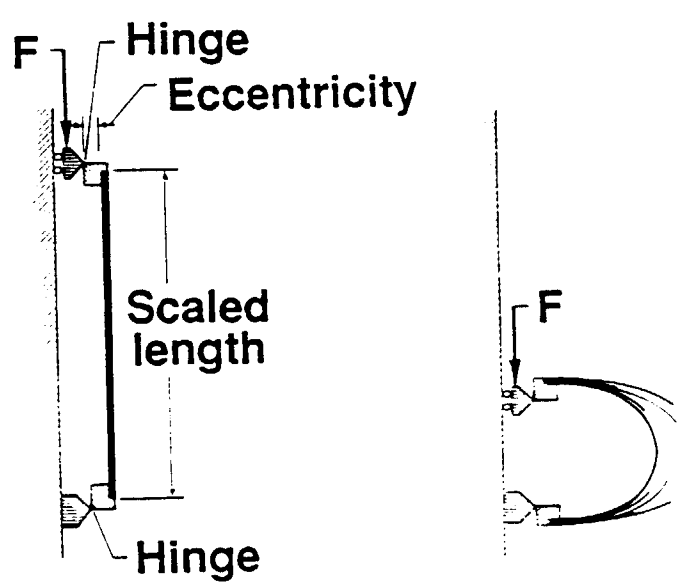
Significance: It was found that lamination theory overpredicts the bending stiffness of composite beams by as much as 20 per cent. It was necessary to determine the bending stiffness experimentally to obtain good agreement with a large rotation beam analysis and a finite element model. With an accurate determination of bending stiffness, the analytical techniques can be used to predict beam stress and strain response for failure analysis.

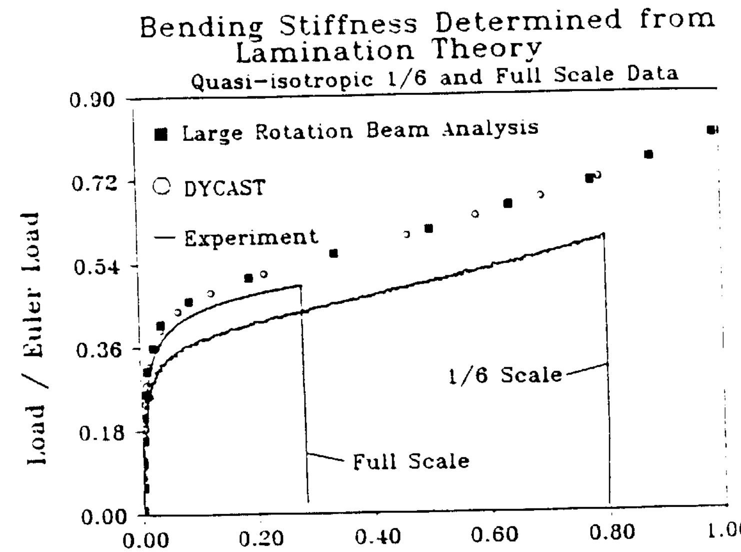
Future Plans: The large rotation beam solution and the finite element model will be used to perform a stress analysis on the eccentrically loaded beam-column to investigate the significant scale effect observed in strength of the scale model beams.

Figure 51 (a).

EXPERIMENTAL DETERMINATION OF COMPOSITE BEAM BENDING STIFFNESS CORRECTS OVERPREDICTION BY LAMINATION THEORY IN LARGE DEFLECTION BEAM ANALYSIS

SCHEMATIC DRAWING OF TEST CONFIGURATION





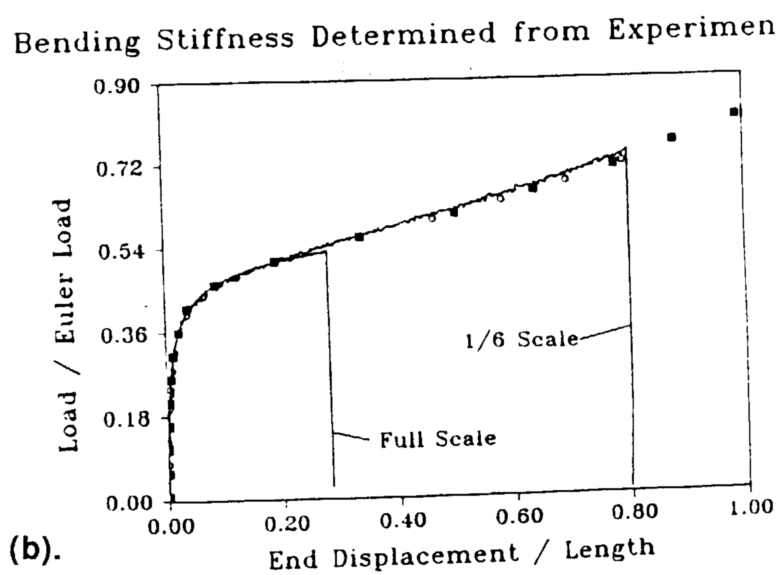


Figure 51 (b).

SUBSTANTIAL REDUCTIONS IN TENSILE STRENGTH OF FIBER COMPOSITES OBSERVED BETWEEN SCALED MODELS AND FULL SCALE STRUCTURES

Karen E. Jackson, U.S. Army Aerostructures Directorate
Landing and Impact Dynamics Branch
and
John Morton and Sotiris Kellas (Virginia Polytechnic Institute and State Univ.)

RTOP 505-63-01

Research Objective: Study & isolate factors causing scale effects on the tensile strength of graphite reinforced epoxy laminates.

Approach: Previous research has shown that the strength and/or stiffness of fiber reinforced composite materials depends upon the size of the composite laminates. This problem appears to be enhanced with increasing laminate and/or ply thickness. Research to date have failed to establish the exact causes of such behavior. The current study presents preliminary experimental data which isolates factors responsible for scale effects in strength and will be used in optimizing subsequent tests. Four laminate types, (0/90), (±45), (±30/90) and (0/±45/90), were chosen with appropriate stacking sequences to highlight individual and interacting failure modes. Furthermore, four scaled sizes were selected for investigation such that at least one of the four specimens complies with ASTM D 3039 standard. All specimens were tested at a constant rate of displacement (0.2"/min) on a 120,000 lbf capacity Tinius Olsen test machine, equipped with wedge type grips. Data in the form of load versus time were acquired on an IBM PC.

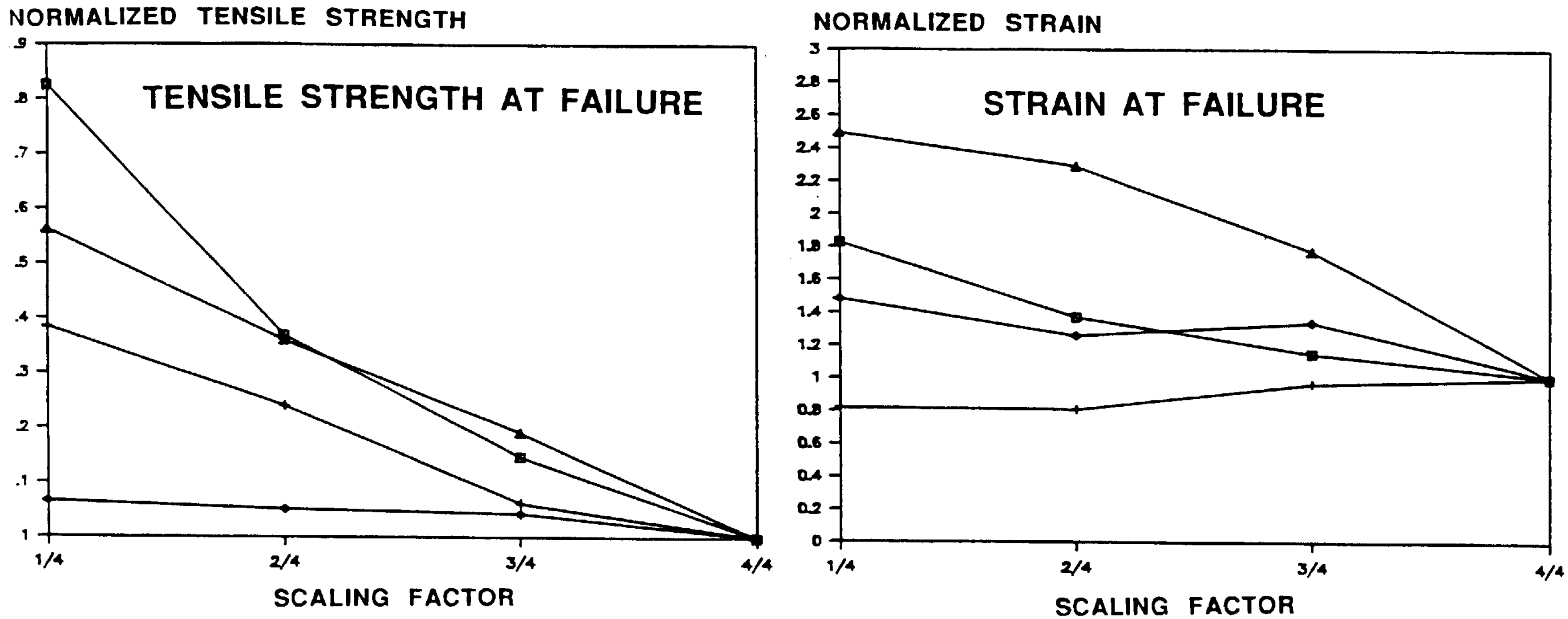
Accomplishment Description: Preliminary results are shown in figure 52b. The effect of scaling upon the tensile strength is demonstrated in the left figure where normalized strength at time of failure is plotted versus model size. There are at least two important points which should be noted. (1) The tensile strength depends upon the specimen size; the greater the size the smaller the strength. (2) The scaling effect appears to be smoothed out with increasing specimen size. The effect of scaling with stacking sequence is demonstrated in the right figure, where normalized strain at failure is plotted against the scaling factor for each stacking sequence. The most important result is the fact that the effect of scaling on the strain depends upon both the model size and the stacking sequence. Moreover, the effect appears to be dependent upon the number of 0 degree plies in each stacking sequence. The more 0 degree plies in a given lay-up the smaller the scaling effect upon the strain at time of failure. One explanation involved in the observed trend is the volume of the material in the larger scale specimens. The probability of inclusion of critical flaws or imperfections increases with volume thus leading to lower failure strains.

Significance: The scaling effects noted in the composite material specimens presents a serious problem when laboratory data have to be used in the design of full scale structures. The understanding of the effects under study should eventually permit numerical models to be developed which can predict the tensile strength in any size laminate.

Future Plans: Design suitable extensometers to measure direct strain and examine damage development with cameras/X-ray, and/or deply techniques for damage evaluation. Apply moire interferometry to examine free edge stress development and devise 3-D finite element models for stress analysis.

Figure 52 (a).

SUBSTANTIAL REDUCTIONS IN TENSILE STRENGTH OF FIBER COMPOSITES
OBSERVED BETWEEN SCALED MODELS AND FULL SCALE STRUCTURES



Stacking Sequence Legend

□	$(\pm 30_n / 90_{2n})_s$	◊	$(\pm 90_n / 0_n / 90_n / 0_n)$
+	$(\pm 45_n / 0_n / 90_n)_s$	Δ	$(\pm 45_n / \pm 45_n)_s$

Figure 52 (b).

DYCAST ANALYSIS OF POSSIBLE NTF ACCIDENT SCENARIO MATCHES POST-MISHAP OBSERVATIONS

Edwin L. Fasanella (Lockheed)
Martha P. Robinson
Landing and Impact Dynamics Branch

RTOP 505-56-01

Research Objective: After the National Transonic Facility (NTF) mishap, researchers were called upon to assist in analyzing possible accident scenarios. The nonlinear dynamic finite element code DYnamic Crash Analysis of STructures (DYCAST) was used to model one possible impact scenario of one of two clamps that spun off the NTF drive shaft at approximately 1624 in/sec and impacted bulkhead fairing plates attached to the 15 foot-diameter ring of the upstream nacelle bulkhead. Post mishap, two adjacent plates were missing in one area and three adjacent plates in another.

Approach: A DYCAST finite element model representing three of the 16 curved 35.75 x 8.125 x 3/8 inch thick 5083-O aluminum bulkhead fairing plates and two 6 x 3 3/8 x 3/8 inch thick splice plates that connect adjacent fairing plates was constructed. Six beam elements per plate representing bolts which fasten each fairing plate to the 15 foot-diameter bulkhead ring, and four beam elements per plate representing splice plate bolts were used. A total of 340 triangular shell elements for fairing and splice plates and 24 contact elements representing the contact surface of the bulkhead ring were also used in the model. A 2.7 pound clamp was assumed to impact the cantilevered edge of the second fairing plate about halfway between the second and third bolts that fasten the plate to the bulkhead ring. From the rotational speed of the fan drive shaft, the impact velocity was assumed to be 1624 in/s at 17.3 degrees to the normal to the plate (see figure 53b). This location corresponded to an imprint of an impact noted on a fairing plate following the accident.

Accomplishment Description: Fifteen milliseconds in real time required about 12 hours run time on a MicroVax II. As shown in the attached figure, the fairing plate bolt to the left of the impact failed first at 0.4 milliseconds and by 3 milliseconds all six bolts that hold the second plate to the bulkhead ring and the two bolts in the splice plate between plates 1 and 2 had failed. The second plate lifted and began to pull the third plate up also. By 15 milliseconds all the bolts in the third plate had failed. This scenario matched the post-mishap observations where two adjacent plates were missing in one area.

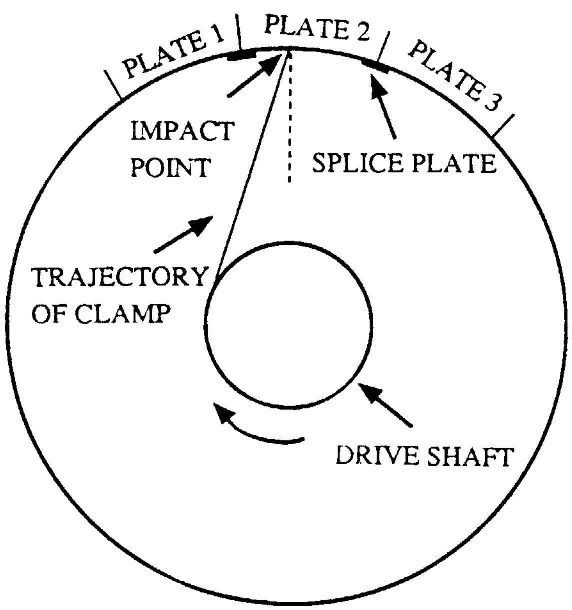
Significance: The 2.7 pound projectile was half of the smaller of the two clamps. Since only one half of the smaller clamp was able to theoretically dislodge two fairing plates, it is evident that the complete clamp assembly (9.5 lbs) and/or the parts of the longer clamp assembly (27 lbs) would have had sufficient energy to displace the other three adjacent plates observed missing post mishap.

Future Plans: Document this analysis in a quick release Technical Memorandum.

Figure 53 (a).

DYCAST ANALYSIS OF POSSIBLE NTF ACCIDENT SCENARIO MATCHES POST-MISHAP OBSERVATIONS

NTF BULKHEAD RING SCHEMATIC



DYCAST THREE PLATE MODEL

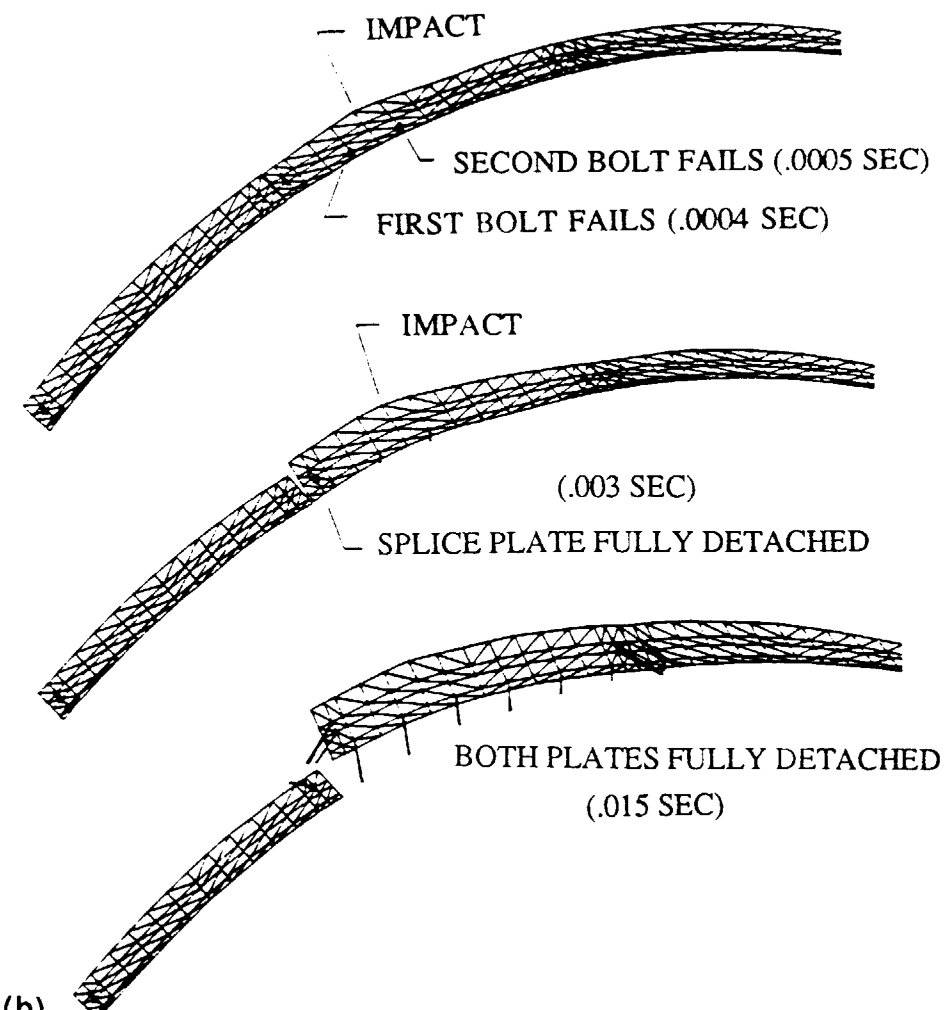


Figure 53 (b).

NEW ANALYTICAL SENSITIVITY DERIVATIVES PROVIDE CAPABILITY TO EVALUATE LAMINATION AND MATERIAL EFFECTS ON THE VIBRATIONAL RESPONSE OF COMPOSITE FUSELAGE FRAMES

Ahmed K. Noor, Jeanne M. Peters
George Washington University

Huey D. Carden
Landing and Impact Dynamics Branch

RTOP 505-63-01

Research Objective: To study the effects of variations in lamination and material parameters of thin-walled composite frames on their vibrational characteristics.

Approach: The flanges and webs of the composite frames were modeled by using highly accurate two-dimensional shell (and plate) finite elements. Analytic derivatives of the vibrational response with respect to lamination and material parameters of each of the flanges and web are generated. These derivatives show the sensitivity of the response to variations in the design parameters (e.g. fiber orientation angles, elastic moduli, etc.).

Accomplishment Description: As shown in the attached figures 54b and 54c, semicircular thin-walled composite frames with I and J sections having quasi-isotropic layups have been studied. In-plane and out-of-plane energies ratioed to total energy allows characterization of the complex vibration modes as in- or out-of-plane modes. Additionally, the vibrational response of the frames was found to be more sensitive to the longitudinal elastic modulus E_1 than to all the other material parameters. The sensitivity derivatives of all the vibration frequencies with respect to E_1 were found to be nearly equal. Also, the response was considerably more sensitive to variations in the +45, -45 degree fiber angles than in the 0, 90 degree angles.

Significance: The sensitivity studies, in addition to identifying the important lamination and material parameters, suggest the feasibility of replacing the quasi-isotropic composite by an equivalent isotropic material in the one-dimensional thin-walled beam analysis.

Future Plans: The sensitivity derivatives will be used in conjunction with experimental data and model adjustment techniques for validating the numerical model, and carrying out dynamic response calculations.

Figure 54 (a).

EXPERIMENTAL SETUP FOR VIBRATIONS OF COMPOSITE FRAMES

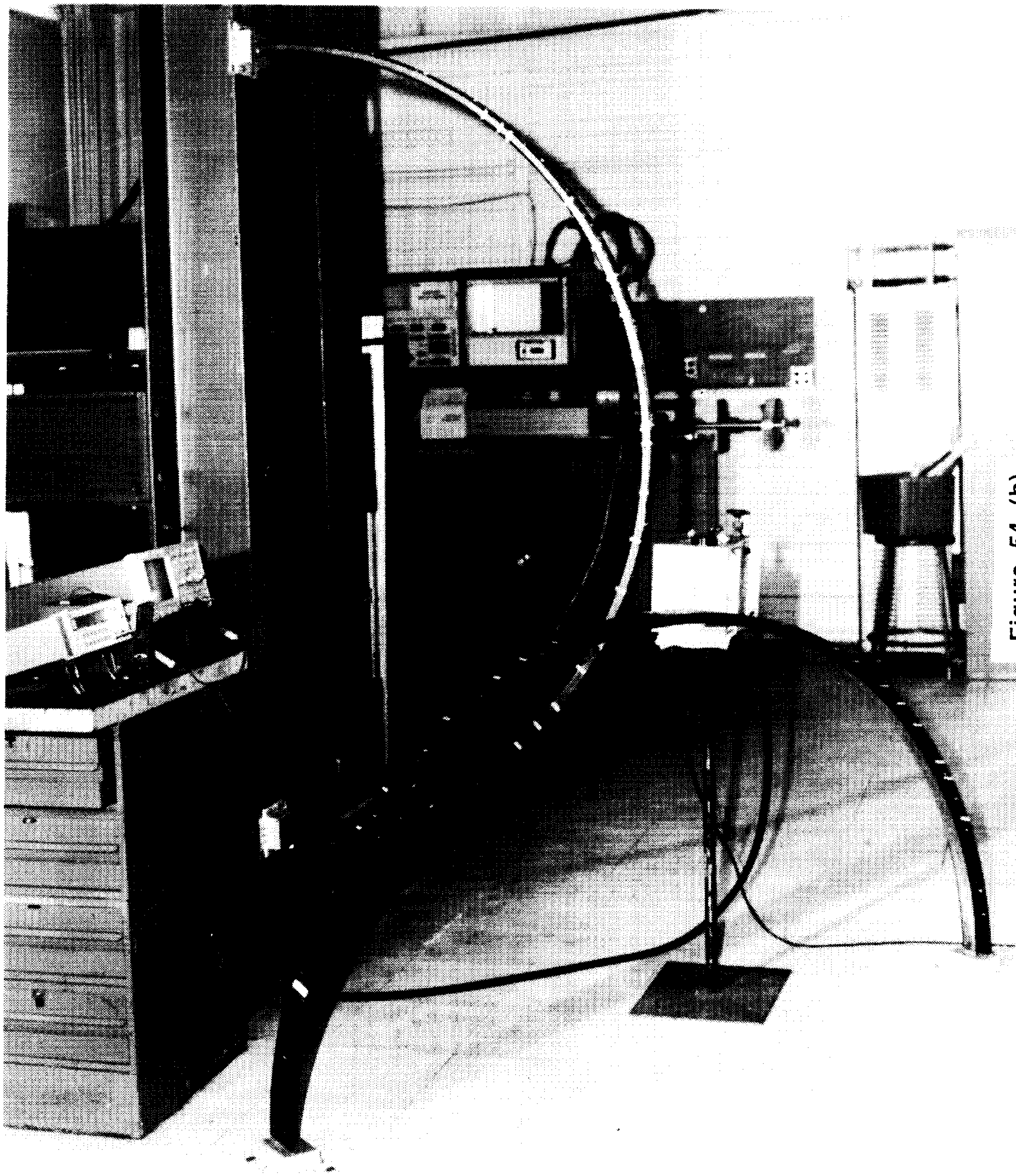
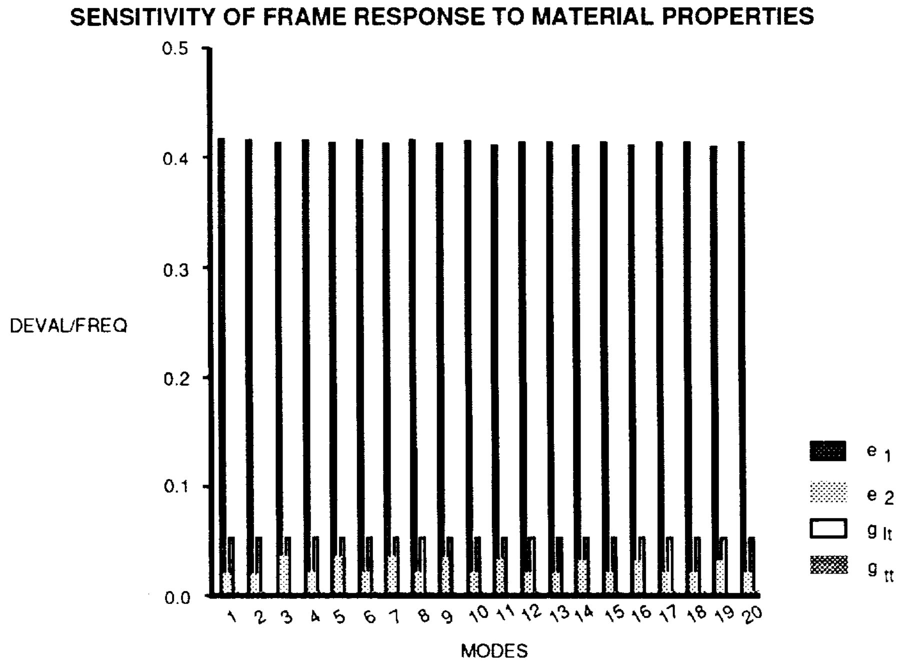


Figure 54 (b).


SENSITIVITY OF FRAME RESPONSE TO MATERIAL PROPERTIES
DEVAL/FREQ
0.0
0.1
0.2
0.3
0.4
0.5
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20
MODES
e_1
e_2
g_{lt}
g_{tt}

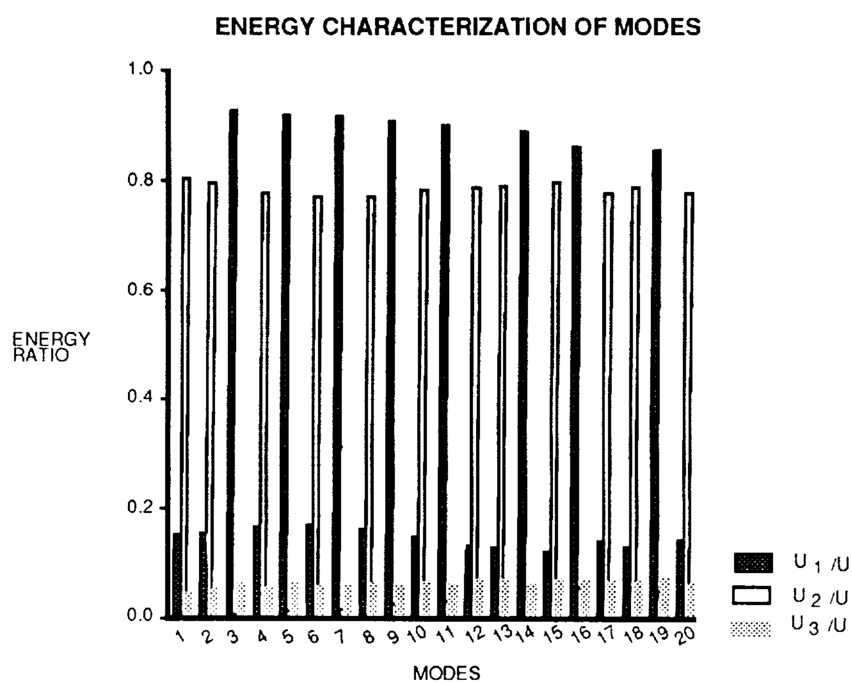

ENERGY CHARACTERIZATION OF MODES
ENERGY RATIO
0.0
0.2
0.4
0.6
0.8
1.0
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20
MODES
U_1/U
U_2/U
U_3/U

Figure 54 (c).

TIRE ANALYSIS CODES VALIDATED WITH EXPERIMENTAL MEASUREMENTS OF SPACE SHUTTLE NOSE GEAR TIRE

Kyun O. Kim and Ahmed K. Noor
George Washington University

John A. Tanner
Landing and Impact Dynamics Branch

RTOP 505-63-41

Research Objective: To develop a family of analysis tools, which have been validated experimentally, that will streamline the aircraft tire design process.

Approach: Numerical modeling of the response characteristics of aircraft tires remains one of the most challenging applications of computational mechanics. The tire is a composite structure composed of rubber, textile, and steel constituents which exhibits anisotropic and nonhomogeneous material properties. Furthermore, all the forces developed by the tire associated with takeoff and landing operations are developed through the tire-pavement interface. These facts emphasize the need to develop modeling strategies and analysis methods which are both powerful and efficient. Two codes that have been developed for tire analyses include a semianalytic model which employs finite elements in the tire cross section that are expanded in a Fourier series around the tire circumference and a two-dimensional shell finite element model that includes a frictionless contact algorithm.
The analytical results from these two models are compared with experimental measurements of the Space Shuttle nose gear tire.

Accomplishment Description: The carcass of the Space Shuttle nose gear tire was treated as an anisotropic material and the properties of the different layers were obtained using the mechanics of materials approach. Because of symmetry only half the tire cross section was modeled and as shown in figure 55b, it was divided into seven segments. Each segment contains a different number of layers, different material properties corresponding to different textile cord content in the carcass, and different fiber orientation. Spline interpolation was used to smooth the measured data and to obtain the geometric and material characteristics of the finite element models. The semianalytic finite element was used to model the Space Shuttle nose gear tire subjected to an internal pressure loading of 300 psi and the analytical results are compared with experimental measurements along a meridional line in the figure. This loading condition is axisymmetric. The figure illustrates the change in tire cross section geometry for this pressure loading condition and demonstrates excellent agree between the model and the experiment. The two-dimensional shell finite element with a frictionless contact algorithm was used to model the load-deflection characteristics of the inflated tire vertically loaded against a rigid plate. The figure clearly illustrates the ability of the finite element model to accurately predict the nonlinear response of the Space Shuttle nose gear tire to this loading condition.

Significance: The excellent agreement between the finite element model predictions and the experimental measurements validates these two tire analysis codes which form the core for the family of analysis tools under development.

Future Plans: The family of analysis tools will continue to grow with the development of contact algorithms with friction included. These analysis tools will eventually be able to handle such complicated conditions as rolling contact with friction, tire dynamics, thermo-mechanical loading conditions.

Figure 55 (a).

TIRE ANALYSIS CODES VALIDATED WITH EXPERIMENTAL MEASUREMENTS OF SPACE SHUTTLE NOSE GEAR TIRE

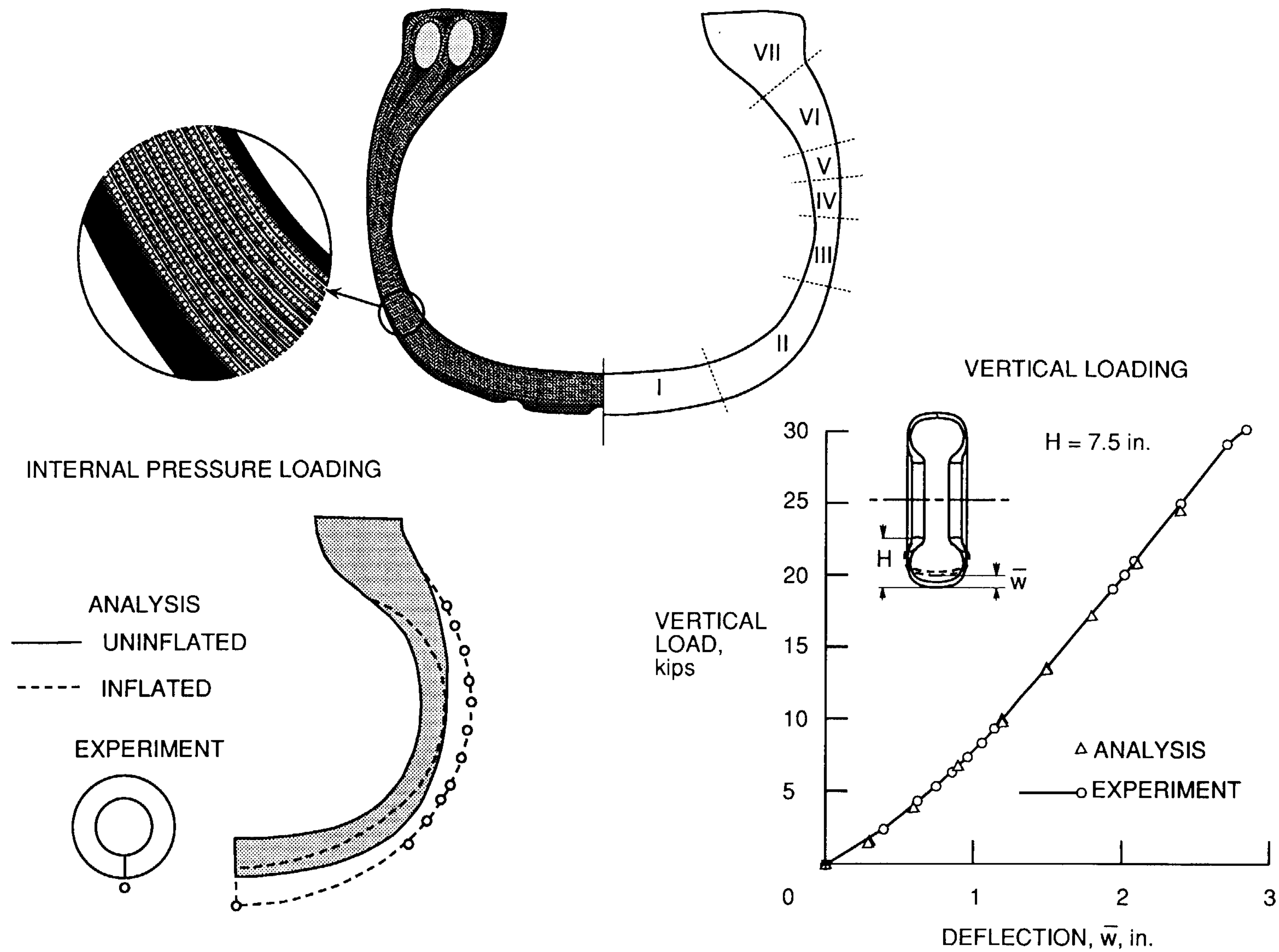


Figure 55 (b).

AIRCRAFT LANDING DYNAMICS FACILITY CAPABILITIES ENHANCED

Robert H. Daugherty and Sandy M. Stubbs
Landing and Impact Dynamics Branch

RTOP 505-63-41

Research Objective: To increase the productivity of the Aircraft Landing Dynamics Facility (ALDF) so that landing gear and tire research programs can be completed in a more timely manner.

Approach: With the implementation of the joint NASA/FAA Radial Tire Program on ALDF, facility operation time is in great demand. The test schedule for ALDF is completely filled over the next five years and there is a considerable backlog of research programs for the facility. In an effort to streamline the operation of the facility, several improvements are being incorporated into ALDF. One improvement was the installation of a test runway wetting system using 45 sprinkler heads to wet the entire 1800-foot length of the runway. As shown in figure 56b, a second improvement which will be completed by the spring of 1990, is the installation of a variable yaw control system to the test carriage that will allow the test tire to be steered through a range of yaw angles over the course of a single test run. The current system only permits testing at constant yaw angles which are preset prior to each test run on ALDF.

Accomplishment Description: The newly installed runway wetting system eliminates the necessity of using three individuals for runway wetting purposes when the test matrix calls for a wet runway. The new wetting system maintains a more consistent wetness condition than the old manual system and completely eliminates test delays associated the effort to maintain a wet runway test condition on a hot summer day. This same wetting system can be used in combination with a system of rubber dams to maintain a flooded runway test condition. The new variable yaw control system for the test carriage will allow the entire range of yaw angles necessary to define the cornering characteristics of an aircraft tire to be covered in a single test run. The current system requires as many as seven test runs to define the tire cornering behavior for each tire vertical load and speed combination. The variable yaw control system also enables dynamic steering testing to define the shimmy characteristics of aircraft tires.

Significance: These two modifications to ALDF will greatly enhance the productivity of this unique national facility and accelerate the development of the data base on radial-belted aircraft tires.

Future Plans: These modifications will be used to streamline the tire testing procedures on ALDF and step up the pace of radial-belted tire testing.

Figure 56 (a).

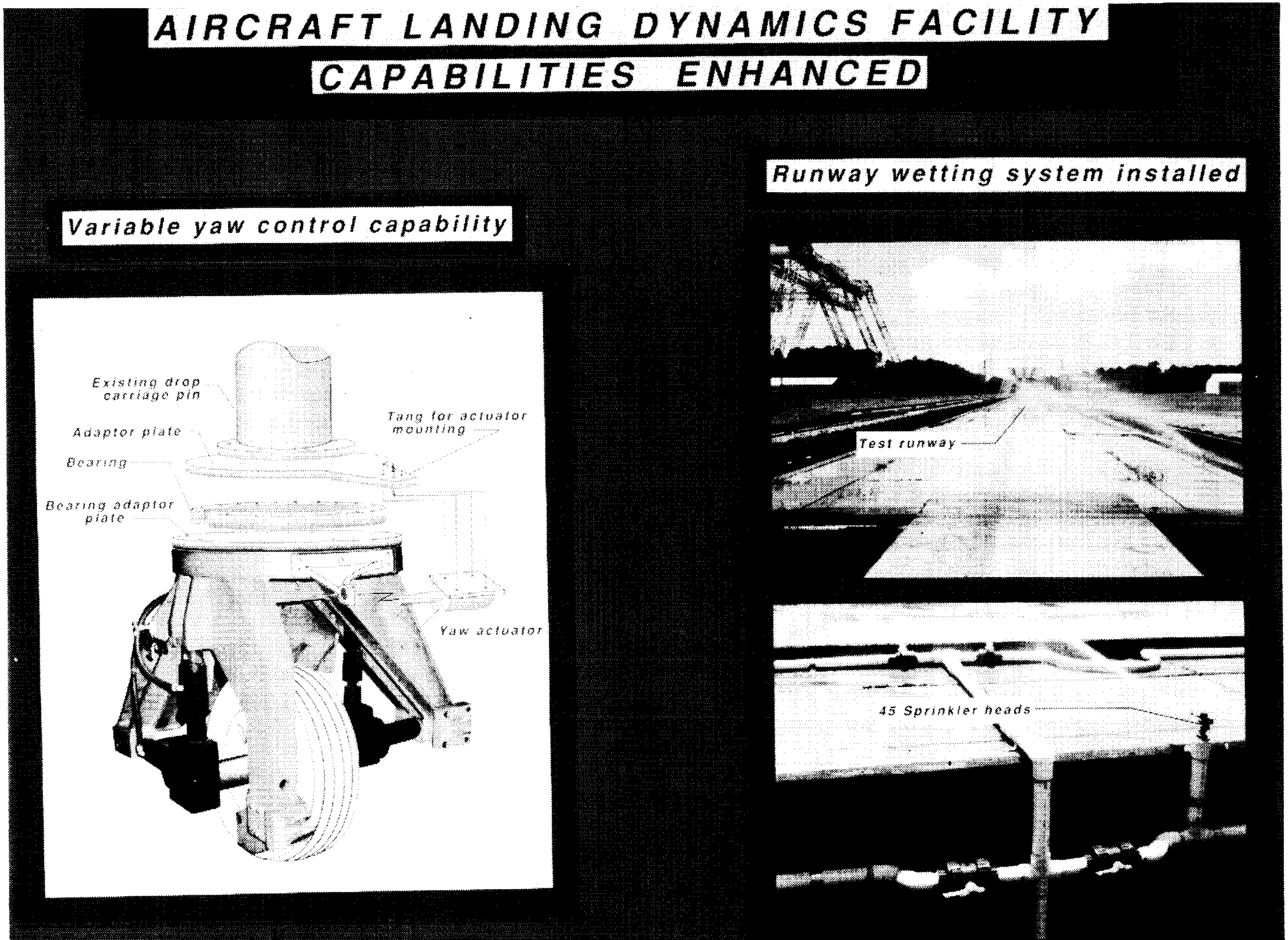


Figure 56 (b).

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PERFORMANCE OF F-106 ACTIVELY-CONTROLLED NOSE GEAR IMPROVED

William E. Howell, John R. McGehee, William A. Vogler (Lockheed), and Robert H. Daugherty
Landing and Impact Dynamics Branch

RTOP 505-63-41

Research Objective: To reduce landing and rollout loads transmitted by landing gear to aircraft structures through the use of active control of strut hydraulics.

Approach: In preparation for flight-testing on the Langley Research Center F-106 aircraft, both the nose and main gear have been modified to allow active control of hydraulic flow in the strut lower chamber. Vertical drop tests using the nose gear are being conducted to verify control system operation. Both passive and active tests are compared. Later, main gear vertical drop tests will be conducted and associated controller modifications will be evaluated.

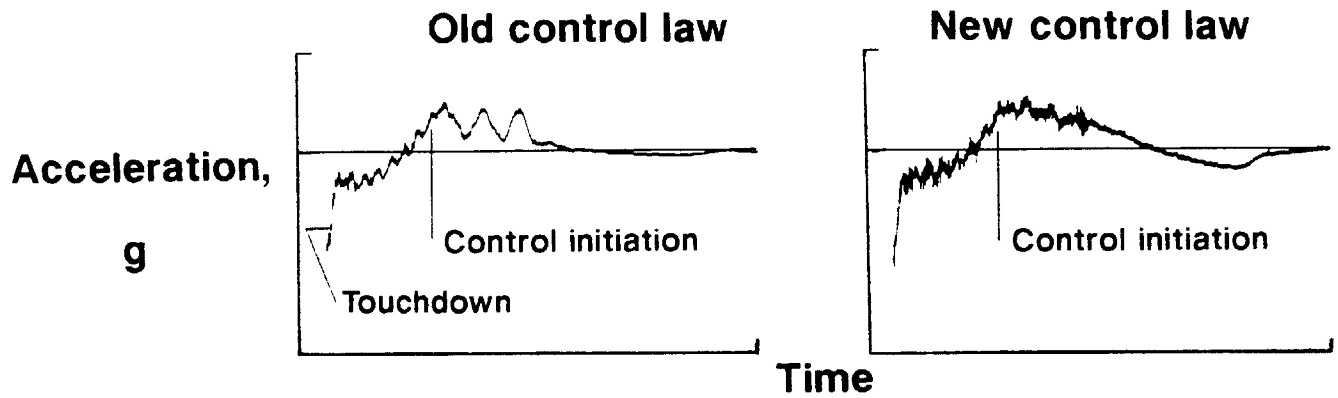
Accomplishment Description: Based upon the computed impact energy and remaining strut stroke after touchdown, the controller computes a minimum control force (acceleration) level which it then attempts to maintain by adding or removing hydraulic fluid to dissipate the energy efficiently at the control force level using the remaining available stroke. On initial checkout of the controller during active nose gear drop tests, several problems were encountered. Differential hydraulic gain (rate of fluid into strut versus rate of fluid out) was identified as contributing to less-than-ideal performance. A more basic problem was the control law itself. Initially, the law simply provided an acceleration feedback term to the controller. This kind of feedback would not allow for any "anticipation" of further events, and, coupled with hardware in the system which caused an approximate 78 degree phase-shift between computed force errors and actual servo valve commands, resulted in load oscillations as shown in the top left of figure 57b. While load reduction did take place, the oscillatory nature of the system was undesirable. Subsequently, the system hydraulic gain was corrected, and the control law was modified to incorporate an acceleration rate feedback term. This term allows for larger servo valve commands when acceleration is sloping away from the desired control level and lowers servo valve commands when acceleration slopes towards the desired control level. This kind of control gives the appearance of being able to "anticipate" acceleration peaks, and after a lead/lag network was incorporated into the servo command hardware to reduce phase shift, very desirable behavior resulted as shown in the top right of the figure.

Significance: With the newer control law and hardware modifications, the performance of the active control system shows a 47 percent decrease in touchdown loads compared to a passive gear as shown in the bottom of the figure. Rebound is also shown to decrease.

Future Plans: Vertical drop tests are scheduled to be completed by September, and flight tests are scheduled to commence after the F-106 vortex flap program is completed.

Figure 57 (a).

PERFORMANCE OF F106 ACTIVELY-CONTROLLED NOSE GEAR IMPROVED



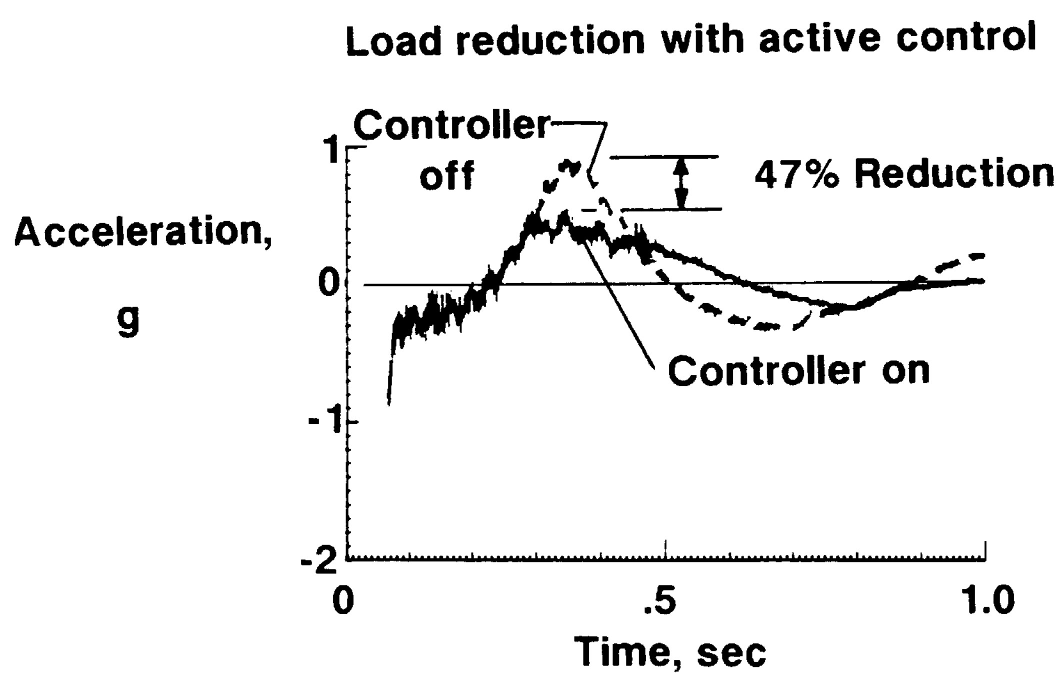


Figure 57 (b).

PRELIMINARY ALDF TEST RESULTS DEFINE MECHANICAL AND FRICTION CHARACTERISTICS OF BIAS-PLY, RADIAL-BELTED, AND H-TYPE TIRES

Pamela A. Davis, Sandy M. Stubbs, and Thomas J. Yager
Landing and Impact Dynamics Branch

RTOP 505-63-41

Research Objective: The objective of the joint FAA/NASA Radial Tire Program is to develop a database of tire mechanical properties and friction characteristics for radial-belted and H-type aircraft tires, and to compare these characteristics with those of comparable bias-ply aircraft tires.

Approach: Static and dynamic testing of several bias-ply, radial-belted, and H-type aircraft tires representing the full range of tire sizes and strength capabilities are being carried out on a variety of laboratory apparatus and the Aircraft Landing Dynamics Facility (ALDF). These tests are aimed at defining the stiffness damping and friction properties of these tires over a range of operational loads and runway wetness conditions. The first phase of this research program involves testing on a smooth, ungrooved concrete runway at the ALDF.

Accomplishment Description: The first three tires to be tested in the Radial Tire Program are shown in figure 58b. These 40 x 14 aircraft tires are the size used on the many versions of the Boeing B-737 and McDonnell Douglas DC-9 transport aircraft. ALDF testing of the bias-ply and radial-belted tires has been completed and testing of the H-type tire is underway. Preliminary results of testing to date is shown in figure 58c. The vertical load-deflection characteristics of the three tires are shown on the upper left of figure 58c. Both the bias-ply and H-type tires exhibit higher stiffness values than the radial-belted tire when all three tires are inflated to 170 psi. The slow yaw roll properties of the bias-ply and radial-belted tires are shown in the upper right in figure 58c. For these tests the tire is loaded on the runway at a preset yaw or steering angle and then slowly rolled forward to define the build up of side force with roll distance. This information defines the tire response to transient steering conditions and is critical for tire shimmy analyses. The cornering friction performance of the bias-ply and radial-belted tires on a wet runway is shown on the bottom of figure 58c. Here the radial-belted tire is shown to produce larger cornering forces than the bias-ply tire over the entire range of yaw angles tested.

Significance: The information contained in the figures is necessary to establish the compatibility of the radial-belted aircraft tires with such landing gear components as the antiskid braking system, the nose-gear steering system, and the shimmy damping system. These data will be used to insure the ground handling safety of future aircraft equipped with radial-belted tires.

Future Plans: This information will eventually be incorporated into a reference publication to be used as a landing gear design guide. Future tests will be conducted of various grooved runway surfaces to establish the effect of pavement texture variations on tire performance.

Figure 58 (a).

40 X 14 TEST TIRES

RADIAL PLY **BIAS PLY** **BIAS PLY, H-TYPE**

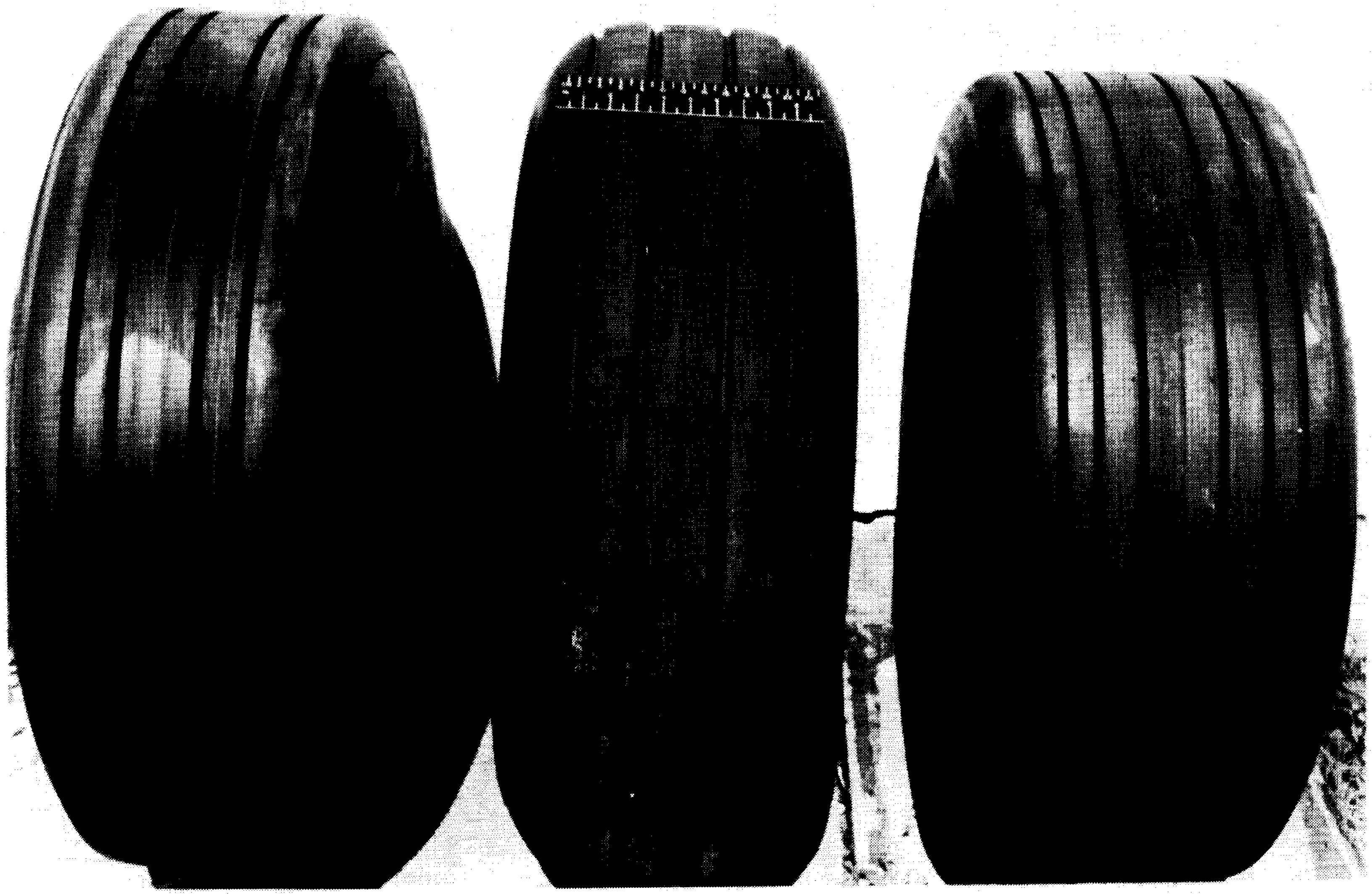


Figure 58 (b).

PRELIMINARY ALDF TEST RESULTS DEFINE MECHANICAL AND FRICTION CHARACTERISTICS OF BIAS-PLY, RADIAL-BELTED, AND H-TYPE TIRES

INFLATION PRESSURE, 170 psi

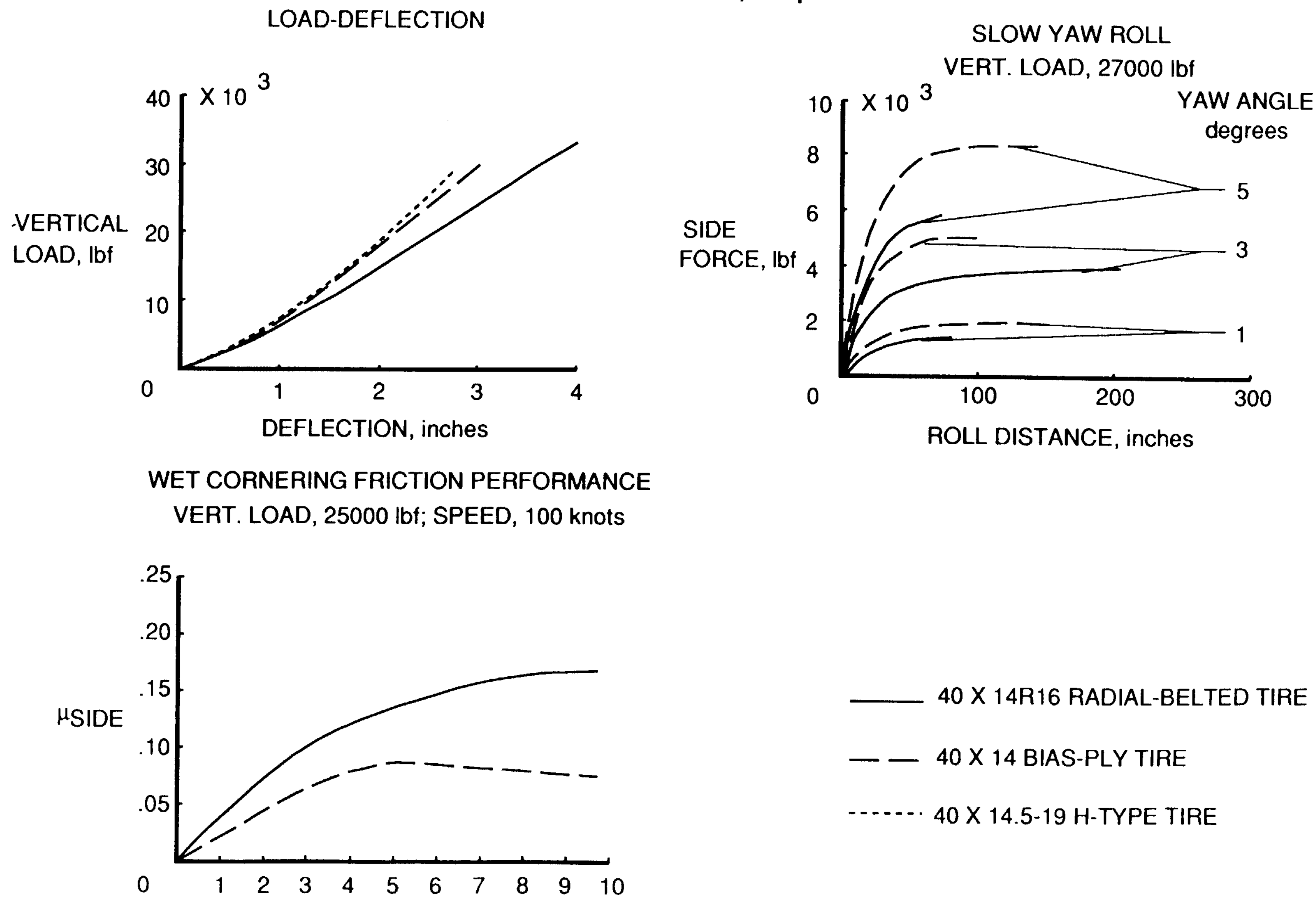


Figure 58 (c).

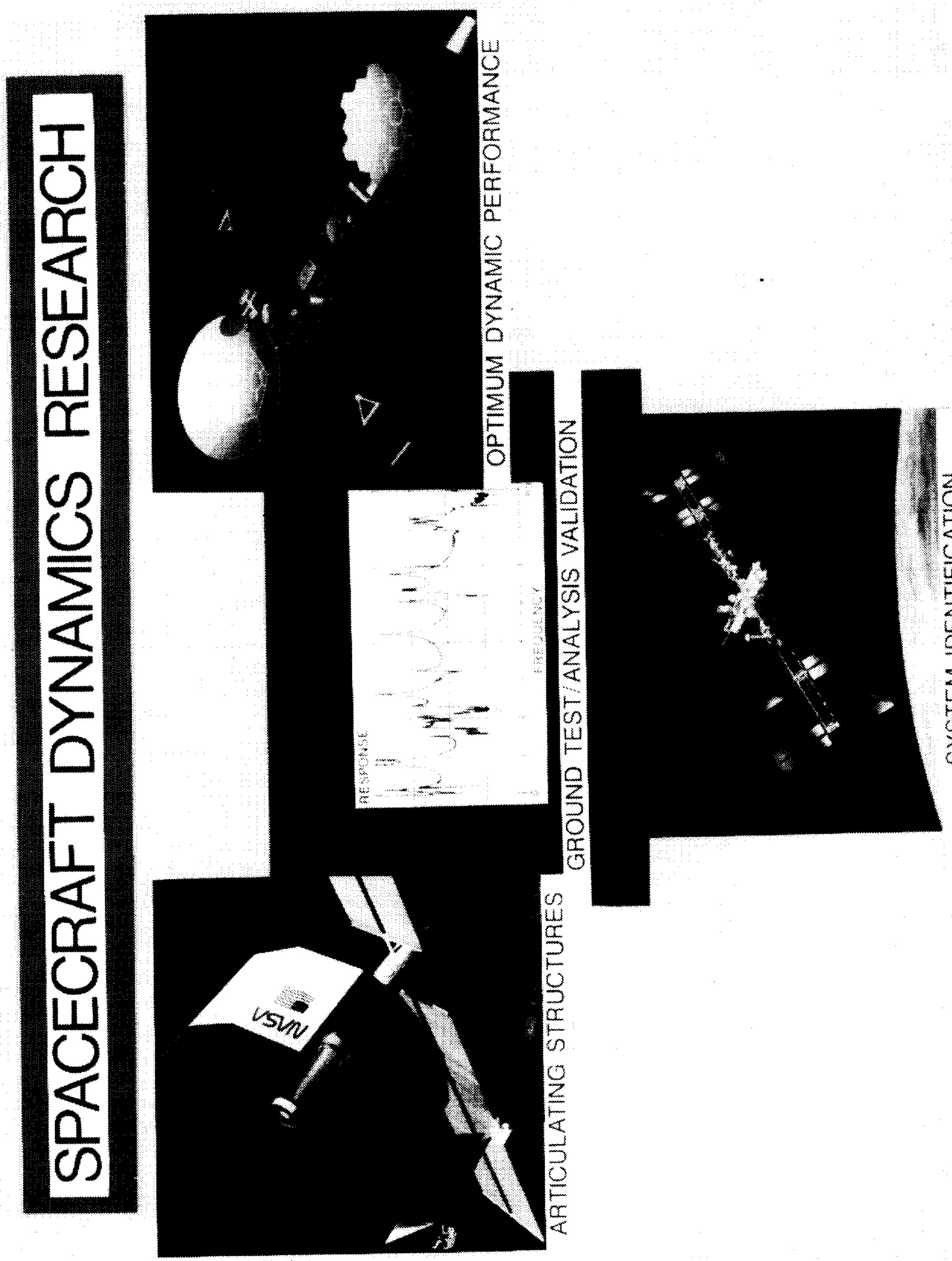


Figure 59.

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SPACECRAFT DYNAMICS BRANCH FIVE-YEAR PLAN

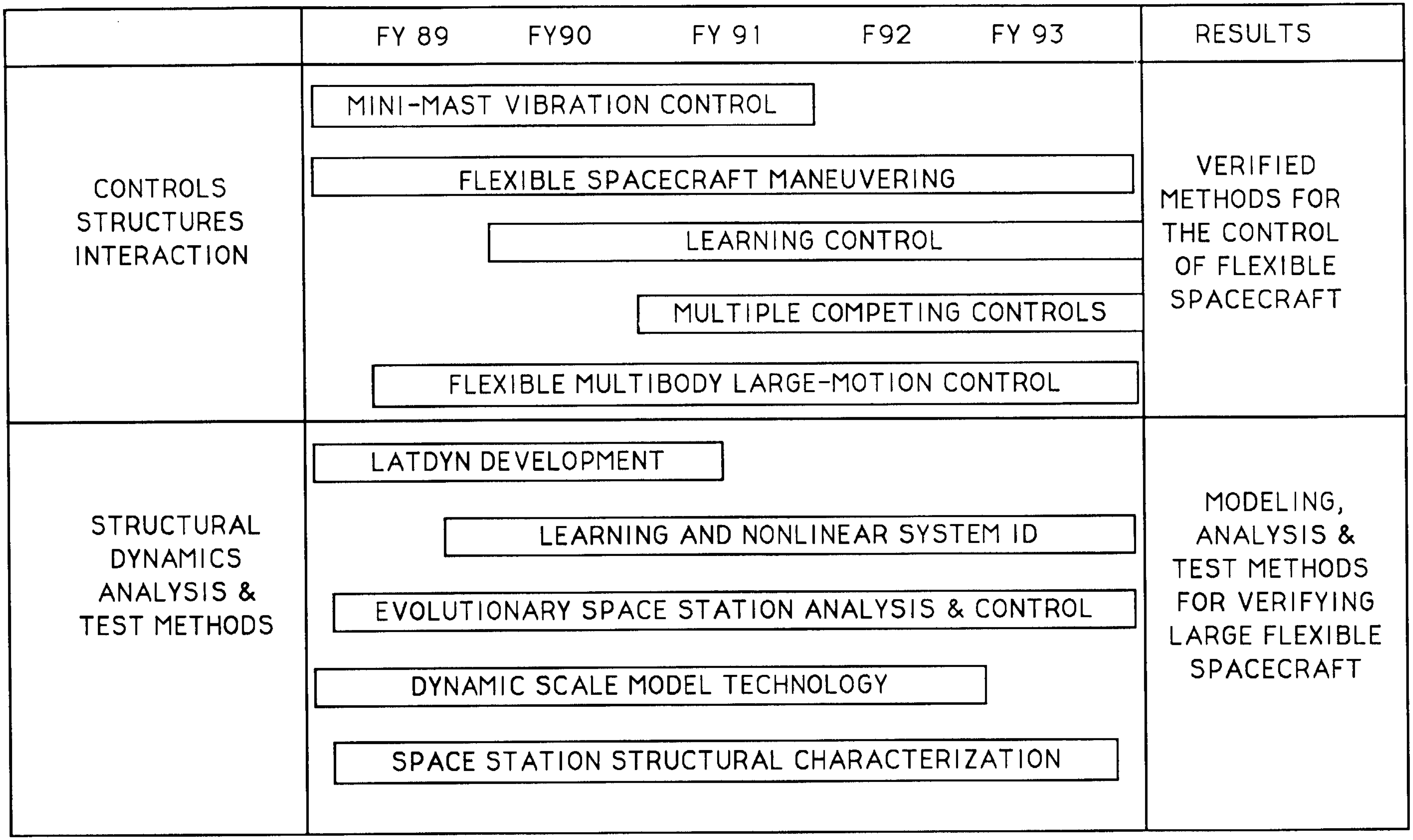


Figure 60.

READINESS OF MINI-MAST CSI TESTBED FACILITY DEMONSTRATED

J.M. Housner, R.S. Pappa, S.E. Perez, R. Miserentino*; K.B. Elliott**; J.L. Sulla and B.L. Williams, Jr.***
*Spacecraft Dynamics Branch,**Old Dominion Univ., and ***Lockheed

RTOP 590-14-61

Research Objective: Actively controlled vibration suppression of flexible large space structures is required for precision pointing of antennas and optical instruments, and assembly of space station and space exploratory vehicles. Since these are low frequency structures, precision control tends to lead to controls-structures interaction (CSI). The objective of this research is to provide a ground-based testbed facility in which active controller designs can be tested and studied in a CSI environment, thus leading to operational designs.

Approach: At LaRC, an 18-bay, foldable, 65-foot, three-longeron, flight-like cantilever truss, known as Mini-Mast, has been instrumented to serve as a ground-based test facility for active control investigation and demonstration. The truss has characteristics associated with future large space structures, namely, low frequencies (starting at 0.8 Hertz), closely spaced modes (e.g. 108 local member modes between 15 and 22 Hertz) and joints which introduce nonlinearity into the truss dynamics.

Accomplishment Description: Displacement, velocity, and acceleration sensors have been installed on the truss. Fifty-one non-contacting displacement transducers are distributed along the truss length. The complete set is used for dynamics characterization and 16 of these are used to provide feedback signals for control. A tip plate and a mid-plate are used as additional instrumentation pallets. Mounted to the tip plate are three angular rate sensors and four accelerometers. The mid-plate has two accelerometers for measuring motion in the plane of the mid-plate. Excitation of the truss is provided by three 50-pound-output shakers located immediately below the mid-plate and/or by three 50-foot-pound output torque wheels located on the tip plate. The torque wheels also act as actuators to control the truss vibrations. As shown in figure 61b, the structural response, after being filtered and digitized, is sent to the controls computer (a CDC Cyber 175) which performs the controller calculations and transmits digitized commands to the torque wheel actuators after conversion to analog signals. A total of 36 channels of sensor measurements are available for controller calculations. Since the actuators can damage the structure, especially if the controller design is unstable, a series of hardware and software safety devices have been installed and safe operational procedures have been implemented. The facility is now available and has been recently utilized by in-house researchers and three out-of-house guest investigators from MIT, Arizona State University, and Purdue University. The results shown were derived for the in-house designed controller. A sinusoidal excitation from one torque wheel is used to produce first mode truss bending for 30 seconds. The controller is then turned on. The results indicate that vibrations are suppressed in about half the time associated with free-decay of the system. More recent results show even more rapid vibration suppression.

Significance: A well-documented ground-based facility for CSI research has been developed to verify and improve active controller designs for flexible space structures. This is the first facility of its kind to utilize realistic flight-like hardware representative of future large space structures.

Future Plans: Further instrumentation of the facility is planned, including the use of proof-mass actuators to augment the torque wheel actuators currently being used. Also, three guest investigators will utilize the facility during the next year.

Figure 61 (a).

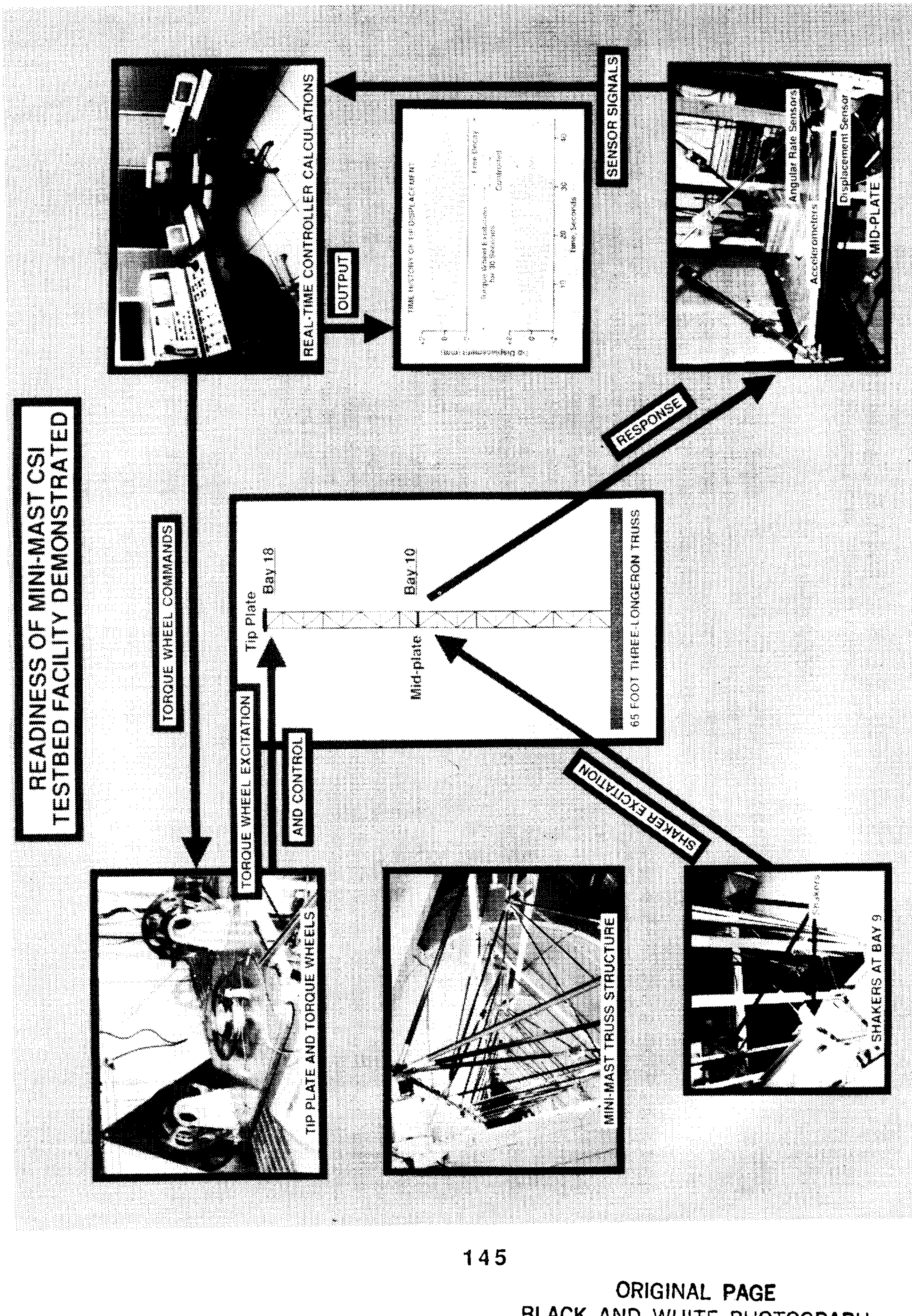


Figure 61 (b).

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

ANALYTICAL SIMULATION PROVIDES PRETEST CHECKOUT OF MINI-MAST CLOSED LOOP EXPERIMENT

S. E. Perez*, Z. C. Kim** and A. E. Stockwell**
Spacecraft Dynamics Branch* and Lockheed**

RTOP 590-14-61

Research Objective: The Controls-Structures Interaction (CSI) Guest Investigator (GI) Program uses a 20-meter deployed flight-quality truss, known as Mini-Mast, as the initial test article. The objective of this research is to ensure the structural integrity of the Mini-Mast during the multi-year life of the program by verifying stability of all control laws and checking the percentage of critical load they impose on the graphite epoxy truss members before approval is granted to proceed with the experiment.

Approach: From a NASTRAN eigensolution identifying 149 natural frequencies between 0-100 Hz, 20 modes were selected for the reduced modal representation of the structure in closed-loop simulations. The 20 modes selected have significant influence on the global response of the structure; modes with more localized effect on individual truss members and higher frequency global modes were omitted to reduce the order of the closed-loop modal plant. GI control laws, provided in matrix format, are incorporated into an analog/digital simulation that includes structure, actuator, filter, and sensor dynamics, as shown in figure 62b. Control and disturbance actuator forces derived from the closed-loop simulations are then applied to the extended 149-mode model and a transient loads analysis is performed that includes the effects of both global and local modes.

Accomplishment Description: More than 15 digital control law matrices, designed by researchers both within NASA and from the university community, have been efficiently incorporated into this comprehensive simulation. The stability of the integrated system with each designed controller was examined, revealing spill-over instability for some of the control laws. In addition, sensitivity studies on various system parameters were completed, aiding the GIs in redesign where necessary. Loads analysis calculations completed on the extended modal model were compared for stress, buckling, and moment critical limits for each truss member type, in addition to global structural displacement and rotation checks made through the same transient analysis on the extended model. Whereas the critical member axial load is based on member buckling and the critical bending moment on strength tests, the combined axial and moment critical limit was arbitrarily established as a linear function, as plotted in the figure. To allow for strength and fatigue uncertainties, a high-risk region was established, highlighted in the graphical representation. Experiments which would produce combined loads in this region were not approved.

Significance: A well-designed analytical simulation process has been developed to safeguard the Mini-Mast test article from overload. While most control laws have been approved for test implementation, this analytical safety net has been effective in identifying unstable control laws and those which, though stable, would impart excessively high loads to the structure.

Future Plans: Further refinements to the modal model are planned, as well as improvements in the dynamic models for sensors and actuators. A more efficient member loads check will also be possible once a data management computer program is completed that will sort NASTRAN output, identifying highest loads for each member type.

Figure 62 (a).

ANALYTICAL SIMULATION PROVIDE PRETEST CHECKOUT OF MINIMAST CLOSED LOOP EXPERIMENT

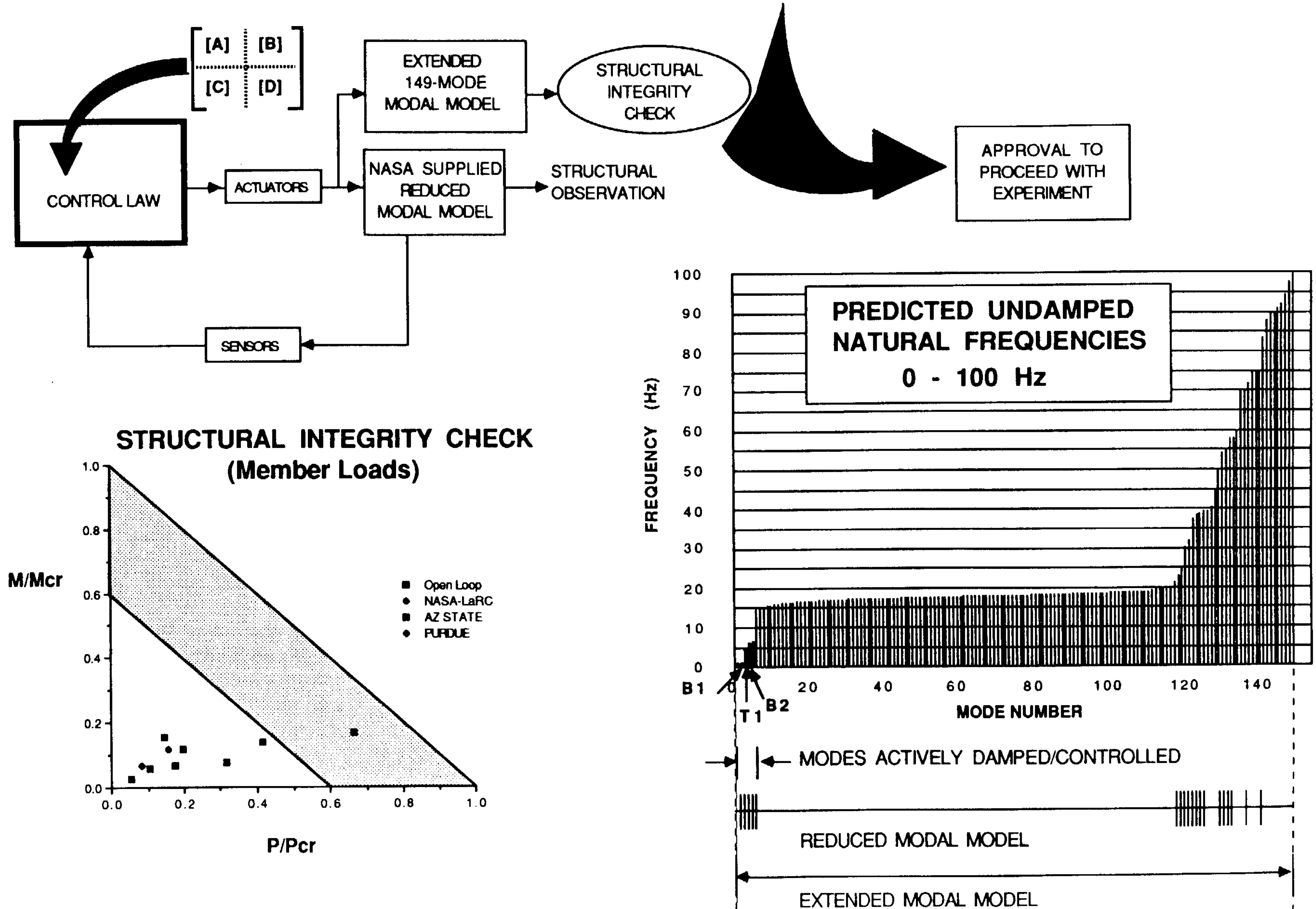


Figure 62 (b).

NEW ADAPTIVE MODAL STATE ESTIMATOR PROVIDES IMPROVED ESTIMATOR PERFORMANCE FOR ACTIVE CONTROL

Chung-Wen Chen*, Jen-Kuang Huang* and Jer-Nan Juang**
*Old Dominion University and **Spacecraft Dynamics Branch

RTOP 590-14-21

Research Objective: For precision active control of large flexible structures, an accurate modal model and accurate knowledge of the modal state are critical items. State estimators, such as the conventional Kalman Filter are used to estimate the system state based on a limited number of sensors and an approximate system modal model. However, the Kalman estimator is known to rely on an accurate knowledge of the system model and noise statistics. The objective of this research is to develop efficient state estimators which work well in the presence of unknown modal model errors and/or measurement noise.

Approach: A new adaptive modal state estimator is developed by combining the two disciplines of modal parameter identification and state estimation. A linear adaptive predictor, which fits measured output, y, to input, u, in a least squares sense, is used as a complementary predictor to the state estimator. The predictor produces the impulse response resulting from predictor and system dynamics, and a reference prediction error which is primarily due to system uncertainties and measurement noise. The impulse response is used by the system identification method, ERA, to update the modal model and state estimator gain so that the estimator error approximately tracks the reference prediction error. This forces the estimator to account for unknown system uncertainties and measurement noise.

Accomplishment Description: An adaptive modal state estimator which operates in the presence of unknown system uncertainties and noise has been developed. A numerical example has shown the potential value of this new estimator. As shown in the figure 63b, the adaptive estimator has been used on a single degree-of-freedom example. In this example, the true signal has been purposely contaminated by noise in order to demonstrate the estimator performance when accurate forcing input and measurement statistics are not known. In the example, the conventional Kalman estimator does track well past sample 300 because of error accumulation. It ignores the noise and continues to estimate the response as decaying to zero in this example. On the other hand, the adaptive estimator, which in this example is turned on after 400 time samples, tracks the response due to forcing input and measurement noise quite well.

Significance: For precision control it is necessary to actively damp response below the noise floor. Also, accurate modal models are often not available for estimating the system state from sensor measurements. This requires the ability to track or estimate in the presence of model uncertainties and/or system noise. This newly developed adaptive modal state estimator has been shown to have this ability on simple systems.

Future Plans: Numerical experimentation on more complex systems is planned and a recursive system identification and state estimation method which can identify modal parameters and state estimation gain from the identified impulse response without using ERA is under investigation.

Figure 63 (a).

NEW ADAPTIVE MODAL STATE ESTIMATOR PROVIDES IMPROVED ESTIMATOR PERFORMANCE FOR ACTIVE CONTROL

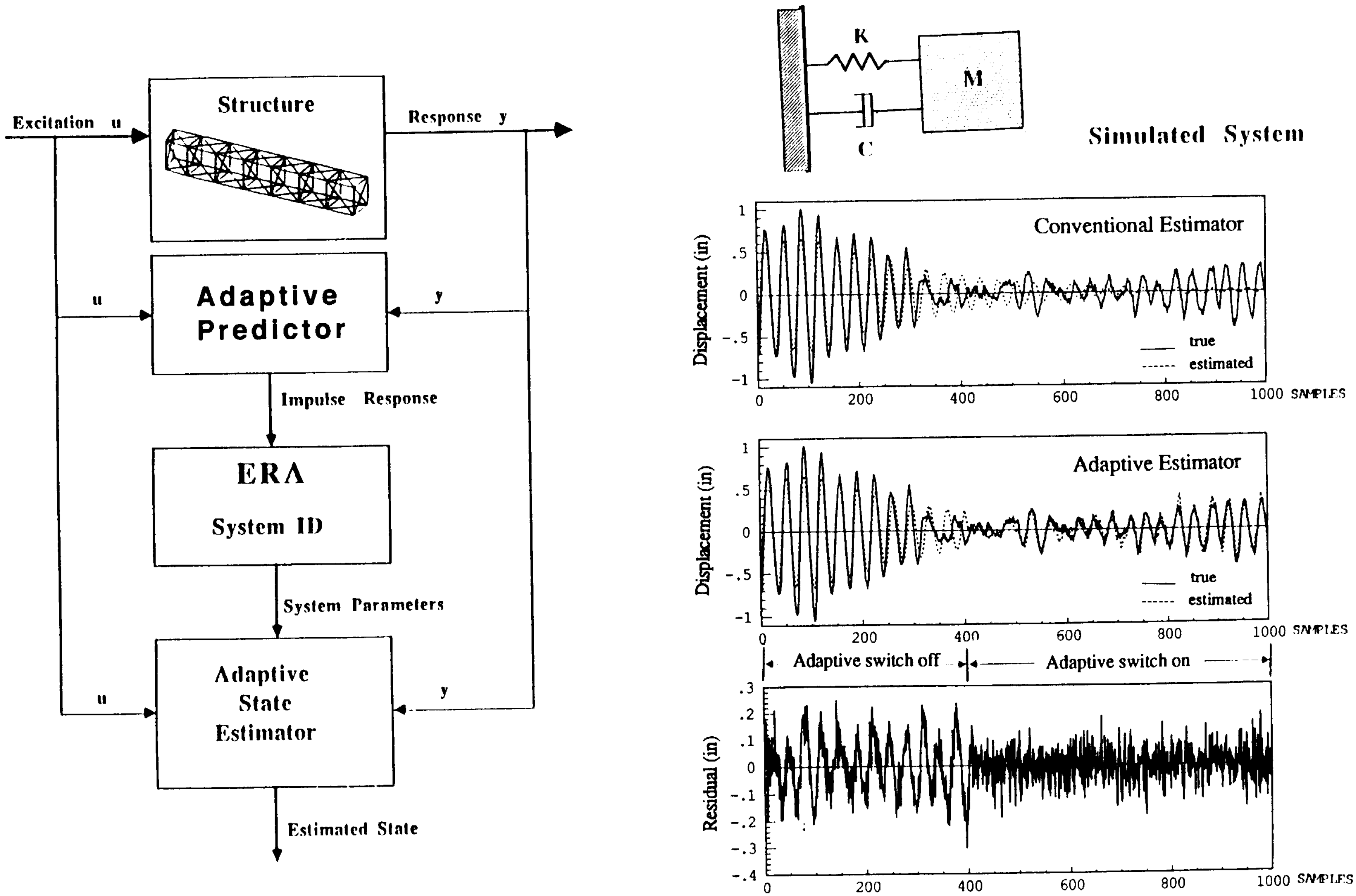


Figure 63 (b).

CANDIDATES IDENTIFIED FOR LINE-OF-SIGHT CONTROLLER ON A PROPOSED CSI STRUCTURE

Kyong B. Lim* and Lucas G. Horta**
Old Dominion Univ.* and Spacecraft Dynamics Branch**

RTOP 590-14-61

Objective: Future NASA missions such as Earth observation platforms require precision of antennas, reflectors, and other mounted instruments. Vibrations of these platforms degrade required pointing performance. Active control of the platform for vibration suppression is one means of meeting performance requirements. The Controls Structures Interaction (CSI) program is exploring, developing, and validating design methods for active control of these systems. This work explores the use of Line-of-Sight (LOS) performance criterion in the controller design.

Approach: The configuration shown in the upper left of figure 64b is employed to assess the use of LOS error criterion in different controller designs. It is composed of a cable suspended 66 ft. L-shape truss with a 16 ft. reflector, a laser source, 9-gas jet trusters, and sensor instrumentation. A laser beam is aimed from the source at a planar target (fixed in space), by reflection off of a mirror surface on the reflector. The deviation of the laser beam off the target is used as a measure of the control system's ability to maintain the LOS.

Accomplishments: The 3-D kinematically nonlinear equation describing the motion of the selected laser beam has been developed, including the effect of structural flexibility and pendulum motion. Three controller design methods have been studied and compared with one other, namely, Robust Eigensystem Assignment (REA), Local Velocity Feedback (LVF), and Linear Quadratic Regulator (LQR). Of the three, only the LQR method uses the LOS performance criterion directly in the design algorithm. However, the LOS performance criterion is used to evaluate all three design methods by performing analytical simulations in which the structure is given an initial distortion in which the first 9 modes participate. This distortion produces an Initial Condition (IC) for the trace of the reflected laser on the target which has an initial 77-inch LOS error off of target center. The trace of the reflected laser beam on the target during the vibrations of the structure are shown in the three bottom traces of the figure. On the basis of the LOS criterion, the REA method produces the least efficient design; it takes 10 seconds to reduce the LOS error to 10 percent of its initial value. This is because this method places active damping in all the nine modes included in the design, irrespective of their contribution to the LOS error. Similarly, the LVF method, though more efficient on this basis than the REA method, is not as efficient as the LQR method which weights the modes depending on their contribution to the LOS error. On the other hand, as shown in the upper right hand corner of the figure, when using a fuel consumption based criterion, the LVR based design tends to require less fuel than the LQR based design.

Significance: A precise statement of the performance criterion based on the LOS error was developed and used for controller evaluation. This can now be applied to refined analysis of the CSI test structure.

Future Work: Investigate system performance when only partial system response information is known. Apply to experimental hardware configuration, and include a detail description of the actuators & sensors which will be used in the actual experiment.

Figure 64 (a).

CANDIDATES IDENTIFIED FOR LINE-OF-SIGHT CONTROLLER ON A PROPOSED CSI STRUCTURE

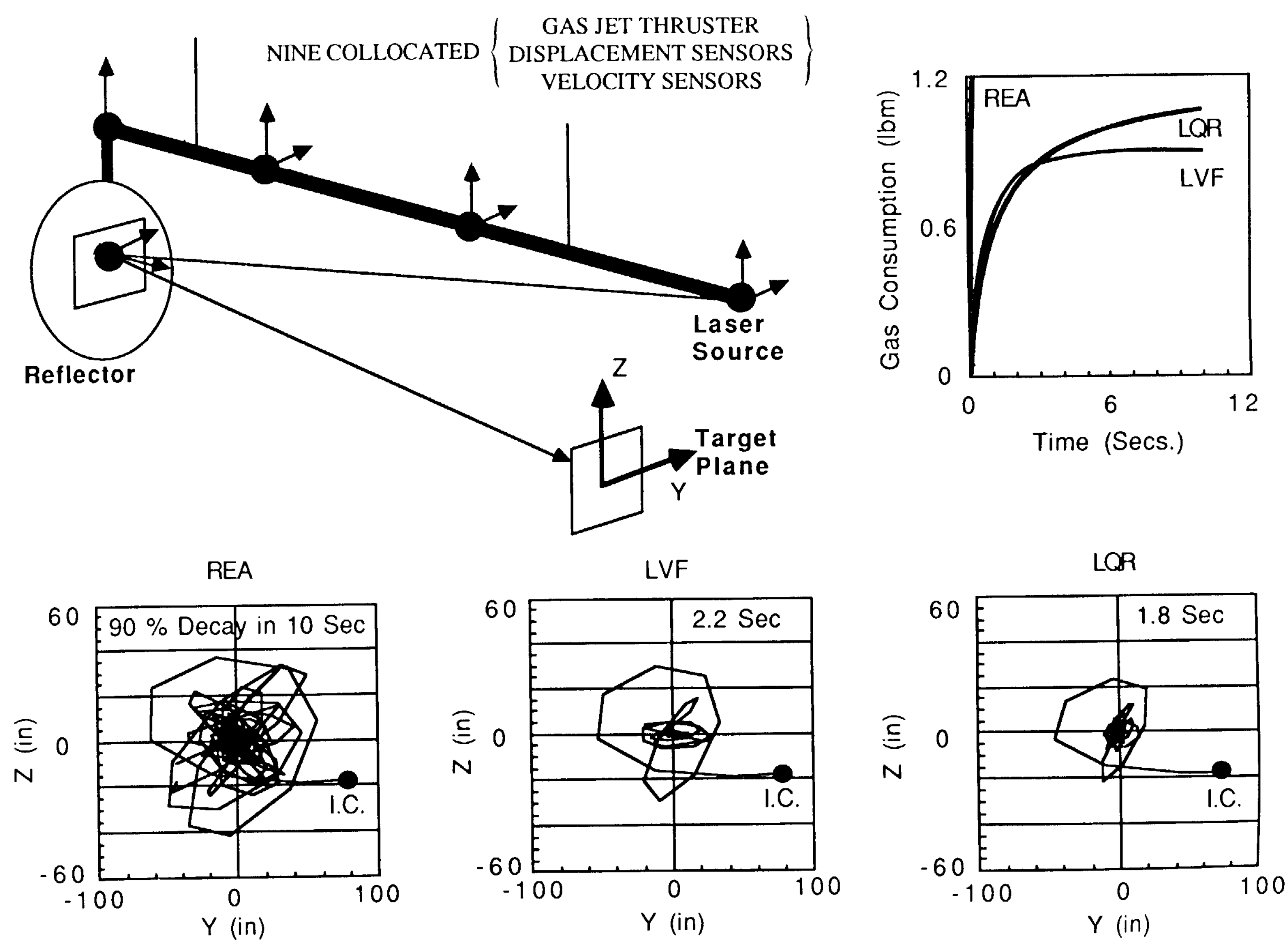


Figure 64 (b).

INTEGRATED DESIGN FOR EARTH POINTING SATELLITE

W. Keith Belvin
Spacecraft Dynamics Branch

RTOP 590-14-61

Research Objective: Pointing requirements for future science missions require spacecraft pointing control systems that are likely to be directly influenced by spacecraft flexibility. The objective of this research is to simultaneously design the control system and the spacecraft structure to permit optimal spacecraft performance while satisfying controller and structural constraints.

Approach: Simultaneous controller and structural design was performed by a new software testbed that incorporates finite element structural modeling, modern control theory design and optimization via nonlinear programming. As indicated in figure 65b, the integrated design code permits the Automated Design Synthesis (ADS) optimizer to govern controller and structural calculations. To enhance the computational attributes of integrated design an efficient transient response algorithm that avoids modal model reduction was incorporated to calculate transient control forces and pointing errors.

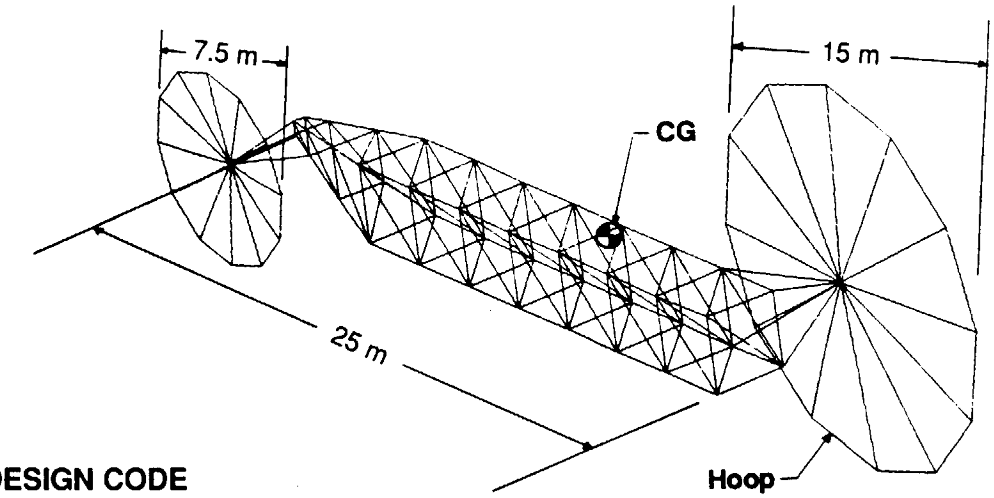
Accomplishments Description: Integrated controller and structural design was performed on a structure typical of geostationary platforms required for future science missions. As shown in the attached chart, a 25 m truss with 15 m and 7.5 m diameter antennas was used in the study. In addition to a three-axis torquer located at the center-of-gravity (CG) for rigid body pointing, 4 small three-axis torquers were placed on the hoop of each antenna. Angular rate measurements at the actuator locations were used for feedback control. A disturbance due to orbit reboost was applied to the satellite and the transient response calculated. In the open-loop mode, 4.5 micro radians of pointing error occurred on the large antenna. The performance constraints were chosen to reduce the pointing error below 0.5 micro radians while maintaining a constant weight for the spacecraft. The objective of the integrated design was to minimize the control force needed to satisfy the constraints. The integrated design used 8 structural design variables, namely the outer diameter of the tubular truss and antenna members, and 12 control design variables that parameterized a set of stable controllers. The figure on the lower right of the attached chart shows the control force history for each iteration of the optimizer. The control force was reduced & all constraints were satisfied. It was found that structural tailoring of the open-loop structure was advantageous as it decreases the number of iterations in the optimizer.

Significance: Integrated design of the structure and controller is predicted to provide the pointing precision required by future science missions while optimally decreasing the controller energy and system mass. Also, the integrated design code permits the examination of computationally efficient design procedures for large spacecraft that exhibit control and structure interactions.

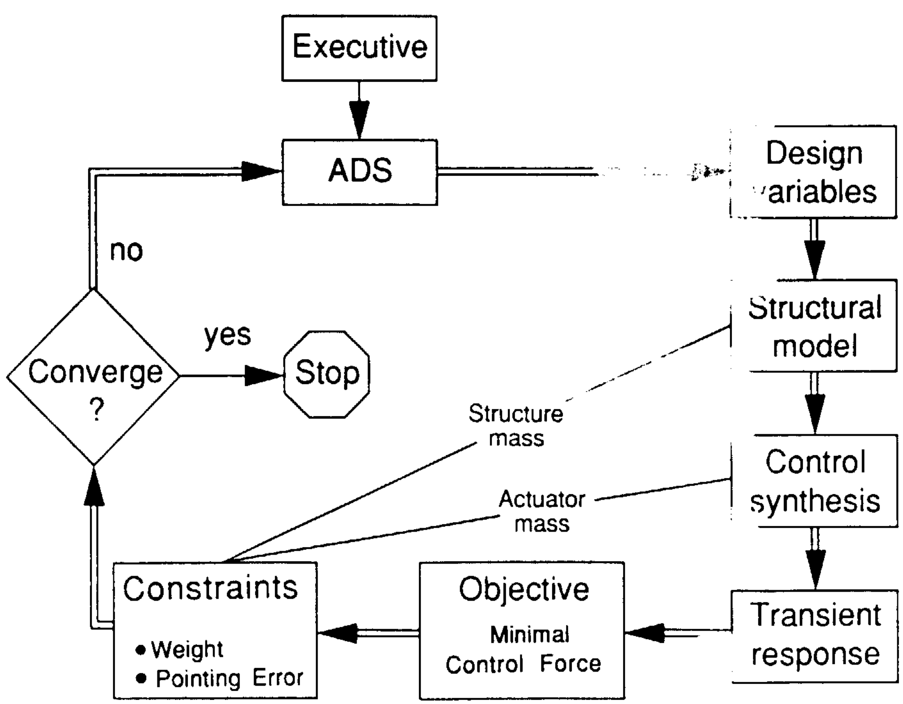
Future Plans: Development of analysis and design methodologies are anticipated as a result of parametric studies using the integrated design code. Computational algorithms will be developed and tested to further enhance the efficiency and applicability of the design code.

Figure 65 (a).

INTEGRATED DESIGN FOR EARTH POINTING SATELLITE



INTEGRATED DESIGN CODE



OPTIMIZATION HISTORY

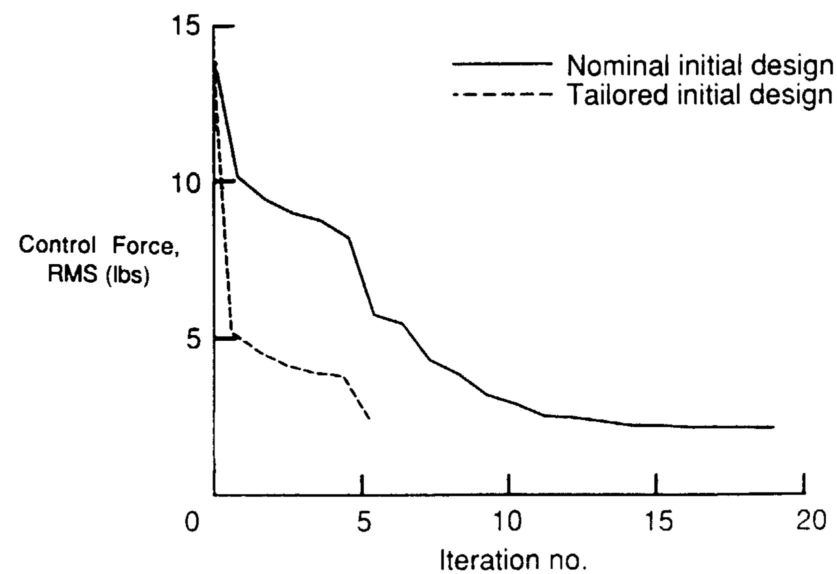


Figure 65 (b).

LEARNING CONTROL METHOD DEVELOPED AND APPLIED IN FLEXIBLE PANEL SLEWING ANALYSIS

Minh Phan*, Richard W. Longman** and Jer-Nan Juang***
NRC*, Columbia University**, and SDB***

RTOP 590-14-21

Research Objective: The objective of this research is to develop simple methods of designing control systems which are capable of learning from previous experience in performing any specified task. The methods are then applied to the control of flexible structures by studying the slewing of a flexible panel.

Approach: When a command is given to a control system, the response is nearly always in error from the command due to modelling uncertainties. The same identical errors are made every time the command is given. The errors may be eliminated by making a control system that can learn from repetitions of the same command or task. This can also alleviate some of the burden on the control system designer by making modelling less critical to good system performance. This is particularly important in the control of large flexible spacecraft. Human behavior very often involves learning through practice, or by gaining experience at a task, and it is therefore to be expected that control systems in the future should be able to learn. Digital or difference equation models are used, and these are converted to obtain a difference equation representation of the dynamics in a repetition domain instead of the usual time or frequency domain. The repetition domain expresses how changes in the control at any time in performing a task, will effect the response of the system during the next execution of the task. The learning control laws developed here make changes in the control history for the next repetition from a linear combination of the errors observed in prior repetitions.

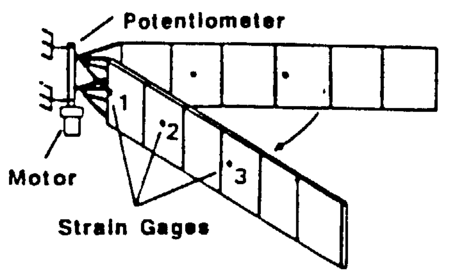
Accomplishment Description: A general mathematical framework for learning control design has been developed. Methods of specifying feasible trajectories are developed and the conditions for convergence of the learning process were established. The approach is illustrated in figure 66b by using a computer model of the slewing flexible panel experiment at the NASA LaRC Spacecraft Dynamics Branch. The curve marked "desired" is used as a reference which is to be duplicated through a learning process. To obtain this reference curve, a good controller, which uses proportional motor angle and strain feedback was designed. Because controller designs are usually based on inaccurate system models, an incorrect model of the panel was used. The incorrect model involved a 25% error in the frequency of the first mode and smaller errors in other modes, and a 10% error in the mass matrix. The curve marked number 1 shows the system behavior when a rather poor controller was designed which uses only proportional feedback of the motor angle. The performance of this controller is improved through the learning process. Curve 2 illustrates the behavior after the learning controller has learned from the initial trial only. After three tries the learning control nearly reproduces the desired response as shown in Curve 3, in spite of the modelling errors. The learning control used is very simple to implement, and only requires storing outputs from the previous run and multiplying them by a gain factor.

Significance: This is the first application of learning control concepts to the control of flexible structures. It is shown that a controller can be made that can learn to give a desired response after a few trials, in spite of rather large modelling errors. The general framework developed, forms a foundation upon which many advances can be made. It is felt that the advances in the theory open up this new field for very rapid and fundamental progress in the near future.

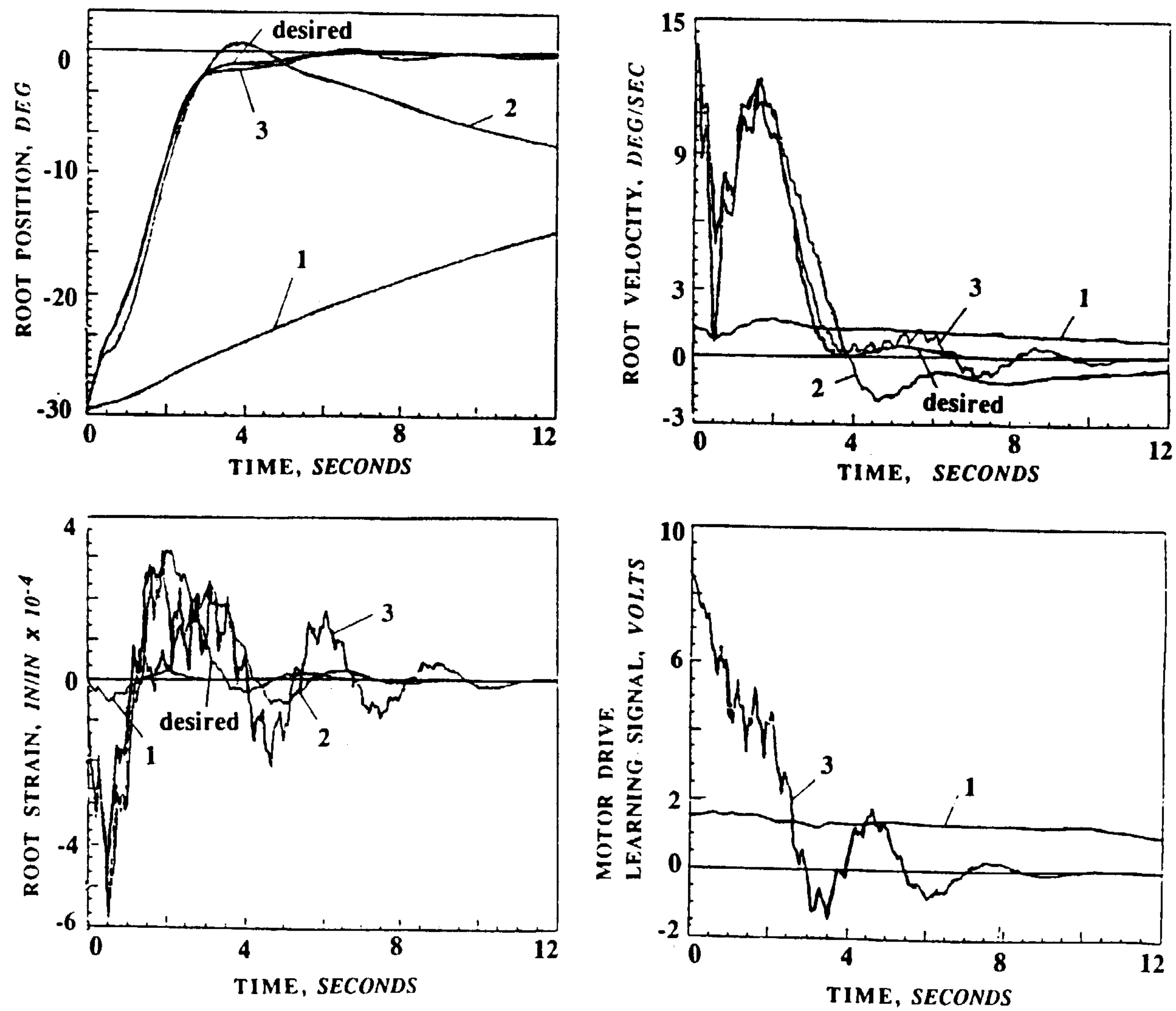
Future plans: As a result of this work a large number of important avenues of research have been identified and work is currently in progress on such issues as optimal learning rates, guaranteed learning, robustness, and endpoint control. For each important new development in the theory, the usefulness will be demonstrated in experiments using the slewing flexible panel apparatus at the Spacecraft Dynamics Branch.

Figure 66 (a).

LEARNING CONTROL METHOD DEVELOPED AND APPLIED IN FLEXIBLE PANEL SLEWING ANALYSIS



30-DEGREE SLEW IN 6 SECONDS



LEGEND:

1. RESPONSE WITHOUT LEARNING
2. LEARNED RESPONSE AFTER ONE MANEUVER
3. LEARNED RESPONSE AFTER THREE MANEUVERS

Figure 66 (b).

REBOOST ANALYSIS DEVELOPED AND UTILIZED FOR SPACE STATION STRUCTURAL CHARACTERIZATION EXPERIMENT

P. A. Cooper*, T. W. Lim**, M. J. Kaszubowski+, and Z. N. Martinovic++
Spacecraft Dynamics Branch*, Lockheed**, Computer Techn. Assoc., Inc.+,
and Analytical Mechanics Assoc.++

RTOP 488-20-07

Research Objective: The Space Station Structural Characterization Experiment (SSSCE) provides a unique opportunity to perform a series of on-orbit modal tests for Space Station Freedom (SSF) and its intermediate flight configurations. Since SSF reboost is a source of excitation for modal identification and since it is desirable not to introduce experiment specific sources, the potential of using reboost to extract modal data must be examined. The objective of this research is to define the excitation experienced by the first flight configuration (MB-1) under an orbital reboost maneuver and to utilize this excitation to define those modes which could be identified through on-orbit modal identification algorithms.

Approach: A finite element model of the MB-1 configuration was developed (see figure 67b) and the undamped structural modes below 5 Hz were determined. Also, a closed-loop attitude control system using the station's proposed Reaction Control System (RCS) jets based on a burn-coast-burn reboost scenario was developed to reboost the MB-1 from 220 nautical miles (NM) to 234 NM. The resulting RCS excitation was employed to examine structural responses at the various points of interest in the structure so as to determine which modes could be identified from a reboost maneuver excitation. Identified modes were defined as those which account for 95% of the global acceleration response of the structure in a modal superposition. A preliminary study on the optimal accelerometer locations and orientations to identify these modes was made using a discrete optimization technique called "simulated annealing" which produces a near-global optimum subset of accelerometer locations and orientations from a provided set of candidates. In this case a set of 20 optimal locations and orientations were found from 91 candidates.

Accomplishment Description: Nineteen modes were identified from a total of 88 flexible modes in the analysis as being ones which were sufficiently excited by reboost. These modes exhibit truss motion coupled with motion of the photovoltaic (PV) arrays and the thermal radiators. Also identified were all those modes which significantly contribute to the local flexible motion in the vicinity of the avionics pallet and which are important for attitude control. Moreover, results indicated that the reboost maneuver can be performed efficiently using the burn-coast-burn scenario and the closed-loop attitude control system designed using the RCS jets. The closed-loop attitude control system was stable even with unfiltered elastic response signals contaminating the feedback signal.

Significance: Techniques necessary for on-orbit modal testing definition such as finite element model development, excitation definition, mode selection, and optimal sensor location and orientation have been examined and the set of modes which could be identified due to a reboost excitation was established for the MB-1 configuration. This configuration as well as others in the SSF build-up have undergone revision, but the techniques studied here are still applicable. The finite element model has been distributed to the Johnson Space Center and contractors, and the model has been used by the contractors in evaluating their attitude control system in a control/structure interaction environment.

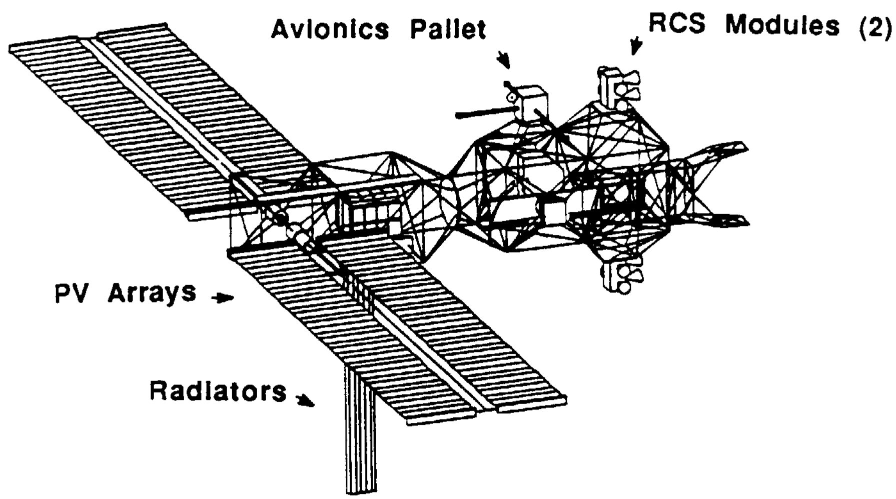
Future Plans: Research effort on the optimal sensor location for modal testing will be continued. A test article among the early-build Space Station Freedom configurations will be selected and a complete end-to-end on-orbit modal testing simulation from structural modeling to modal parameter identification is planned.

Figure 67 (a).

REBOOST ANALYSIS DEVELOPED AND UTILIZED FOR SPACE STATION STRUCTURAL CHARACTERIZATION EXPERIMENT

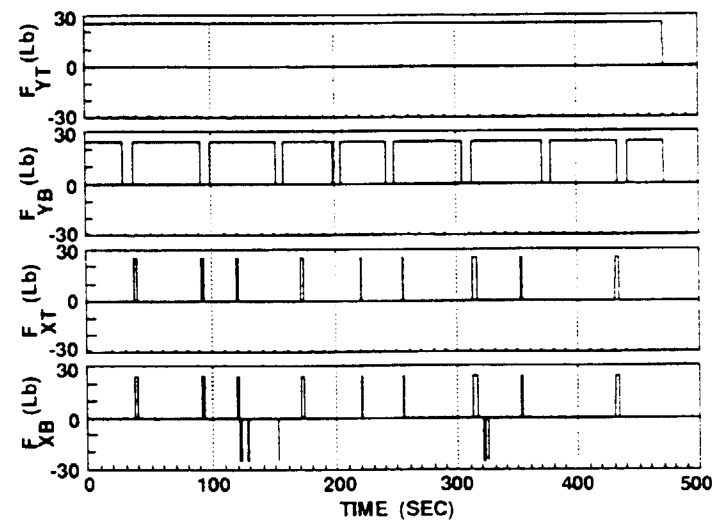
SELECTED TEST ARTICLE

SSF FIRST FLIGHT CONFIGURATION



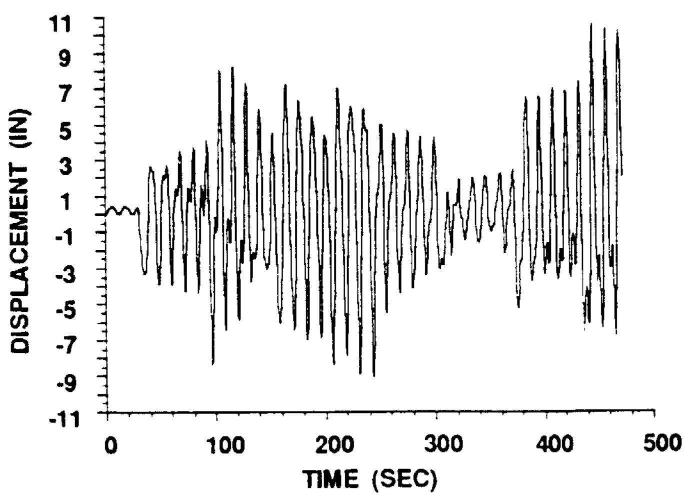
DEVELOPED POSSIBLE REBOOST EXCITATION

RCS JET FIRINGS DURING REBOOST MANEUVER



INVESTIGATED STRUCTURAL RESPONSES

PV ARRAY TIP Z DISPLACEMENT



SELECTED BEST ACCELEROMETER LOCATIONS

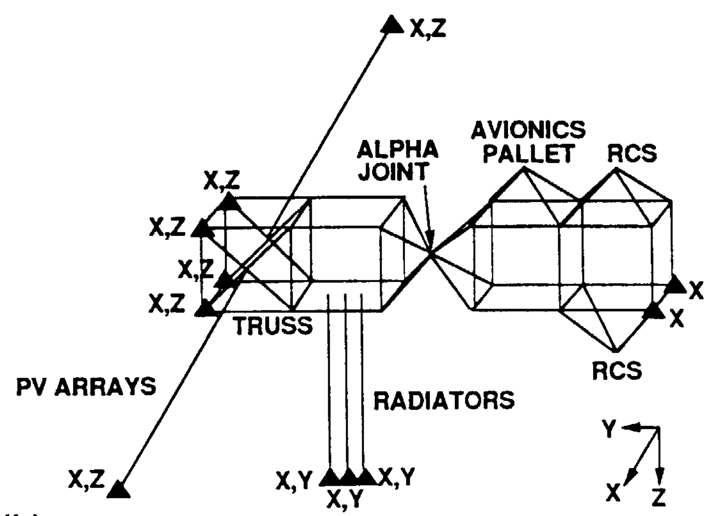


Figure 67 (b).

MARS EVOLUTIONARY SPACE STATION STUDIES INDICATE MINIMAL IMPACT ON SOLAR DYNAMIC PERFORMANCE DURING REBOOST

P. A. Cooper*, T. W. Lim**, J. K. Ayers** and A. E. Stockwell**
Spacecraft Dynamics Branch*; and Lockheed**

RTOP 488-20-07

Research Objective: One concept for a manned mission to Mars uses an evolutionary version of Space Station Freedom (SSF) as a transportation node. The station is modified by the addition of dual keels, an upper and lower boom, additional laboratory and habitation modules, increased power provided by Solar Dynamic (SD) systems and an assembly platform for a Mars Piloted Vehicle (MPV). The objective of this research is to calculate the low-frequency dynamics of the Mars Evolutionary Reference Configuration (MERC), develop an attitude-controlled reboost procedure, and investigate the flexible dynamic response at critical points of the station during a reboost.

Approach: A detailed finite element model of the MERC was developed (see figure 68b) based on a recent NASA baseline model of SSF. The model consists of 2420 beam elements, 434 concentrated mass elements and 1000 nodes. Rigid-body mass properties were computed, and an eigenvalue analysis was used to obtain the undamped natural frequencies and mode shapes below 2 Hz. A Reaction Control System (RCS) jet-firing sequence was determined to reboost the station from an orbit of 220 NM to 230 NM, and a closed-loop attitude control system was developed to maintain the attitude of the station to ±3 deg. during the reboost maneuver. Feedback signals in the control system contain both rigid-body rotation information and the local elastic rotations at the attitude sensor. Elastic dynamic response was calculated for critical points of the station during the reboost procedure.

Accomplishment Description: The MERC space station finite element model which was developed from the SSF model predicts 126 modes below 2 Hz and has a fundamental frequency which is 50 percent lower than the corresponding frequency for SSF. A reboost procedure developed for the MERC maintains the attitude of the spacecraft to ±3 deg. during the jet firings. The jet-pulsing frequency was well below the fundamental frequency of the station and no adverse effects due to excessive structural response were observed. The local elastic rotation at the solar dynamic location during reboost never exceeded 0.1 deg. control requirement and should therefore present no control problems.

Significance: A study of the low-frequency characteristics of an evolutionary space station configuration has shown that although the structure is significantly larger than SSF and has more than twice as much mass, its attitude can still be controlled during an orbital reboost maneuver without excessive control/structure interaction. Flexible motion during the firing of the RCS jets should not adversely affect pointing accuracy of the solar dynamic power system.

Future Plans: Continue control/structures studies of evolutionary concepts for manned Lunar and Mars missions as these develop.

Figure 68 (a).

Mars Evolutionary Space Station Studies Indicate Minimal Impact on Solar Dynamic Performance During Reboost

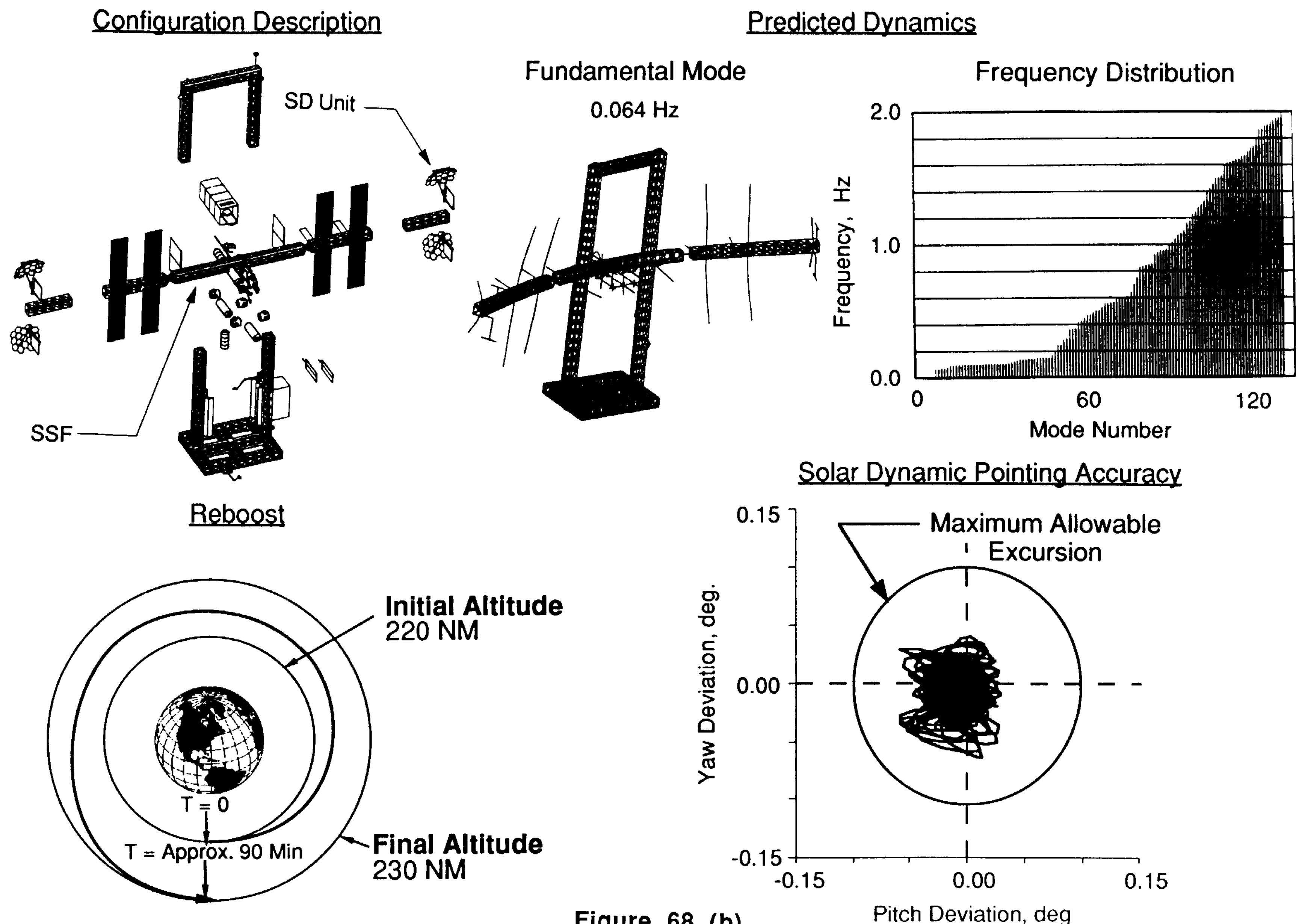


Figure 68 (b).

SPACE CRANE DYNAMICS AND OPERATIONS INVESTIGATED USING LATDYN

C-W. Chang*, G-S. Chen**, and S-C. Wu*
*COMTEK, **Jet Propulsion Laboratory

RTOP 488-20-07

Research Objective: An important component of an orbiting construction facility is a manipulator arm which is used to move large components during assembly operations. Its dynamic behavior is of significant importance since inertial loads and flexible deformation govern the speed with which assembly operations can be performed. The objective of this research is to study the dynamic performance of a conceptual Space Crane during the gross motion phase of the Crane's manipulator manuevers.

Approach: Space Crane operations were studied using LATDYN which is a new computer code for modeling the non-linear deformation and inertial effects involved in large angular motions of flexible multibody systems. The Crane's performance during the gross motion phase of the operational maneuvers was simulated and the effect of flexibility on maneuver precision was determined. A combiniation of inverse and forward dynamic analyses was used to perform the study. Inverse analysis refers to a solution procedure in which the motion and speed of the system are prescribed and the driving torques (or forces) required to produce them are found. Forward analysis refers to the normal procedure of solving for the motion of a dynamic system when the driving forces or torques are prescribed.

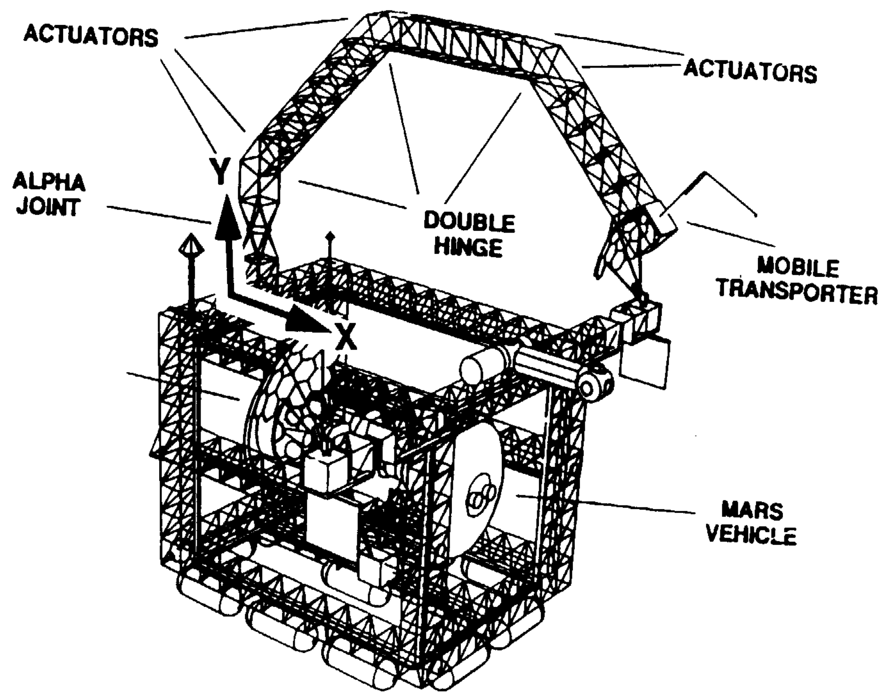
Accomplishment Description: The prescribed trajectories of the interconnection joints between the Crane's members, which are associated with the desired operational maneuver of the Crane, are shown in figure 69b. With the members assumed rigid, LATDYN was used in an inverse mode to derive the motor torques which act at each of the interconnection joints and which are required to produce the desired maneuver. The torque histories were then input to a second LATDYN execution, but this time with flexible Crane members. The resulting flexible Crane motions were then compared to the desired motions and deviations determined. Also, since LATDYN is a finite element based program, stresses in the members and joints were easily determined to provide additional design information. Though the deviations are small compared to the Crane's overall large dimensions, they are not insignificant. Thus, these results indicate that for the desired maneuver trajectory and speed considered here, the Crane's flexiblity must be accounted for in performing precision operations.

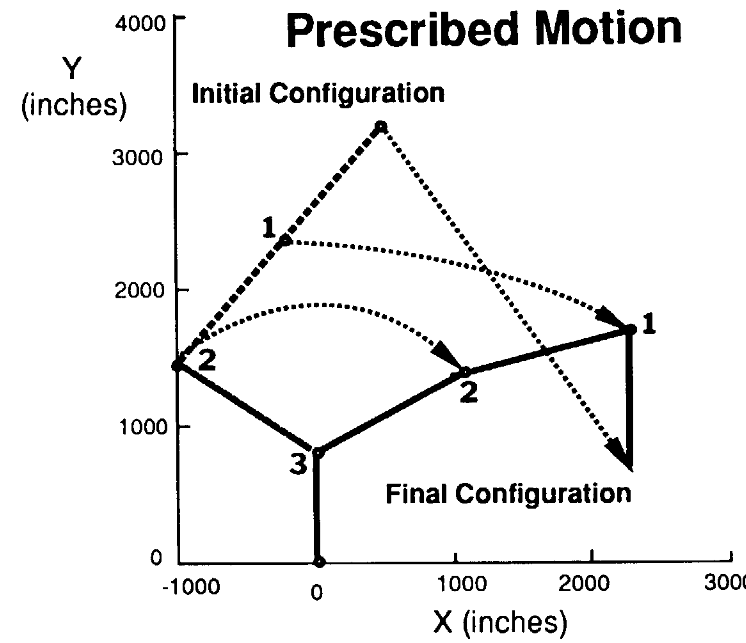
Significance: The results demonsatrate that this procedure provides a systematic method for computing tracking errors due to flexibility in multibody systems such as the Space Crane.

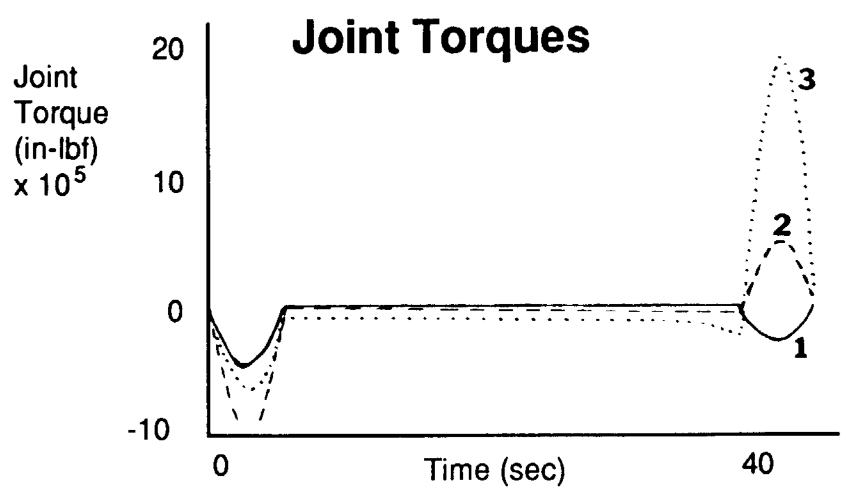
Future Plans: More thorough studies are planned inluding the use of active controls for vibration suppression.

Figure 69 (a).

Space Crane Dynamics and Operations Investigated Using LATDYN







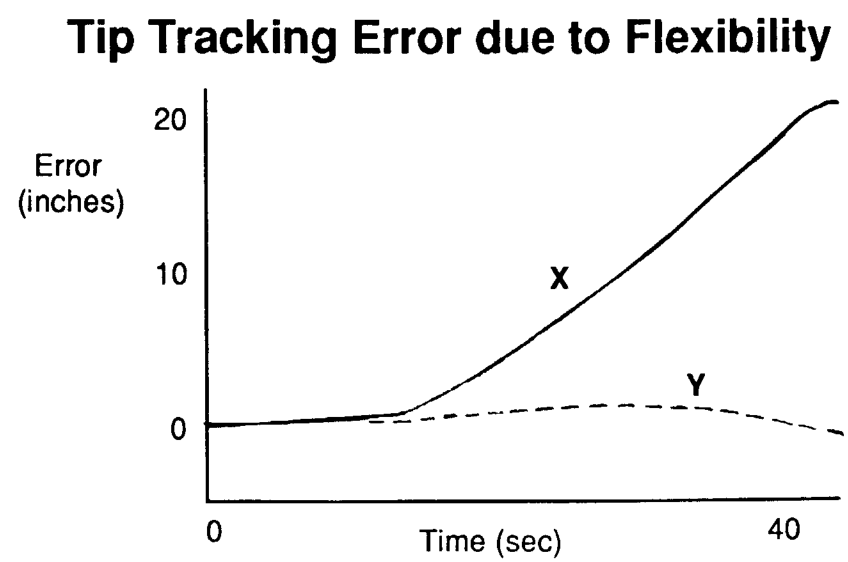


Figure 69 (b).

CONFIGURATION AEROELASTICITY

FY 1990 PLANS

- Complete final design of CoF upgrades to TDT gas reclamation system
- Complete testing of A-12 model in TDT in support of DoD program
- Complete unsteady aerodynamic tests of NACA 0012 wing model in TDT
- Complete design/fabrication of oscillating shock model for TDT tests
- Complete TDT test of advanced hingeless rotor system on ARES
- Complete demonstration tests of ARES 1.5 in Hover Facility and TDT
- Complete tests/analyses of extension-twist-coupled rotorcraft blades
- Complete Bell ACAP difficult components study

Figure 70.

AIRCRAFT AEROELASTICTY

Maynard C. Sandford
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective: The objectives in the aircraft aeroelasticity technical area are (1) to determine and solve the aeroelastic problems of current designs, and (2) to develop the aeroelastic understanding and prediction capabilities needed to apply new aerodynamic and structural concepts to future flight vehicles.

Approach: The types of research included in the aircraft aeroelasticity area are illustrated in figure 71b. This research is a combination of experimental and complementary analytical studies. The experimental work focuses on the use of the Langley Transonic Dynamics Tunnel (TDT) which is specifically designed to meet the unique needs of aeroelastic testing. On occasion flight research programs are undertaken when it is necessary to simulate important parameters that cannot be accurately accounted for in ground-based facilities. Often research is a cooperative effort with other government agencies and/or industry.

Status/Plans: Work for the coming year includes several activities, some of which are described in the following statements. Flutter clearance testing of the Navy A-12 will continue with additional tests in FY 1990. Analytical and experimental investigations will be conducted to provide additional flutter data in support of the High Speed Civil Transport (HSCT). Joint efforts with NASP contractors, primarily McDonnell Douglas and Rockwell International, will include investigations of (1) panel flutter; (2) airfoil thickness, planform leading edge sweep angle, and hinge stiffness effects on aileron buzz; (3) fuselage shape and flexibility effects on flutter; and (4) evaluation of concerns about engine inlet leading edge panel flutter.

Figure 71 (a).

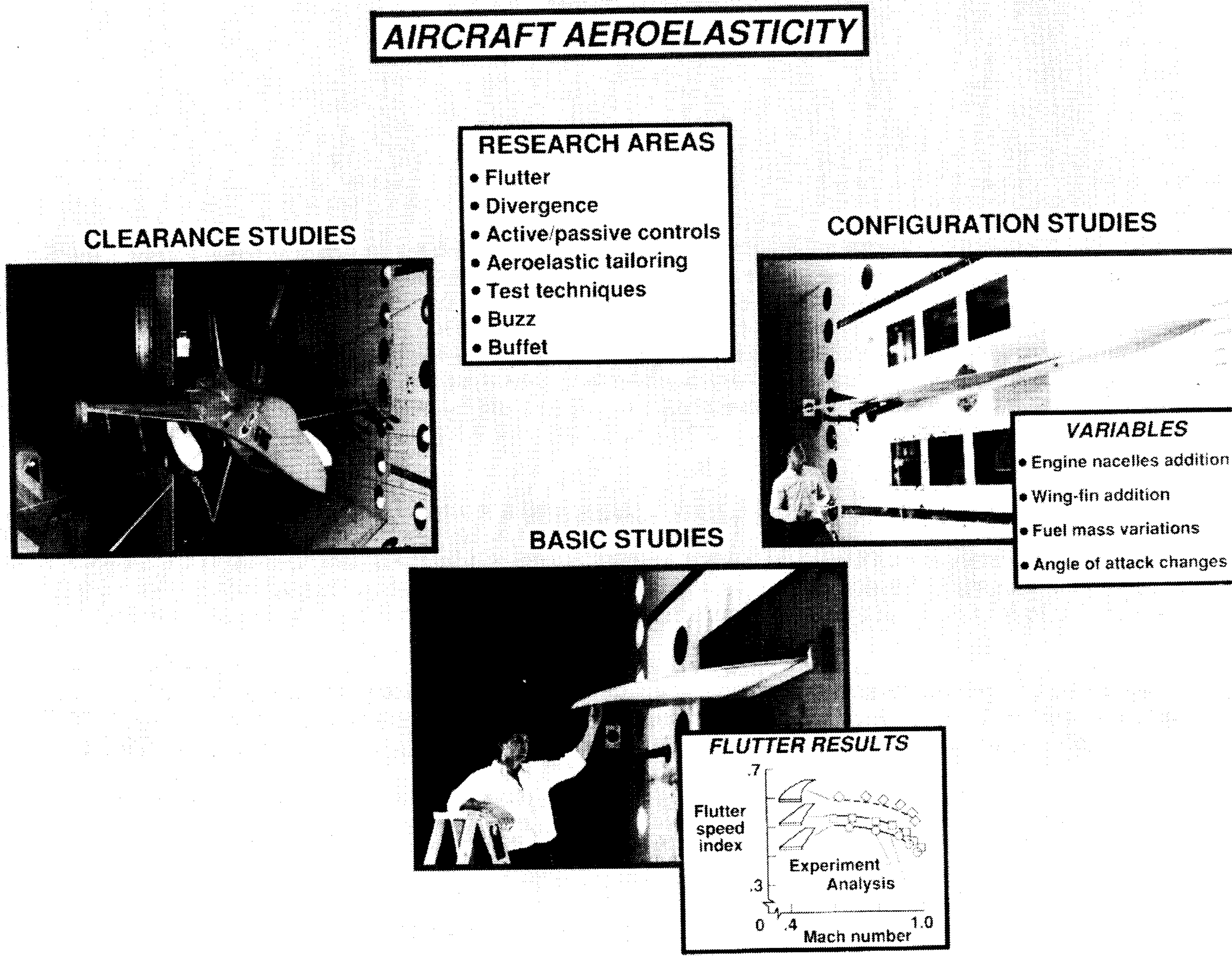


Figure 71 (b).

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AIRCRAFT AEROELASTIC VALIDATION

Clinton V. Eckstrom
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective: The objectives in the Aircraft Aeroelastic Validation area are to (1) provide test data for validating capabilities of computational aeroelasticity codes, (2) increase physical understanding of unsteady flow phenomena, and (3) provide test data useful for developing empirical design methods where computational methods need further development.

Approach: This research is a joint effort of the Configuration Aeroelasticity Branch and the Unsteady Aerodynamics Branch. Critical aeroelastic conditions needing validation involve dynamic, mixed flow phenomena at off-design conditions near envelope boundaries. The purpose is to measure dynamic response and stability and to measure corresponding flow field characteristics as shown in figure 72b. Planned areas of testing for a multi-year effort include measurement of conventional flutter, limit-cycle oscillation flutter, non-classical flutter, the effects of vortex-flow on flutter, and burst-vortex induced buffeting.

Status/Plans: An instrumented rigid rectangular wing model has been designed and is now being fabricated. Activities prior to testing will include installation and checkout of instrumentation and development and checkout of computer programs for acquisition and reduction of static and dynamic data. Plans are for the model to be tested on the Pitch and Plunge Apparatus in the TDT during FY 1990. Testing will provide accurate flutter characteristics and limited unsteady pressure measurements at flutter conditions through the transonic speed range. A second series of models to be tested during FY 1990 are thick circular arc airfoils which experience a natural shock-induced aerodynamic oscillation. The objective is to measure unsteady force characteristics on a rigid model and then observe aerodynamic and structural coupling characteristics for models with varying bending/torsion stiffnesses.

Figure 72 (a).

AIRCRAFT AEROELASTIC VALIDATION

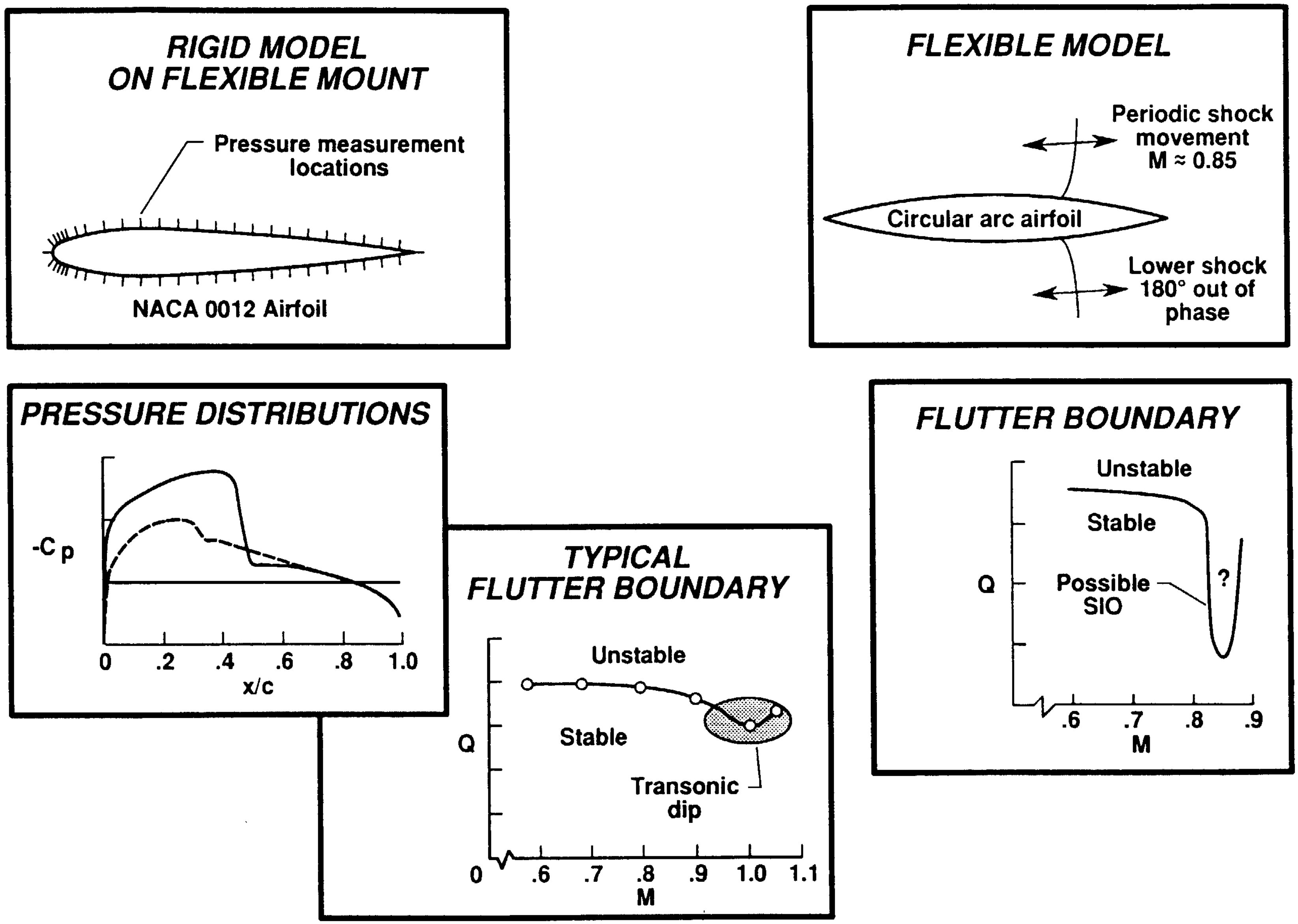

Figure 72 (b).

ROTORCRAFT AEROELASTICTY

William T. Yeager, Jr. (Army)
Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective: The objectives in this technical area are to (1) conduct research in the aeroelastic, aerodynamic, and dynamic characteristics of rotors; (2) support design of advanced performance rotorcraft in the areas of loads, vibration, and aeroelastic stability; and (3) develop the experimental and analytical techniques necessary to extend wind tunnel and laboratory capabilities to future research requirements and opportunities.

Approach: This research area is a joint effort of SDyD and the U.S. Army Aerostructures Directorate which is co-located at Langley. The in-house civil research is supplemented by industry contracts and university grants. As indicated in figure 73b, experimental studies are conducted in the TDT and the General Rotor Aeroelasticity Laboratory (GRAL). Analytical studies include the use of existing methods and the development of new and improved methods. The Aeroelastic Rotor Experimental System (ARES) is the key test bed in the experimental studies. This system which has drive mechanisms, a strain-gage force and moment balance, and other equipment housed in a generic fuselage shape provides a means for studying a variety of rotor systems in simulated forward flight in the TDT and in hover in the GRAL. Two advanced versions of the ARES are being developed which make it possible to better model the coupling of the rotor and the body. The ARES 1.5 design mounts in the metric section of the existing ARES model on a static gimbal or "soft mount" to allow adjustment of the model fixed system stiffness and damping characteristics in both pitch and roll. The ARES II design mounts the metric section of ARES on a platform supported by six computer controlled hydraulic actuators which are used to obtain the desired body roll, pitch, yaw, side, normal and axial motion.

Status/Plans: All parts of the ARES 1.5 have been fabricated and the model has been assembled. System frequency and damping characteristics will be determined, hover tests will be conducted, and then the model will be used for tests in the TDT. Fabrication has also been completed on parts for the ARES II model and assembly should take place in CY 1990. Additionally, a closed-loop controller for ARES-II is being developed in-house. A dynamic calibration of the ARES strain-gage balance was conducted and data are being reduced. Also data from a wind-tunnel test of the tailored Growth Blackhawk (GBH-T) are being evaluated to determine the reduction in fixed-system vibratory loads resulting from rotor blade modal shaping. A test in the TDT of a new ARES hingeless rotor design will be conducted in FY 1990. A Syracuse University grant to modify the CAMRAD code to provide the capability of analyzing multiple-load-path rotors is continuing.

Figure 73 (a).

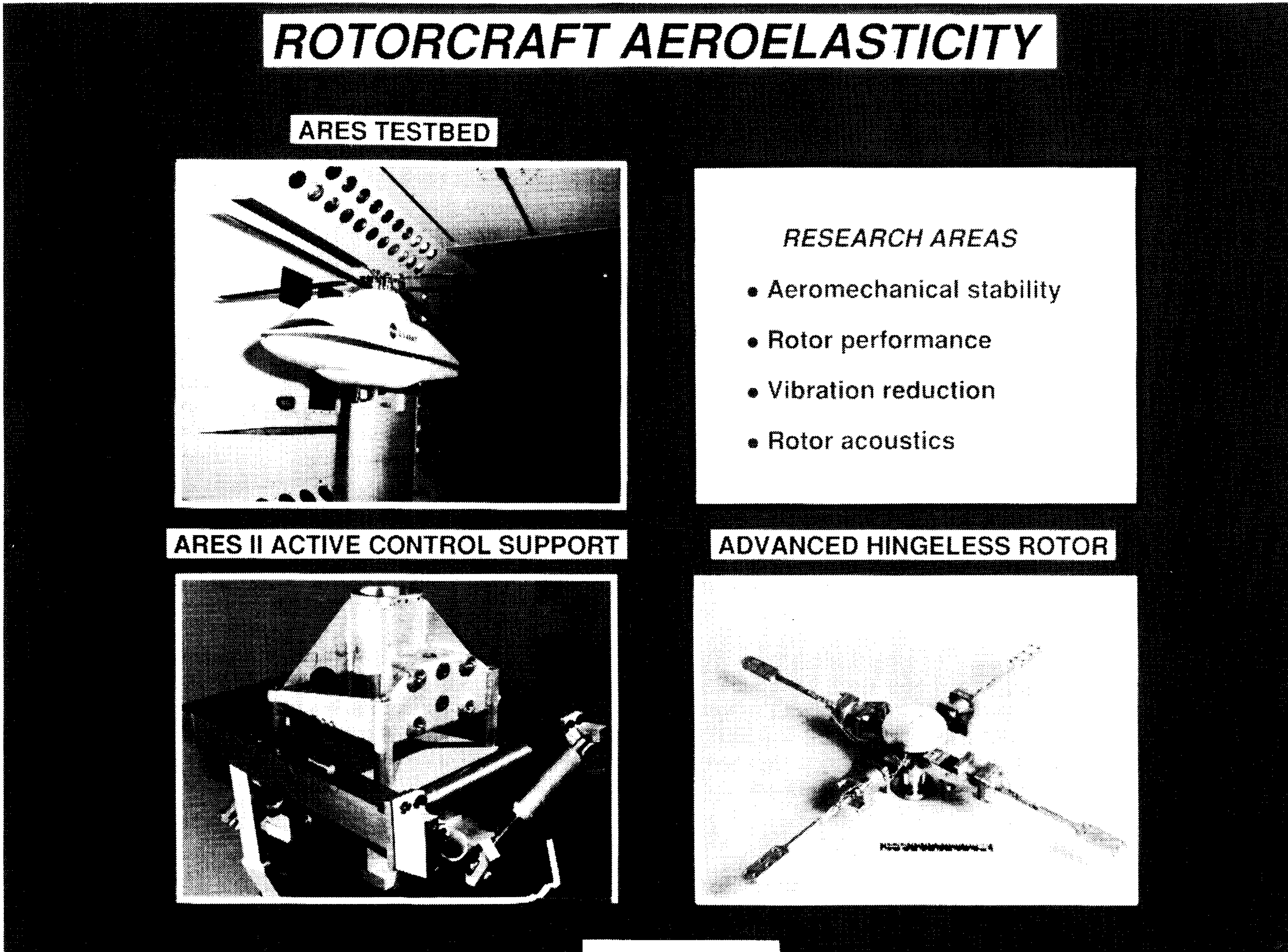

Figure 73 (b).

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ROTORCRAFT STRUCTURAL DYNAMICS

Raymond G. Kvaternik
Configuration Aeroelasticity Branch

RTOP 505-63-51

Research Objective: Helicopters are prone to vibrations which can seriously degrade both service life and ride quality. With only a few exceptions, vibrations problems have not been identified and attacked until the flight test and operational stages of a new helicopter. There is now a recognized need to account for vibrations during the analytical phases of design. The advent of modern methods of computer analysis has provided the opportunity to achieve such a capability. The objective is to establish the critical elements of the technology base needed for the development of a superior finite element dynamics design analysis capability in the U.S. helicopter industry.

Approach: The rotorcraft structural dynamics program in the area of vibration predictions is a cooperative effort involving NASA, the Army, academia, and the helicopter industry in a series of research activities aimed at: (1) discovering and removing technology barriers to analytical modeling for rotorcraft vibrations; and (2) establishing critical elements of the technology base needed for development of a superior finite element dynamics design analysis capability. During the industry phase of the program, referred to as DAMVIBS (Design Analysis Methods for VIBrationS), teams from the four major manufacturers of helicopter airframes conducted finite element modeling, analysis, measurement, and correlation studies on real aircraft. The second phase is primarily an in-house effort using the airframe finite element models developed by the contractors as the basis for the development of advanced design analysis techniques in such areas as coupled rotor-airframe vibrations and airframe structural optimization under vibration constraints. Figure 74b illustrates the technical areas which are currently receiving attention under the program. As suggested by the figure, future emphasis will be on establishing the technology needed to support the development of advanced high-speed rotorcraft such as tiltrotors.

Status/Plans: Several areas are mentioned as examples of the work planned for FY 1990. A GVT of the Sikorsky S76 ACAP will be conducted at the Army Aviation Applied Technology Directorate, Fort Eustis. This test would be a continuation of the difficult components studies which are intended to identify components of an airframe that contribute to the lack of correlation between test and analysis at higher frequencies. In-house static and dynamic testing of composite model rotor blades employing extension-twist coupling will be performed. The development of improved finite element modeling techniques as well as improved mathematical representations of finite elements will be continued. Development of analysis methods which more realistically account for damping and which are suitable for use in airframe design work will also continue.

Figure 74 (a).

ROTORCRAFT STRUCTURAL DYNAMICS

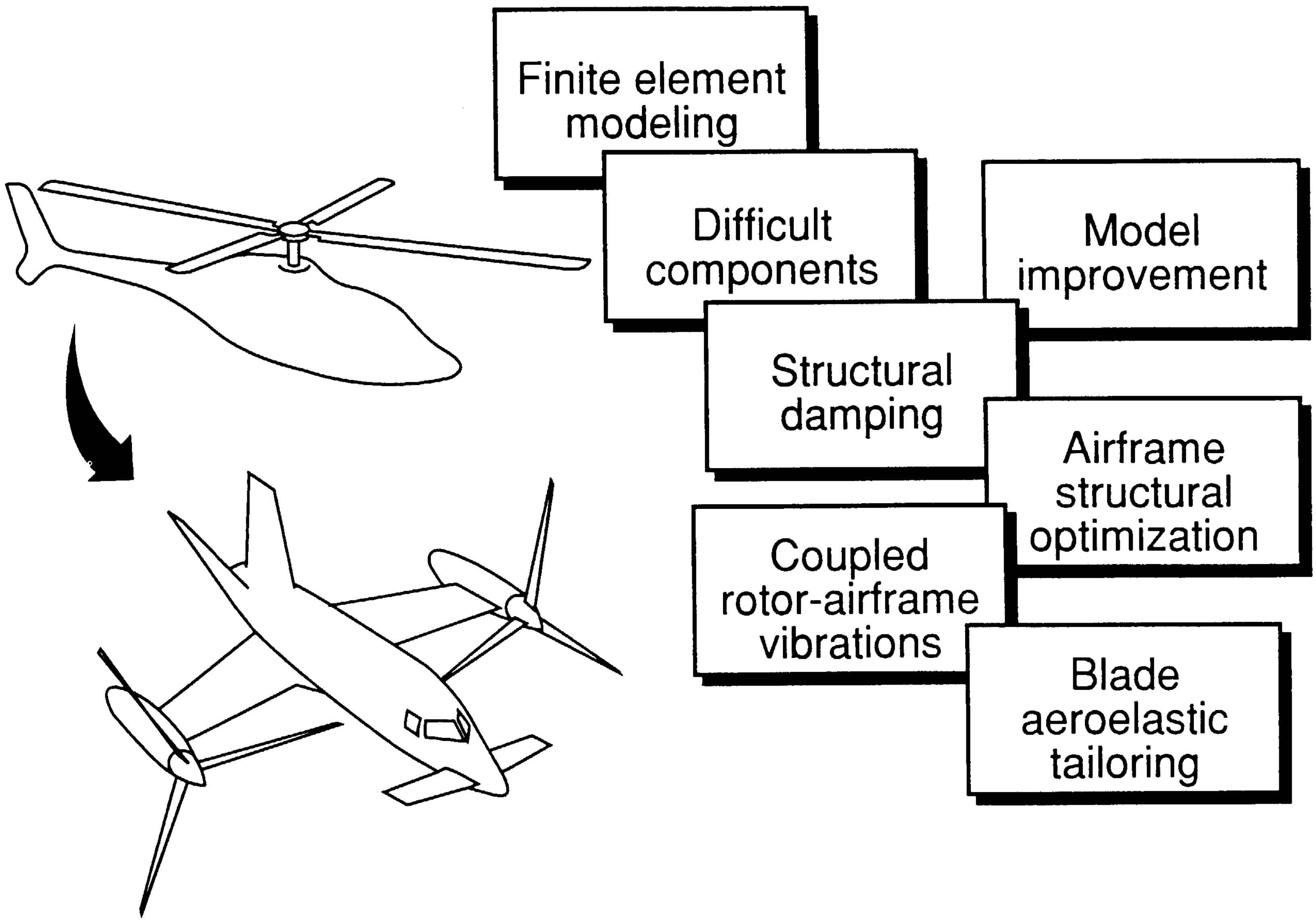

Figure 74 (b).

UNSTEADY AERODYNAMICS

FY 1990 PLANS

- CAP-TSD code application and support
  - Continue in-house and cooperative applications to verify code's range of accuracy
  - Continue to provide programming support
- Develop Euler and Navier-Stokes capabilities for steady/unsteady aerodynamic analysis
  - Grid generation methodology
  - Structured grid flow solvers
  - Unstructured grid flow solvers
  - Turbulence modeling
- Development of aeroelastic analysis methods for vortex dominated and buffeting flows
  - Coupled Navier-Stokes/structural dynamics procedure
  - Initial code validation studies

Figure 75.

AEROSERVOELASTICITY

FY 1990 PLANS

- Extend matched filter theory for nonlinear systems applications
- Design/test RMLA/FSS system on AFW in the simulator and in TDT
- Integrate nonlinear/time-domain aero into ASE analysis/ design methods
- Develop aeroelastic methods for hot hypersonic vehicles
- Apply multilevel decomposition methods to HSCT and CSI
- Exploit smart structures technology for aeroelastic application

Figure 76.

MAXIMIZED GUST LOAD COMPUTATIONS FOR NONLINEAR SYSTEMS

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Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective: Modern transport aircraft employ active control systems which augment stability, improve flying characteristics, and reduce gust and maneuver loads. These active control systems almost always contain significant hardware nonlinearities such as actuator rate and deflection limits and deadbands, and software nonlinearities. Many of the established methods used to calculate aircraft gust loads are applicable to linear systems only. Some of the methods developed in the past few years are applicable to the nonlinear aircraft, but have the disadvantage of being computationally time consuming to use. The objective of this work is to extend the Matched Filter Theory Method (MFT) for obtaining maximized gust loads to nonlinear systems.

Approach: The MFT Method offers a means of computing maximized and time-correlated gust loads for linear systems. The proposed extension of the method (see figure 77b) offers a means of computing the gust loads for nonlinear systems. This extension uses constrained optimization and its major features are illustrated in the flow diagram in the accompanying figure. The original method is used to provide an initial estimate of a unit-energy excitation waveform. This waveform is approximated by a high-order polynomial whose coefficients must first be computed. The approximated waveform is then passed through a "pre-filter," the output of which is a gust velocity profile as a function of time. This gust profile is then applied to a nonlinear representation of an aircraft to obtain the resulting output load time response. The optimization code varies the coefficients of the approximation of the unit-energy excitation waveform and terminates when the unit-energy excitation is found which produces maximum loads.

Status/Plans: The extended method has been applied to the nonlinear aeroservoelastic model of a drone aircraft shown on the right side of the figure. Execution was very slow and the task had to be terminated prior to a converged solution. In an attempt to reduce computation time, the plan is to modify codes to take advantage of modern numerical techniques which permit rapid convergence to the solution. In addition, preliminary results will be presented at a Work-In-Progress Session at the 1990 AIAA Structures, Structural Dynamics, and Materials Conference and at the 1990 Gust Specialist Meeting, both to be held in April in Long Beach, CA.

Figure 77 (a).

MAXIMIZED GUST LOAD COMPUTATIONS FOR NONLINEAR SYSTEMS

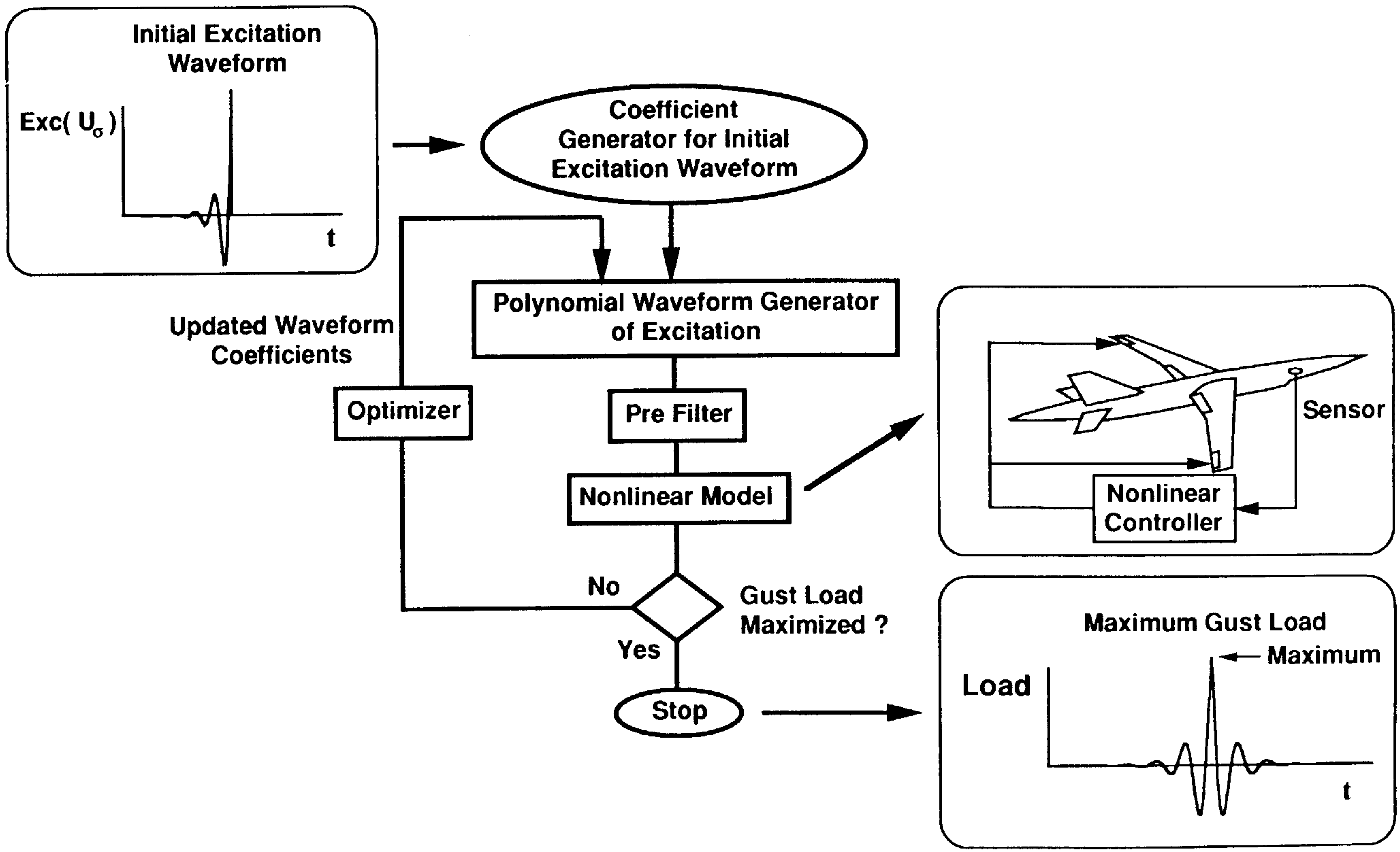


Figure 77 (b).

MODELLING OF NONLINEAR AERODYNAMICS USING THE VOLTERRA-WIENER THEORY

Walter A. Silva (Lockheed)
Aeroservoelasticity Branch

RTOP 505-63-21

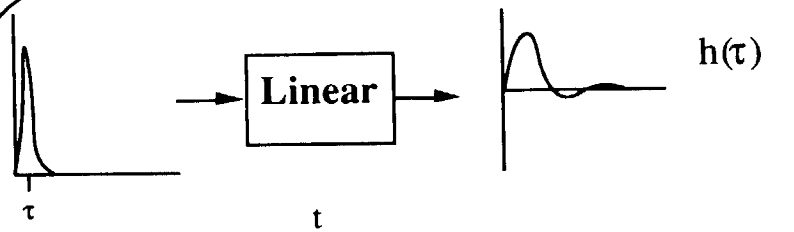
Research Objective: Modern aeroservoelastic analysis methods model linear aerodynamic response as an approximate linear system (rational function approximation) that augments the structural portion of the equations of motion. There is, however, a very real need to perform aeroservoelastic analyses in the transonic regime where dynamic instabilities can be very severe and sudden. Transonic aerodynamics are nonlinear and cannot accurately be approximated using linear methods. The objective of this research is to develop a methodology to model nonlinear aerodynamic response using the Volterra-Wiener theory of nonlinear systems for subsequent use in aeroservoelastic analysis.

Approach: The basic premise of the Volterra-Wiener theory of nonlinear systems is that any nonlinear system can be modelled by an infinite sum of subsystems of increasing order. These subsystems contain a kernel that describes the behavior of the system at that order (see equation in fig. 78b). The first-order kernel is simply the linear unit impulse response of the system which assumes that the response of the system at a given time t is independent of the response of the system at a previous time t-T. This is why superposition of responses is valid for linear systems. The second-order kernel is the response of the system to two unit impulse responses at different time intervals. This is a measure of the relative influence of a previous input on the current response, which is a measure of nonlinearity. The current research will be limited to a second-order formulation. The second-order kernel is computed using the pulse response option within the CAP-TSD (Computational Aeroelasticity Program-Transonic Small Disturbance) code for two pulses at varying time intervals. Once the kernel has been computed, a bilinear state-space realization of the system can be analytically obtained, analogous to the linear realization obtainable from the unit impulse response of a linear system.

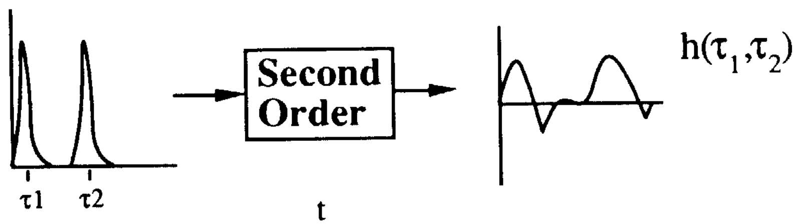
Status/Plans: Since bilinear state-space equations are not addressed in modern linear systems software, a simple finite-difference model of the equations has been generated for the purpose of investigating the Volterra-Wiener theory. This finite-difference formulation has been verified for linear systems. The analysis is currently focused on the application of Volterra-Wiener theory to simple, well-defined systems for the purpose of comparing analytical results to numerical results so as to understand issues of accuracy and application. The next step is to apply the methodology to a simple configuration using CAP-TSD.

Figure 78 (a).

Modelling of Nonlinear Aerodynamics Using Volterra-Wiener Theory



$$y(t)_{linear} = \int_0^t h(\tau)\, u(\tau)\, d\tau$$

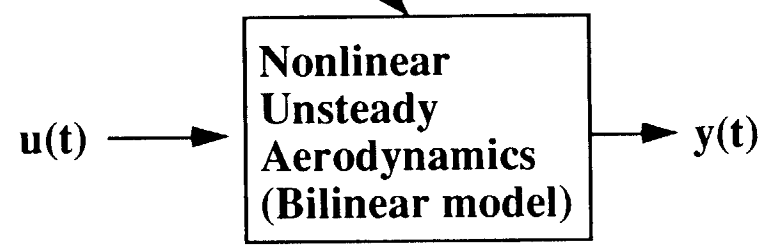


$$y(t) = y(t)_{linear} + \int_0^t \int h(\tau_1,\tau_2) u(\tau_1) u(\tau_2) d\tau_1 d\tau_2$$

$$h(\tau_1,\ldots,\tau_n) = C[\exp(A\tau_n)]D \ldots D[\exp(A\tau_1)]B$$

$$\dot{x} = Ax + Dxu + Bu$$

$$y = Cx$$



u - modal coefficient

y - aerodynamic force

- Kernels evaluated numerically by multidimensional impulse responses
- Truncated series converted to bilinear state-space realization
- Directly usable in existing ASE evaluation methods

Figure 78 (b).

INVESTIGATE TIME-DOMAIN AERODYNAMIC APPROXIMATION METHODS FOR AEROSERVOELASTIC ANALYSIS

Jessica A. Woods and Michael G. Gilbert
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective: The objective is to develop the methodology necessary to utilize subsonic and supersonic unsteady time-domain aerodynamics directly in aeroservoelastic analyses. The intent is to evaluate the stability of flexible flight vehicles without transforming the aerodynamic forces into the frequency domain and without the time consuming computation of the aeroelastic system's forced response. Further, a model incorporating time-dependent aerodynamics may be used in control system design to determined stability of aeroservoelastic systems undergoing arbitrary motion.

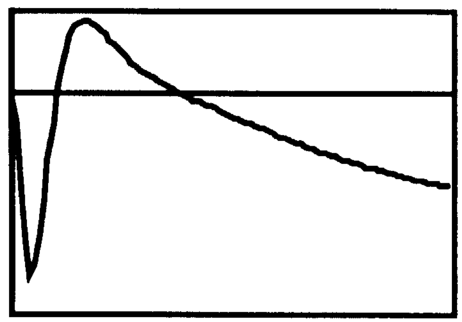
Approach: The approach is to utilize the Laplace transform of a commonly used frequency domain approximation in state-space form to model the aerodynamic forces in the time domain (see figure 79b). System identification techniques frequently used in modern control systems analysis are applied to obtain an approximation of the generalized aerodynamic forces. Essentially, the linear method involves a constrained least squares approximation to the system impulse and step responses to determine the system matrices **[B]** and **[D]**. A parameter identification technique using a Taylor series approximation is employed to update aerodynamic poles for an improved approximation. The converged aerodynamic model is then coupled with a commonly used state-space representation of the vehicle's structural dynamics. Aeroelastic stability is evaluated by considering the eigenvalues of the resulting system matrix.

Status/Plans: A viable state-space model has been developed which models initial aerodynamic forces. Improvements to the aerodynamic approximation are underway using parameter identification techniques. Case studies will be conducted to evaluate the potential of using time-domain aerodynamic forces for aeroelastic and aeroservoelastic applications. The research will be presented at a Work-in-Progress session at the AIAA Structures, Structural Dynamics, and Materials Conference in April 1990. A full paper will be presented later in the year.

Figure 79 (a).

INVESTIGATE TIME-DOMAIN AERODYNAMIC APPROXIMATION METHODS FOR AEROSERVOELASTIC ANALYSIS

Aerodynamic Force



State-Space Equations

$$\dot{x} = Ax + Bu$$

$$\overline{Q} = Cx + Du$$

Methodology

- Incorporate time dependent aerodynamic forces into existing ASE methodology by representing aerodynamic forces in state-space form.
- Develop a constrained time domain least squares solution for B and D matricies.
- Improve approximation of the system dynamics using parameter identification techniques.

Figure 79 (b).

EVALUATE SMART STRUCTURES TECHNOLOGY FOR APPLICATION TO AEROELASTICITY

H. J. Dunn and Robert C. Scott (Purdue University)
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective: The objectives of this effort include: (1) evaluate the potential of using piezoelectric materials or shape memory alloys for aeroelastic and aeroservoelastic applications; (2) develop the analytical techniques necessary to examine these materials and to determine the best design methodology; and (3) use the analytical results as a basis for designing models and performing wind-tunnel experiments to further investigate the feasibility of "smart structures" technology.

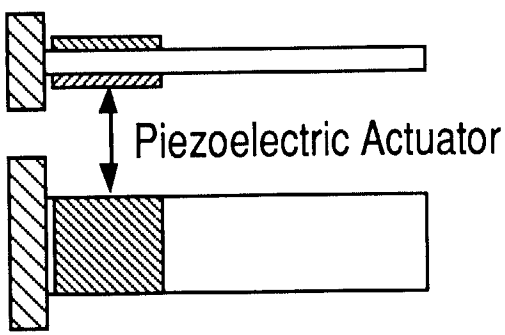
Approach: As shown in figure 80b, this is an analytical investigation to define the use of "smart" materials for aeroelastic applications has been initiated. Two piezoelectric materials, lead zirconate titinate (PZT) and polyvinylidene flouride (PVDF), and the shape memory alloy (SMA) NITINOL have been considered in these studies. Aeroelastic structures consisting of simple beams, thin panels, and cantilever plates have been analyzed using an assumed mode method utilizing strip theory for subsonic aerodynamics and piston theory for the supersonic aerodynamics. Aeroelastic problems considered include wing flutter, panel flutter, and structural response to random excitation. Thus far, the alleviation of panel flutter has been shown to be the most promise application for "smart" structures. SMAs can significantly increase the flutter speed of panels while decreasing the structural weight. PZT has also been shown to have promise for flutter suppression of low-aspect ratio wings where camper modes are important. In addition, NASTRAN plate models have indicated that a washout wing could change to a washin wing by stimulating the imbedded SMA fibers; thus active aeroelastic tailoring is possible. Finite element models are now being developed to incorporate the piezoelectric material PZT and dynamic simulation techniques are being considered for analyzing the structural equations with active control systems.

Status/Plans: Determination of candidate configurations for wind-tunnel validation experiments should be completed by early CY 1990. Simple plate and beam models with added piezoelectric materials are being considered for the wind-tunnel model. Only linear analysis techniques have been used in current work; nonlinear models are being considered for future analytical investigations.

Figure 80 (a).

EVALUATE SMART STRUCTURES TECHNOLOGY FOR APPLICATION TO AEROELASTICITY

- Evaluate feasibility of piezoelectric materials and shape memory alloys for controlling aeroelastic response
- Define and perform low-cost laboratory, ground vibration, and wind tunnel validation experiments
- Develop differential stiffness modelling approach for nonlinear simulation with active control systems



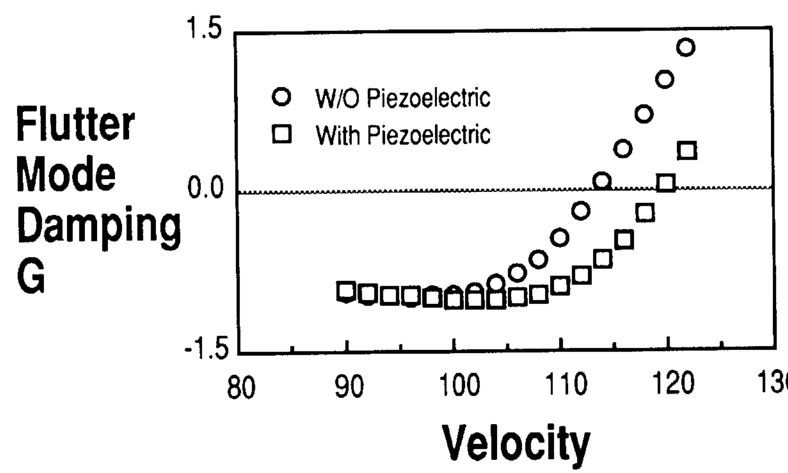


Figure 80 (b).

LaRC HIGH-SPEED AIRFRAME INTEGRATION RESEARCH

Michael G. Gilbert, Jessica A. Woods, Carol D. Wieseman, Thomas E. Noll
Aeroservoelasticity Branch
and
Maynard Sandford
Configuration Aeroelasticity Branch

RTOP 505-63-21

Research Objective: The objective of participation in the High-Speed Airframe Integration Research (HiSAIR) program is to incorporate aeroelastic and aeroservoelasticity technology at the preliminary design level, to assist in developing integrated multidisciplinary design methodology, and to validate new code developments and control concepts through wind-tunnel test demonstrations.

Approach: As shown in figure 81b, the SDyD team will provide dynamic aeroelastic and aeroservoelastic stability evaluations of HiSAIR concepts in the subsonic, transonic, and supersonic speed regimes. Aerodynamic sensitivity information will be derived for selective state-of-the-art unsteady subsonic and supersonic codes. Initially this data will be calculated using finite differencing techniques; later this data will be derived based solely on analytical expressions, if practical. Team members will implement and apply sensitivity derivative methodology involving structural dynamics, optimal control, and aerodynamics to support the HiSAIR working groups needs. Active control systems for improving or exploiting the aeroelastic response characteristics of the vehicle will be developed and offered for integration into the vehicle design process. Actively-controlled aeroelastic wind-tunnel models will be designed, fabricated, and tested to validate code enhancements and to verify the feasibility of new active and passive concepts for alleviating undesirable aeroelastic response or for using aeroelasticity to improve the aircraft's performance characteristics.

Status/Plans: The team has assisted in organizing the exchange of data between working groups within the Structures Directorate and represents the Directorate as a focal point for data exchange with other directorates. Thus, an organized communication network is being established with researchers in other disciplines. The team will develop an understanding of how changes in other technical disciplines effect the aeroelastic and aeroservoelastic characteristics of a high-speed vehicle. Likewise, other disciplines will recognize the potential for aeroelasticity and aeroservoelasticity to improve aircraft stability and control and of its importance at the preliminary design stage. In addition, the HiSAIR multidisciplinary activities will stimulate the development of aeroelastic analysis capabilities in the supersonic regime.

Figure 81 (a).

LaRC HIGH-SPEED AIRFRAME INTEGRATION RESEARCH

Multidisciplinary Skills and Communication

- Recognize that ASE is important during preliminary design
- Demonstrate ability to effectively contribute during multidisciplinary research
- Understand how changes in other technical disciplines affect ASE

Contributions

- Develop communication and data exchange with other LaRC working groups
- Provide dynamic ASE evaluation of HiSAIR concepts throughout flight envelope
- Conduct wind-tunnel demonstration for verification of ASE concepts

Benefits

- Stimulate development of supersonic analysis capabilities
- Gain experience with multidisciplinary data bases

Figure 81 (b).

DEVELOP UNSTEADY AERODYNAMIC FORCE SENSITIVITY ANALYSIS CAPABILITY

Carol D. Wieseman
Aeroservoelasticity Branch

RTOP 505-63-21

Research Objective: The objective of this activity is to develop analytical or semi-analytical tools for determining sensitivity expressions of steady and unsteady aerodynamic forces with respect to planform, control surfaces, or other important physical parameters of interest. The goal of the effort is to eventually incorporate the aerodynamics discipline into an integrated multilevel optimization design procedure based on hierarchal decomposition.

Approach: As shown in figure 82b, an integrated design approach currently under development is based on hierarchal problem decomposition, formal optimization techniques, and information that describes the sensitivity of the independent technical disciplines to a set of multidisciplinary integration parameters. The design problem is decomposed into multiple levels. The integrated design requirements are formulated in the top level of the process and are based on the set of integration parameters; at the lower level, the independent technical discipline designs and the sensitivity of those designs to the integration parameters are obtained. The lower level design sensitivity information is used as gradient information at the top level in the formal design optimization process. Presently, structural and control system sensitivity information can be obtained. However, aerodynamic sensitivity data has not been incorporated into the integrated design process. To begin this task, a review of the pertinent technical literature will be conducted to establish the state-of-the-art and to provide guidance. The investigation will focus on the subsonic and supersonic unsteady codes commonly used for aeroservoelasticity evaluations. Initially, finite differencing techniques will be used to obtain the sensitivity data. This will allow early incorporation of aerodynamic sensitivity data to continue the development of the integrated design approach. As experience and data becomes available analytical sensitivity derivatives will derived where practical.

Status/Plans: The paper entitled "Aerodynamic Sensitivities from Subsonic, Sonic, and Supersonic Unsteady, Nonplanar Lifting Surface Theory" by Dr. Yates has been obtained and is being reviewed. In terms of initial subsonic doublet lattice analyses, the sensitivity of unsteady rigid body aerodynamics with aspect ratio was calculated for a rectangular supercritical wing using the finite differencing approach. Future plans involve deriving analytical sensitivity expressions if feasible for the subsonic doublet lattice method and for the supersonic kernal function and piston theory methods. If analytical sensitivities can not be derived, it will be necessary to automate the change of the input to the computer programs to reflect the changes in the geometry for which sensitivity information is desired in order to calculate sensitivities by finite differencing.

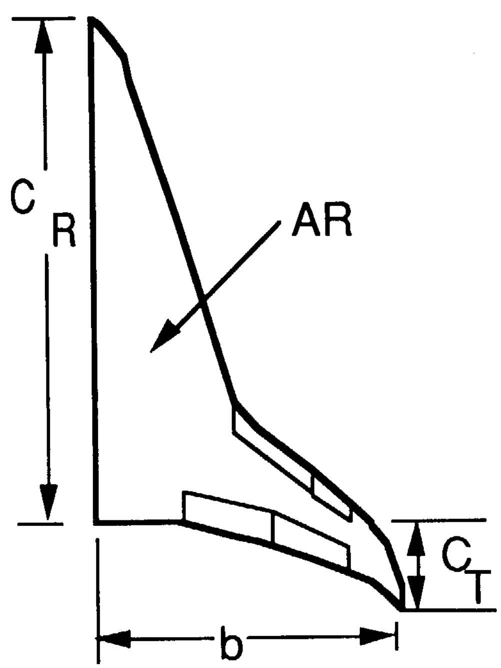
Figure 82 (a).

DEVELOP UNSTEADY AERODYNAMIC FORCE SENSITIVITY ANALYSIS CAPABILITY

Objective: Develop Analytical Tools to Determine Sensitivities of Steady and Unsteady Aerodynamic Forces with Respect to Planform, Control Surfaces, and Other Important Physical Parameters

Approach: Use Existing Aerodynamics Tools:

- Doublet Lattice Subsonic Lifting Surface Theory
- Avaliable Supersonic Lifting Surface Theories
- Piston Theory



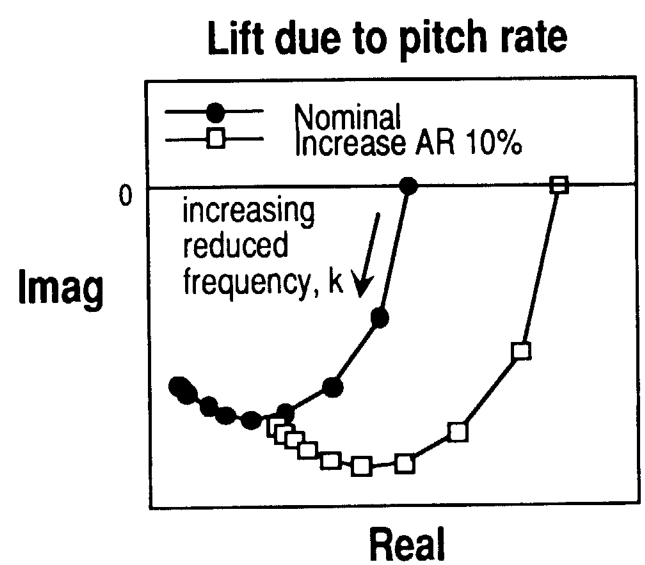


Figure 82 (b).

MULTIFUNCTION DIGITAL ACTIVE CONTROL LAW DESIGN AND VALIDATION

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RTOP 505-63-21

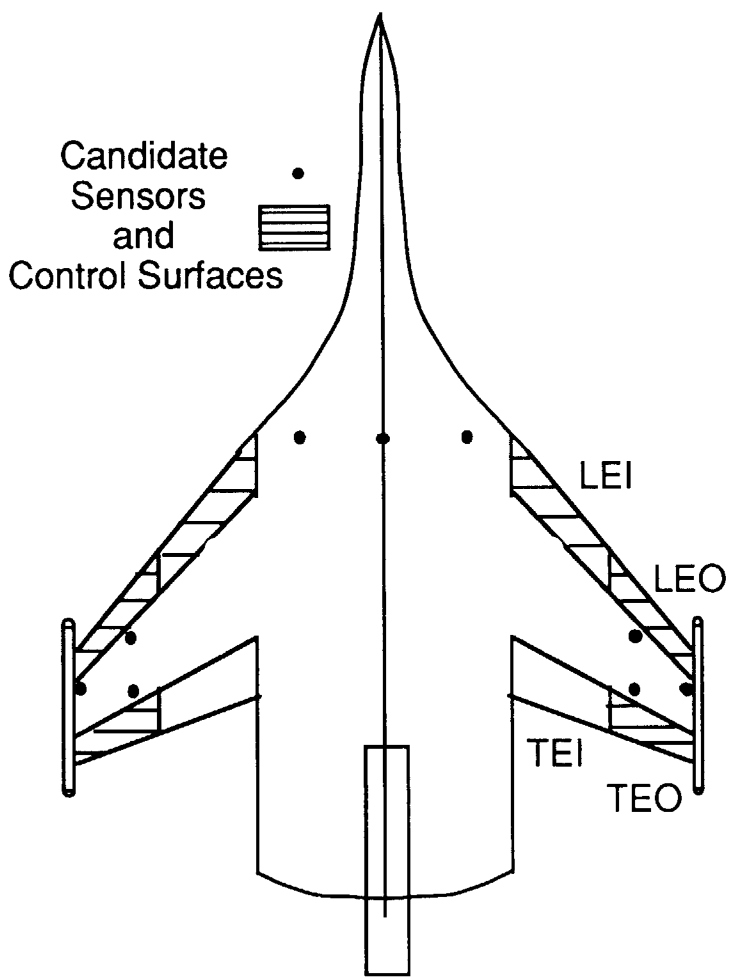
Research Objective: The objective of this activity is to develop a control law which can simultaneously suppress the symmetric and antisymmetric flutter modes associated with the Active Flexible Wing (AFW) wind-tunnel model while undergoing rolling maneuver load alleviation (RMLA) functions.

Approach: As shown in figure 83b, the state-space model of the AFW wind-tunnel model will be estimated from recently acquired experimental data, compared with theoretical predictions, and modified as appropriate prior to design the multifunction control system. The antisymmetric flutter suppression system (FSS) and the RMLA control laws will be designed simultaneously, and then combined with the symmetric FSS. Both classical and a LQG technique will be used to design the FSS and RMLA control laws. Constrained optimization procedures will be applied to improve the robustness of the combined system and to reduce the dynamic loads. It is expected that the multifunction control system will use the four trailing-edge and the two leading-edge outboard control surfaces as the active surfaces, and accelerometers, strain gages, and the roll rate gyro as feedback signals.

Status/Plans: The first test of the AFW model has been completed; a significant amount of data has been recorded on analog and digital tapes for post-test reduction. Software to process the experimental data and to construct an updated state-space model will be developed. The state-space model will be improved using parameter estimation techniques, and updated based on measured control surface effectiveness data and the effects of loaded actuator dynamics prior to the design of the multifunction control system.

Figure 83 (a).

MULTIFUNCTION DIGITAL ACTIVE CONTROL LAW DESIGN AND VALIDATION



Development and Simulation of Flutter Suppression and Rolling Maneuver Load Alleviation (FSS-RMLA) Control Laws

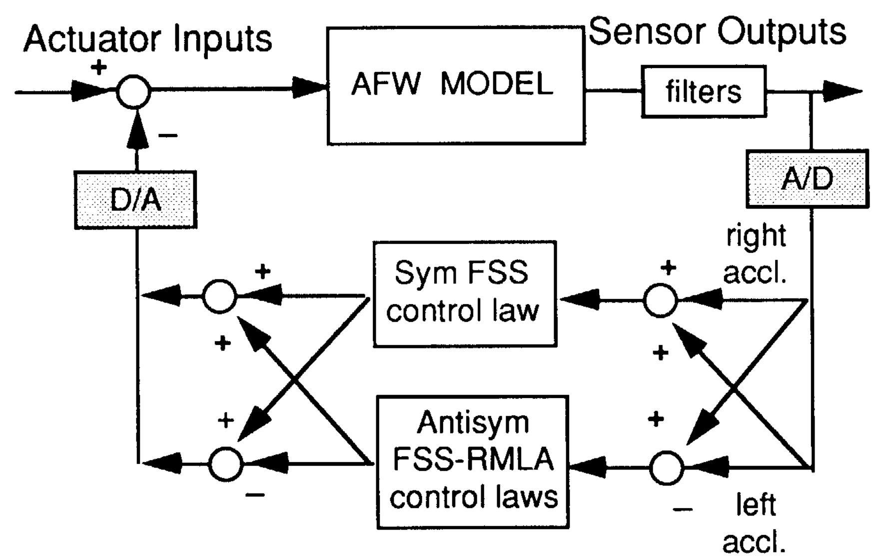


Figure 83 (b).

APPLY INTEGRATED MULTIDISCIPLINARY DESIGN METHODS TO CSI PROBLEMS

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RTOP 505-63-21

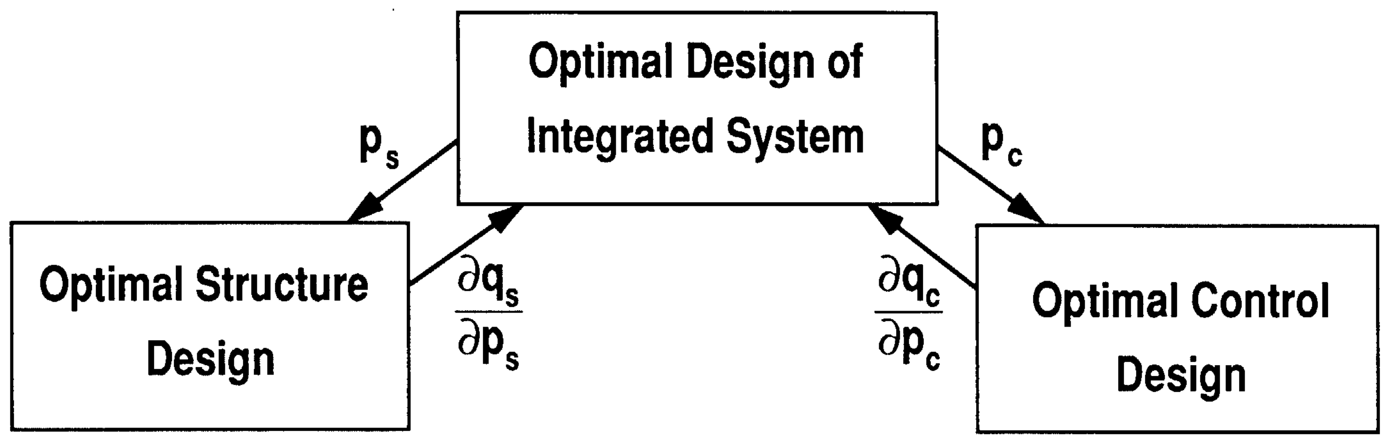
Research Objective: Existing aerospace vehicle design methodologies do not take maximum advantage of multidisciplinary interactions. A design methodology that integrates structural design and active control design has been successfully tested on a space structure design problem that is relatively small. The smaller problem served as an appropriate test bed for development of the new method. However, the real benefit of the design method is that it decomposes a large design problem into a number of smaller, more easily managed design problems. The present research objective is to apply the method to a large space structure configuration being used in Controls/Structure Interaction (CSI) studies.

Approach: The integrated design methodology is based on hierarchical multilevel problem decomposition and optimization techniques. The actively controlled structure is decomposed into its structural and control subsystems as shown in figure 84b. At the topmost, integrated level, a system measure is optimized with structural and control parameters as design variables. The structural and control subsystems are optimized using design variables and methods suitable to their respective disciplines, but with those parameters specified at the top level held constant. Sensitivity derivatives, with respect to the parameters, of these optimum solutions are used in the top level optimization as gradient information.

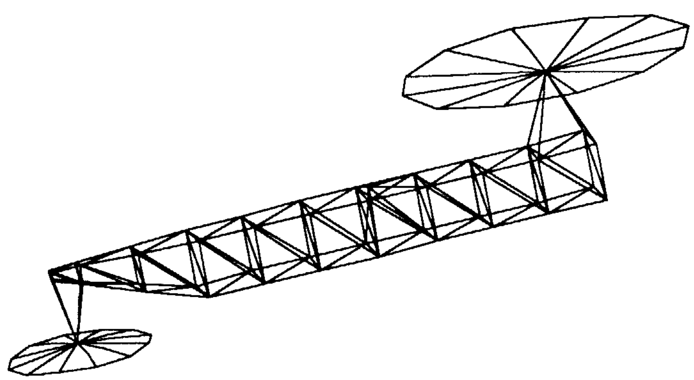
Status/Plans: A code exists for performing analysis and design of the Earth Pointing Satellite (EPS) model shown in the figure. This code uses a single level optimization format in which detail design variables, such as structural member sizes and control gains, are synthesized simultaneously as a group. This requires either that a large number of design variables be determined at the cost of manageability, or that the number of variables be reduced to manageable levels at the cost of design freedom. In the present research, the existing EPS design code will be exercised on various study problems to gain insight into the effects of different combinations of objective and constraint functions, to test methods of dynamic model reduction, and to provide benchmark examples. The code will then be altered or adapted to the multilevel format. It is anticipated that application of the multilevel method to a large system will provide a means to investigate several different decomposition schemes as well as reveal limitations of the method.

Figure 84 (a).

APPLY INTEGRATED MULTIDISCIPLINARY DESIGN METHODS TO CSI PROBLEMS



EARTH POINTING SATELLITE



ISSUES TO BE ADDRESSED:

- Objective and Constraint Functions
- Hierarchical Decomposition Schemes
- Limits of Hierarchical Method
- Dynamic Model Reduction

Figure 84 (b).

LANDING DYNAMICS

FY 1990 PLANS

- Complete first phase of bias-ply, radial-belted, and H-type tire tests on ALDF on smooth ungrooved concrete surface
- Initiate test program on ALDF for F-4 and F-16 radial-belted and bias-ply tires
- Complete drop tests of F-106 modified main and nose gears and prepare all hardware for flight tests at end of calendar year 1990
- Continue heavy rain experiment on ALDF
- Develop 2D shell finite element code for tire analysis with friction contact algorithm

Figure 85.

IMPACT DYNAMICS

FY 1990 PLANS

- Acquire MSC/DYNA for crash analysis efforts on LaRC's Convex computer system
- Initiate task assignment for energy absorbing composite structural components design and fabrication
- Continue enhancement of DYCAST computer code to handle non-linear response of curved composite beams
- Conduct static and dynamic tests of composite frame specimens with I-, J-, and C-cross sections
- Continue development and extension of computationally efficient algorithms for composite structural analyses (GWU Grant)
- Conduct strength scaling studies of composite structures (VPI Grant)

Figure 86.

SPACECRAFT DYNAMICS

FY 1990 PLANS

Control Structures Interaction (CSI)

- Complete CSI evolutionary model to include pneumatic thrusters
- Complete structural tests & initiate controls tests of evolu. model
- Conduct dynamics and controls experiments on Mini-Mast for guest investigators from three universities
- Add on-site test control room to space structures research lab

Structural dynamics analysis and test methods

- Initial testing of hybrid Space Station Freedom (SSF) model
- Determine impact of exploration configurations of SSF on dynamics and control requirements
- Conduct LATDYN workshop for industry/university users
- Experimentally verify system identification algorithm which learns

Figure 87.

Report Documentation Page

National Aeronautics and Space Administration

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16. Abstract

The purpose of this paper is to present the Structural Dynamics Division's research accomplishments for FY 1989 and research plans for FY 1990. The work under each Branch (technical area) is described in terms of highlights of accomplishments during the past year and highlights of plans for the current year as they relate to five year plans for each technical area. This information will be useful in program coordination with other government organizations and industry in areas of mutual interest.

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