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LOADS AND AEROELASTICITY DIVISION RESEARCH  
AND TECHNOLOGY ACCOMPLISHMENTS FOR FY 1984  
AND PLANS FOR FY 1985

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LOADS AND AEROELASTICITY DIVISION  
RESEARCH AND TECHNOLOGY ACCOMPLISHMENTS FOR FY 1984  
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SUMMARY

The purpose of this paper is to present the Loads and Aeroelasticity Division's research accomplishments for FY 84 and research plans for FY 85. The work under each branch (technical area) will be 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 and industry in areas of mutual interest.

ORGANIZATION

The Langley Research Center is organized by directorates as shown on figure 1. The top three perform support functions and the bottom four conduct the research program. A directorate is organized into divisions and offices as illustrated on the figure for the Structures Directorate. The Structures Directorate was partially reorganized in June, 1984. The effect on The Loads and Aeroelasticity Division (LAD) follows:

1. The Multidisciplinary Analysis and Optimization Branch (MAOB) was abolished. Part of this branch is now the Interdisciplinary Research Office (IRO) reporting to the Director for Structures. The former Langley Chief Scientist heads the office and the former head of MAOB is his deputy. Most of the rest of MAOB is now designated the Aeroservoelasticity Branch (AB).
2. Part of the Structural Concepts Branch, Structures and Dynamics Division, Applied Materials Branch, Materials Division, MAOB, LAD, and the Aerothermal Loads Branch, LAD, were grouped into a new branch, the Thermal Structures Branch, LAD.

The Loads and Aeroelasticity Division consists of five branches as shown on figure 2. This figure lists the key people in the division which consists of 68 NASA civil servants and seven members of the Army Structures Laboratory (Army Aviation Systems Command) located at Langley Research Center. Each branch represents a technical area and disciplines under the technical areas are shown on the figure. All of the Army personnel work on the Rotorcraft Aeroelasticity and Rotorcraft Structural Dynamics disciplines.

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 (AB) all work in the area of Control of Aeroelastic Stability and Response. The Aerothermal Loads Branch (ALB) and the Thermal Structures Branch (TSB) work the areas of Loads and Structures for High Speed Vehicles (Aeronautics) and Loads and Structures for Advanced Space Transportation Systems (Space).

## RESEARCH PHILOSOPHY

The basic philosophy and motivation of the Loads and Aeroelasticity Division research program can be captured in some quotes from some leaders in Aerospace Research and Development. In his 13th Von Karman lecture on Aeroelasticity (ref. 1), I. E. Garrick related the following: "Von Karman's sense of humor, which was remarkably appropriate to a given occasion, has become legendary. Recognizing that the poor structures engineer was usually held accountable for structural integrity, he quipped, 'The aerodynamicist assumes everything but the responsibility.'"

"It has been gratifying to me to observe that in major aerospace industry the aeroelastician is no longer the stepchild he once was. From an almost parochial isolated specialist, he is now the generalist who tends to pull together the separate efforts in structures, aerodynamics, stability and control, and propulsion, even in early design stages. Yet, there are still human problems such as one-way communications and barriers between departments as well as physical problems that are often so recondite and difficult that aeroelastic problems may slip through the cracks."

In his Wright Brothers Lectureship in Aeronautics on Optimization (ref. 2), Holt Ashley observed: "Further mention will be made in what follows of the keen disappointment felt by many specialists because their theories have received so little practical application. This phenomenon is frequently attributed to a reluctance by developmental engineers to adopt unfamiliar and untried methods of analysis."

In an appraisal of the study of Hypersonic Airframe Structures, (ref. 3), Rene Miller stated: "The cost effectiveness of .... (Thermal) Structural Concepts is greatly dependent on solutions to the detailed design problems. In fact, it is likely that these detailed design problems as demonstrated in the X-15 program will prove to be the pacing item in the development of Hypersonic Aircraft."

The Loads and Aeroelasticity Division program is aimed at producing the data and analysis methods required by those who are accountable for the structural integrity of aerospace vehicles; to provide the detailed design data and methods for the pacing item of development of hypersonic vehicles - cost effective thermal structures; to continue to pull together those separate efforts that ought to (or must) be considered as a single task; to preclude aeroelastic problems from slipping through the cracks; and to alleviate the reluctance by developmental engineers to adopt unfamiliar and untried methods by making them both familiar and proven.

## FACILITIES

The Loads and Aeroelasticity Division has two major facilities available to support its research as shown in figure 4.

The Transonic Dynamics Tunnel (TDT) is a Mach 0.2 to 1.2 continuous flow, variable-pressure wind tunnel with a 16 ft. square test section which uses a freon-12 test medium primarily for dynamic aeroelastic testing. This unique facility is used primarily by the Configuration Aeroelasticity Branch.

Semi-span, side-wall mounted models and full-span cable-mounted models are used for aeroelastic studies of fixed wing aircraft. The ARES (Aeroelastic Rotor Experimental System) test stand is used in the tunnel to study the aeroelastic effects on rotors. A Hover Facility, located nearby, is used to setup the ARES test stand in preparation for entry into the TDT. A modernization of the TDT Data Acquisition System is underway along with a major Coff activity for density increase. After these modifications, currently in progress, are completed in early 1985, the tunnel will operate at dynamic pressures up to 600 psf and Reynolds numbers up to  $8 \times 10^6/\text{ft}$ . Replacement cost for this facility is \$63M.

The Aerothermal Loads Complex consists of six facilities which are operated by the Aerothermal Loads Branch to carry out their research. The 8-Foot High Temperature Tunnel (8' HTT) is a unique hypersonic Mach 7 blowdown wind tunnel with an 8' diameter test section (uniform temperature test core of 4') that uses products of combustion (methane and air under pressure) as the test medium. The tunnel operates at dynamic pressures of 250 to 1800 psf, temperatures of 2400 to 3600°R and Reynolds numbers of 0.3 to  $2.2 \times 10^6/\text{ft}$ . The tunnel is used to test flat and curved surface type models to determine aerothermal effects and to evaluate new high temperature structural concepts. A major Coff item is underway to provide alternate Mach number capability and provide O<sub>2</sub> enrichment for the test medium. This is being done primarily to allow the tunnel to test models that have hypersonic air breathing propulsion applications. Replacement cost for the tunnel is \$45M.

The 7-Inch High Temperature Tunnel (7" HTT) is a 1/12 scale of the 8' HTT with basically the same capabilities as the larger tunnel. It is used primarily as an aid in the design of larger models for the 8' HTT and for aerothermal loads test on subscale models. The 7" HTT is currently being used to evaluate various new systems for the 8' HTT. Replacement cost for the tunnel is \$0.8M.

The 1 x 3 High Enthalpy Aerothermal Tunnel (1 x 3 HEAT) is a unique facility designed to provide realistic environments and times for testing thermal protection systems proposed for use on high-speed vehicles such as the Space Shuttle. The facility is a hypersonic blowdown wind tunnel that uses products of combustion as the test medium. Test panels mounted on the sidewalls can be as large as 2' wide x 3' long. The facility operates at dynamic pressures of 1 to 10 psi, Mach numbers from 4.7 to 3.5 depending on the temperatures, which range from ambient to 5800°F, an altitude range simulating flight of 130,000 to 80,000 ft., and enthalpy levels from 1100 to 4400 BTU/lb depending on the oxygen levels used in the test medium. Replacement cost for the tunnel is \$8M.

The three Aerothermal Arc Tunnels (20 MW, 5 MW and 1 MW) are used to test models in an environment that simulates the flight reentry envelope for high speed vehicles such as the Space Shuttle. The amount of usable energy to the test medium in these facilities is 9 MW, 2 MW, and 1/2 MW. The 5 MW is a three phase AC arc heater while the 20 MW and 1 MW are DC arc heaters. Test conditions such as temperature, flow rate, and enthalpy vary greatly since a variety of nozzles and throats are available and since model sizes are different (3" diameter to 1' x 2' panels). Replacement cost for these arc tunnels are \$24M.

## FY 84 ACCOMPLISHMENTS

### Configuration Aeroelasticity Branch

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

The Configuration Aeroelasticity FY 84 accomplishments listed below are highlighted by figures 7 through 17.

#### Aircraft Aeroelasticity:

- Spanwise Curvature Raises Flutter Dynamic Pressure
- Flutter Testing Techniques Developed for Use in Cryogenic Wind Tunnels
- X-Wing Aeroelastic Divergence Analytical Methods Validated by TDT Test Results
- Flight Test Shows Laminar Flow Control Research Aircraft Free of Flutter
- Flutter of Four-Engine Transport Wing With Winglets Predictable by Analysis
- Effects of 600 Gallon Fuel Tanks on F-16 Flutter Characteristics Studied in TDT
- Effects of New Multi-purpose Pylons on F-16 Flutter Characteristics Studied in TDT
- Decoupler Pylon Program

#### Rotorcraft Aeroelasticity:

- Higher Harmonic Control Technology Transfer and Flight Demonstration
- Initial TDT Test Provides Essential Data Base for JVX Preliminary Design Development
- Langley Analysis Predicts Measured Stability Characteristics of JVX Model

Each highlight is accompanied by descriptive material.

### Unsteady Aerodynamics Branch

The Unsteady Aerodynamics Branch conducts research (figure 18) to produce, apply, and validate through experiments a set of analytical methods for predicting steady and unsteady aerodynamic loads and aeroelastic characteristics of flight vehicles--with continued emphasis on the transonic range and emerging emphasis on high angle-of-attack maneuvering subsonic and supersonic conditions. This work is more clearly identified in figure 19 which shows the five year plan of the two disciplines and their expected results.

The Unsteady Aerodynamics FY 84 accomplishments listed below are highlighted by figures 20 through 29.

Theory Development:

- Generalized Unsteady Airload Capability Allows Assessment of PADE Approximations for Nonharmonic Motions
- Efficient Accurate Viscous Boundary Layer Model Coupled with XTRAN2L Code
- Inviscid Transonic Code Predicts Dynamic Airfoil Lift for Transient Ramping Motion
- XTRAN2L Extended to Multiple Surface Configuration
- Airfoil Shape, Thickness, Camber, and Angle-of-Attack Effects on Transonic Unsteady Airloads
- Transonic Code Used for Airfoil Active Flutter Control
- Non-Isentropic Unsteady Transonic Small Disturbance Theory
- Nonreflecting Far-Field Boundary Conditions Improve Unsteady Airload Calculations
- Lifting Surface Theory for a Helicopter Rotor in Forward Flight

Unsteady Pressure Measurements:

- Reynolds Number Effects on Unsteady Pressure Studied in 0.3M Cryo Tunnel

Each highlight is accompanied by descriptive material.

Aeroservoelasticity Branch

The Aeroservoelasticity Branch conducts research (figure 30) to develop methodologies for the analysis and synthesis of multifunctional active control systems and conceives, recommends, and provides technical support for experiments to validate the methodologies, including the development of drone flight techniques for validation of high risk aeroelastic control concepts such as flutter suppression; and generates mathematical models needed to support NASA projects and uses them to verify the theoretical developments and their computer implementations. This work is more clearly identified in figure 31 which shows the five year plan of the three areas of concentration and their expected results.

The Aeroservoelasticity FY 84 accomplishments listed below are highlighted by figures 32 through 35.

Analysis Methods:

- A New Formulation of Airplane Dynamic Loads Equations

Control Laws Synthesis Methods:

- A Method To Stabilize Linear Systems Using Eigenvalue Gradient Information

Applications and Validations:

- Flexstab Results Used to Modify Wind Tunnel Rigid Model Data for Prediction of Flexible Airplane Performance
- Wing Surface Pressures Measured During Flight Test Define Shock Location

Each highlight is accompanied by descriptive material.

## Aerothermal Loads Branch

The Aerothermal Loads Branch conducts research (figure 36) to develop and validate solution algorithms, modeling techniques, and integrated finite elements for flow-thermal-structural analysis and design; to identify and understand flow phenomena and flow/surface interaction parameters required to define detailed aerothermal loads for structural design via analysis and test; and to define methods for testing in high enthalpy flow environments including capability for testing of air breathing engines at hypersonic speeds. This work is more clearly identified in figure 37 which shows the five year plan of the three disciplines and their expected results.

The Aerothermal Loads FY 84 accomplishments listed below are highlighted by figures 38 through 43.

### Thermal Loads:

- Aerothermal Test of Shuttle Split-Elevon Model in 8' HTT
- Aerothermal Tests of Spherical Dome Protuberance Models in 8' HTT Completed

### Integrated Analysis:

- Finite Element Multiple Time Domain Algorithm
- Automatic Finite Element Mesh Refinement
- Finite Element Flow-Thermal Analysis

### Facilities Operations and Development:

- Mixer Prototype for the 8' HTT

Each highlight is accompanied by descriptive material.

## Thermal Structures Branch

The Thermal Structures Branch conducts research (figure 44) to develop and validate concepts for aerospace structures whose design is significantly controlled by the thermal excursions of the operating environments of aerospace vehicles. Systems studies in concert with the Space Systems Division or High-Speed Aerodynamics Division help to identify structures and materials technology needs. Structural concepts are then developed, analyzed, fabricated, and tested to verify the required technology advances. This work is more clearly identified in figure 45 which shows the five year plan of the three major disciplines and their expected results. Thermal structures experimental needs are currently in the defining stage. Some static testing of small components is being done for structural concepts being developed and fabricated, with the support of contractors, ADFRF, and the Aerothermal Loads Branch.

The Thermal Structures FY 84 accomplishments listed below are highlighted by figures 46 through 54.

### Structural Systems Studies:

- Structural System Study of Aeroassisted Orbital Transfer Vehicle

### Concept Development:

- Aerothermal and Environment Tests Verify Titanium and Superalloy TPS Concepts

- Aerothermal Tests of Metallic TPS
- Advanced Carbon-Carbon Heat Shield Research
- Method for Reducing Stress Concentration in Diffusion Bonded Joints
- Preliminary Test to Cryogenic and Elevated Temperatures Verifies Potential of New Foam for Reusable Cryogenic Insulation
- Carbon-Carbon Hot Structure Design
- Derivation and Test of Elevated Temperature Thermal-Stress-Free Fastener Concept

Analytical Methods and Applications:

- Hierarchical Integrated Thermal/Structural Analysis

Each highlight is accompanied by descriptive material.

## PUBLICATIONS

The FY 84 accomplishments of the Loads and Aeroelasticity Division resulted in a number of publications. The publications are listed below and are identified by the categories of journal publications, formal NASA reports, conference presentations, contractor reports, and other.

### Journal Publications

1. Kelly, H. N.; and Webb, G. L.: Assessment of Alternate Thermal Protection Systems for the Space Shuttle Orbiter, Progress in Astronautics and Aeronautics, 1983, Vol. 85, Pages 300-324.
2. Ruhlin, C. L.; Watson, J. J.; Ricketts, R. H.; and Doggett, R. V., Jr.: Evaluation of Four Subcritical Response Methods for On-Line Prediction of Flutter Onset in Wind Tunnel Tests. Journal of Aircraft, Volume 20, Number 10, October 1983, Pages 835-840.
3. Edwards, J. W.; Bennett, R. M.; Whitlow, W., Jr.; and Seidel, D. A.: Time-Marching Transonic Flutter Solutions Including Angle-of-Attack Effects. Journal of Aircraft, Volume 20, Number 11, November 1983, Pages 899-906.
4. Eckstrom, C. V.; and Spain, C. V.: Experiences in the Use of Composite Material for a Wing Skin. Journal of Aircraft, Volume 20, Number 11, November 1983, Pages 913-919.
5. Camp, D.; Campbell, W.; Frost, W.; Murrow, H.; and Painter, W.: NASA's B-57B Gust Gradient Program. Journal of Aircraft, Volume 21, No. 3, March 1984, Pages 175-182.
6. Bland, S. R. and Edwards, J. W.: Airfoil Shape and Thickness Effects on Transonic Airloads and Flutter. Journal of Aircraft, Volume 21, Number 3, March 1984, Pages 209-217.
7. Taylor, A. H.; Jackson, L. R.; Cerro, J. A.; and Scotti, S. J.: Analytical Comparison of Two Wing Structures for Mach 5 Cruise Airplanes. Journal of Aircraft, Volume 21, No. 4, April 1984, Pages 272-277.

8. Rowe, W. S. and Cunningham, H. J.: On The Convergence of Unsteady Generalized Aerodynamic Forces. Journal of Aircraft, Volume 21, Number 6, June 1984, Pages 420-427.
9. Shideler, J. L.; Swegle, A. R.; and Fields, R. A.: Honeycomb Sandwich Structure for Future Space Transportation Systems with Integral Cryogenic Tankage. Journal of Spacecraft and Rockets, Volume 21, Number 3, May-June 1984, Pages 246-252.
10. Sawyer, J. W.: Mechanical Properties of the Shuttle Orbiter Thermal Protection System Strain Isolator Pad. Journal of Spacecraft and Rockets, Volume 21, No. 3, May-June 1984, Pages 253-260.
11. Ricketts, R. H.; Sandford, M. C.; Seidel, D. A.; and Watson, J. J.: Transonic Pressure Distributions on a Rectangular Supercritical Wing Oscillating in Pitch. Journal of Aircraft, Volume 21, Number 8, August 1984, Pages 576-582.

#### Formal NASA Reports

12. Bey, K. S.: Analytical Prediction of Aerothermal Environment in a Wing-Elevon Cove. NASA TM-85711, November 1983.
13. Newsom, J. R. and Mukhopadhyay, V.: The Use of Singular Value Gradients and Optimization Techniques to Design Robust Controllers for Multiloop Systems. NASA TM-85720, November 1983.
14. Ricketts, R. H.; Watson, J. J.; Sandford, M. C.; and Seidel, D. A.: Geometric and Structural Properties of a Rectangular Supercritical Wing Oscillated in Pitch for Measurement of Unsteady Transonic Pressure Distributions. NASA TM-85673, November 1983.
15. Singh, J. J.; Davis, W. T.; and Puster, R. L.: Proposed Fast-Response Oxygen Monitoring and Control System for the Langley 8-Foot High-Temperature Tunnel. NASA TP-2218, November 1983.
16. Woodrow, W.: XTRAN2L: A Program for Solving the General-Frequency Unsteady Transonic Small Disturbance Equation. NASA TM-85723, November 1983.
17. Gardner, J. E. and Dixon, S. C.: Loads and Aeroelasticity Division Research and Technology Accomplishments for FY 1983 and Plans for FY 1984. NASA TM-85740, January 1984.
18. Osher, S.; Whitlow, W., Jr.; and Hafez, M.: Entropy Condition Satisfying Approximations for the Full Potential Equations of Transonic Flow. NASA TM-85751, January 1984.
19. Pittman, C. M.; Brown, R. D.; and Shideler, J. L.: Evaluation of a Non-Catalytic Coating for Metallic TPS. NASA TM-85745, January 1984.
20. Cunningham, H. J. and Desmarais, R. N.: Generalization of the Subsonic Kernel Function in the s-Plane, With Applications to Flutter Analysis. NASA TP-2292, March 1984.

21. Batina, J. T. and Yang, T. Y.: Transonic Calculation of Airfoil Stability and Response With Active Controls. NASA TM 85770, March 1984.
22. Kelly, H. N. and Wieting, A. R.: Modification of NASA Langley 8-Foot High Temperature Tunnel to Provide a Unique National Research Facility for Hypersonic Air-Breathing Propulsion Systems. NASA TM-85783, May 1984.
23. Bland, S. R. and Seidel, D. A.: Calculation of Unsteady Aerodynamics for Four AGARD Standard Aeroelastic Configurations. NASA TM-85817, May 1984.
24. Adams, W. M., Jr.; Tiffany, S. H.; Newsom, J. R.; and Peele, E. L.: STABCAR - A Program for Finding Characteristic Roots of Systems Having Transcendental Stability Matrices. NASA TP-2165, June 1984.
25. Sawyer, J. W.: Effect of Preforming Adherends on Static and Fatigue Strength of Bonded Composite Single-Lap Joints. NASA TP-2324, June 1984.
26. Singh, J. J. and Puster, R. L.: New Technique for Calibrating Hydrocarbon Gas Flowmeters. NASA TM-85792, June 1984.
27. Avery, D. E.: Aerothermal Performance and Damage Tolerance of a Rene 41 Metallic Standoff Thermal Protection System at Mach 6.7. NASA TM-85773, July 1984.
28. Kelly, H. N. and Gardner, J. E. (Compilers): Advances in TPS and Structures for Space Transportation Systems. NASA CP-2315, July 1984.
29. Newsom, J. R.; Adams, W. M., Jr.; V. Mukhopadhyay, V. and Tiffany, S. H.: Active Controls: A Look at Analytical Methods and Associated Tools. NASA TM-86269, July 1984.
30. Edwards, J. W.; Bland, S. R.; and Seidel, D. A.: Experience with Transonic Unsteady Aerodynamic Calculations. TM-86278, August 1984.
31. Olsen, G. C. and Smith, R. E.: Aerothermal Loads Analysis for High Speed Flow Over a Quilted Surface Configuration. NASA TM-86280, August 1984.
32. Ricketts, R. H.; Sandford, M. C.; Watson, J. J.; and Seidel, D. A.: Subsonic and Transonic Unsteady-and Steady-Pressure Measurements on a Rectangular Supercritical Wing Oscillated in Pitch. NASA TM-85765, August 1984.
33. Shideler, J. L.; Webb, G. L.; and Pittman, C. M.: Verification Tests of Durable TPS Concepts. NASA TM-85745, September 1984.
34. Tai, H. and Runyan, H. L.: Lifting Surface Theory for a Helicopter Rotor in Forward Flight. NASA TM-86315, September 1984.

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35. Newsom, J. and Abel, I.: Experiences with the Design and Implementation of Flutter Suppression Systems. Presented at the 1st Annual NASA Aircraft Controls Workshop, October 25-27, 1983, Hampton, Virginia. NASA CP-2296.
36. Adams, W. M.; Tiffany, S. H.; and Newsom, J. R.: Tools for Active Control Systems Design. Presented at the 1st Annual NASA Aircraft Controls Workshop, October 25-27, 1983, Hampton, Virginia. NASA CP-2296.

37. Albertson, C. W.: Aerothermal Environment of an Apparatus for Testing Curved TPS Concepts. Presented at the NASA Symposium on Recent Advances in TPS and Structures for Future Space Transportation Systems, December 13-15, 1983, Hampton, Virginia. NASA CP-2315.
38. Robinson, J. C.; Bailey, J. P.; Wurster, K. E.; and Powell, R. W.: Overview of a Preliminary Aerothermostructural Design of a Tip-Fin Controller for Space Shuttle Orbiter. Presented at the NASA Symposium on Recent Advances in TPS and Structures for Future Space Transportation Systems, December 13-15, 1983, Hampton, Virginia. NASA CP-2315.
39. Shideler, J. L.; and Blair, W.: Design and Fabrication of Curved Superalloy Honeycomb Prepackaged TPS. Presented at the NASA Symposium on Recent Advances in TPS and Structures for Future Space Transportation Systems, December 13-15, 1983, Hampton, Virginia. NASA CP-2315.
40. Shideler, J. L.; Webb, G. L.; and Dearing, B.: Verification Tests of Advanced Multiwall and Prepackaged TPS. Presented at the NASA Symposium on Recent Advances in TPS and Structures for Future Space Transportation Systems, December 13-15, 1983, Hampton, Virginia. NASA CP-2315.
41. Tyson, R. W.: Acoustic Tests of Direct Bond FRCI Tiles on GR/PI Structure. Presented at the NASA Symposium on Recent Advances in TPS and Structures for Future Space Transportation Systems, December 13-15, 1983, Hampton, Virginia. NASA CP-2315.
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43. Hafez, M.; Whitlow, W., Jr.; and Osher, S.: Improved Finite Difference Schemes for Transonic Potential Calculations. Presented at the AIAA 22nd Aerospace Sciences Meeting, January 9-12, 1984, Reno, Nevada. AIAA Paper No. 84-0092.
44. McCain, W. E.: Comparison of Measured and Calculated Airloads on an Energy Efficient Transport Wing Model Equipped with Oscillating Control Surfaces. Presented at the AIAA 22nd Aerospace Sciences Meeting, Reno, Nevada, January 9-12, 1984. AIAA Paper No. 84-0301.
45. Bey, K. S.: Analytical Prediction of Aerothermal Environment in Wing-Elevon Cove. Presented at the 5th Symposium of Finite Element in Flow Problems, January 23-26, 1984, University of Texas. NASA TM-85711.
46. Taylor, A. H.; Sakata, I. F.; and Jenkins, J. M.: Thermostructural Analysis and Characterization of a Hybrid Metal Matrix Composites Spar Element. 8th Conference on Composites and Advanced Ceramic Materials, Cocoa Beach, Florida, January 1984.
47. Kelly, H. N. and Wieting, A. R.: Modification of NASA Langley 8-Foot High Temperature Tunnel to Provide a Unique National Research Facility for Hypersonic Air-Breathing Propulsion Systems. Presented at the AIAA 13th Aerodynamic Testing Conference, March 5-7, 1984, San Diego, California. AIAA Paper No. 84-0602.

48. Corridan, R. E.; Given, J. G.; and Kepley, B. M.: Transonic Wind Tunnel Investigation of the Galileo Probe Parachute Configuration. Presented at the AIAA 8th Aerodynamic Decelerator and Balloon Technology Conference, April 2-4, Hyannis, Massachusetts. AIAA Paper No. 84-0823-CP.
49. Batina, J. T. and Yang, T. Y.: Transonic Calculation of Airfoil Stability and Response with Active Controls. Presented at the AIAA/ASME/ASCE/AHS 25th Structures, Structural Dynamics and Materials Conference, Palm Springs, California. May 14-16, 1984. AIAA Paper No. 84-0873.
50. Bhatia, K. G.; Nagaraja, K. S.; and Ruhlin, C. L.: Winglet Effects on the Flutter of Twin-Engine-Transport-Type Wing. Presented at the AIAA/ASME/ASCE/AHS 25th Structures, Structural Dynamics and Materials Conference, May 14-16, 1984, Palm Springs, California. AIAA Paper No. 84-0905-CP.
51. Desmarais, R. N. and Rowe, W. S. A Method for Computing the Kernel of the Downwash Integral Equation for Arbitrary Complex Frequencies. Presented at the 25th AIAA SDM Conference and AIAA Dynamics Specialist Conference, Palm Springs, California, May 14-18, 1984. AIAA Paper No. 84-0983-CP.
52. Taylor, A. H.; Cerro, J. A.; and Jackson, L. R.: An Analytical Study of Reusable Flightweight Cryogenic Propellant Tank Design. Presented at the AIAA/ASME/ASCE/AHS 25th Structures Conference, Palm Springs, California, May 14-16, 1984. AIAA Paper No. 84-0865-CP.
53. Taylor, A. H.; Cerro, J. A.; Davis, R. C.; and Jackson, L. R.: Design Considerations for Reusable Flightweight Cryogenic Propellant Tanks. Presented at the 25th Structures, Structural Dynamics and Materials Conference, Palm Springs, California, May 14-16, 1984. AIAA Paper No. 84-0865-CP.
54. Martin, F. W., Jr. and Albertson, C. W.: Aerothermal Environment of a Blunted Three-Dimensional Nonaxisymmetric Body at Mach 6.8. Presented at the AIAA 19th Thermophysics Conference, June 25-28, 1984, Snowmass, Colorado. AIAA Paper No. 84-1698.
55. Olsen, G. C. and Smith, R. E.: Aerothermal Loads Analysis for High Speed Flow Over a Quilted Surface Configuration. Presented at the AIAA 17th Fluid Dynamics, Plasma Dynamics and Lasers Conference, June 25-27, 1984, Snowmass, Colorado. AIAA Paper No. 84-1630.
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57. Swann, R. T., Wood; G. M., Brown, R. D., Upchurch, B. T.; and Allen, G. J.: Non-Catalytic Surfaces for Metallic Heat Shields. Presented at the AIAA 19th Thermophysics Conference, June 25-28, 1984, Snowmass, Colorado. AIAA Paper No. 84-1734.
58. Newsom, J. R.; Adams, W. M., Jr.; V. Mukhopadhyay, V. and Tiffany, S. H.: Active Controls: A Look at Analytical Methods and Associated Tools. Presented at the 14th Congress of the International Council of the Aeronautical Sciences (ICAS), Toulouse, France, September 10-14, 1984.

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59. Gabel, R.; Kesack, W.; Reed, D. A.; and Ricks, R.: Planning, Creating, and Documenting a NASTRAN Finite Element Vibrations Models of a Modern Helicopter, Summary Report. Contract NAS1-16460, Phase II, Task 3, NASA CR-172229, October 1983.
60. Blair, W.; Meaney, J. E.; and Rosenthal, H. A.: Re-Design and Fabrication of Titanium Multi-Wall Thermal Protection System (TPS) Test Panels. Contract NAS1-15646, NASA CR-172247, January 1984.
61. Chipman, R.; Rauch, F.; Rimer, M.; and Muniz, B.: Body-Freedom Flutter of a 1/2-Scale Forward-Swept-Wing Model, An Experimental and Analytical Study. Contract NAS1-17102, NASA CR-172324, April 1984.
62. Borland, C. J.: Further Development of XTRAN3S Computer Program. Contract NAS1-17072, NASA CR-172335, May 1984.
63. Weinberg, B. C. and Shamroth, S. J.: Three-Dimensional Unsteady Viscous Flow Analysis Over Airfoil Sections. Contract NAS1-17573, NASA CR-172368, June 1984.
64. Bhatia, K. G. and Nagaraja, K. S.: Flutter Parametric Studies of Cantilevered Twin-Engine-Transport Type Wing Models With and Without Winglets, Volume I - Low Speed Investigations. Contract NAS1-17539, NASA CR-172410, September 1984.
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#### FY 85 PLANS

The FY 85 plans for the Loads and Aeroelasticity Division are broken out by each of the branches (technical areas) and selected highlights of proposed FY 85 milestones are presented.

##### Configuration Aeroelasticity Branch

For FY 85 the Configuration Aeroelasticity Branch (CAB) will continue its broadly based research program on dynamic and aeroelastic phenomena of aircraft and rotorcraft as summarized in figure 55.

Although a large portion of this work is associated with tests in the Langley Transonic Dynamics Tunnel (TDT) with companion theoretical studies, flight test programs are included as well. Currently two major flight test programs are nearing completion. These are the Higher Harmonic Control (HHC) program which uses an active control system for rotorcraft vibration reduction and the Decoupler Pylon (DCP) program for passive flutter suppression of wings with external stores. Both of these programs began with successful tests in the TDT and advanced to the flight test phase to evaluate characteristics that cannot be properly studied in wind-tunnel experiments.

With respect to wind-tunnel tests in the TDT, research studies are planned for both rotorcraft and airplanes. The rotorcraft studies will use the aeroelastic rotor experimental system (ARES). Rotorcraft work will focus on studies of aerodynamically and structurally advanced UH-60 blades and on new rotor concepts such as the hingeless rotor. Airplane focused studies include such items as active flutter suppression, aeroelastic tailoring, and shock induced oscillations. In addition to research studies, an aeroelastic verification test is planned for the JAS-39 airplane.

Work will continue in the area of prediction of helicopter vibration characteristics by using finite element modeling procedures. Studies involving the major airframe manufacturers will be continued. Basic modeling exercises for metal airplanes will be completed this year. In addition, the analysis/experiment correlation for coupled rotor-airframe vibrations of the AH-1G will be completed.

The density increase modification to the TDT will be completed early in 1985. Following a lengthy checkout period, the tunnel is expected to be back in operation in the spring of the year.

Significant progress in the development of a new data acquisition, display, and control system for the TDT is expected. Delivery of the data acquisition subsystem equipment is expected early in the year. The major components of the digital subsystem will be delivered this year also. The top-level software design will be completed in the first half of the year with the detailed software design finished by the end of the year.

Selected highlights of proposed FY 85 milestones are listed below and are shown by figures 56 through 60.

Aircraft Aeroelasticity:

- Pitching and Plunging Suspension System for 2-D Transonic Flutter Testing
- Modifications to Upgrade the Langley Transonic Dynamics Tunnel (Density Increase)
- Upgrading the Data Acquisition System for the Langley Transonic Dynamics Tunnel

Rotorcraft Aeroelasticity:

- Rotorcraft Dynamics and Aeroelasticity

Rotorcraft Structural Dynamics:

- A National Capability to Analyze Vibration as Part of Helicopter Structural Design

Each highlight is accompanied by descriptive material.

Unsteady Aerodynamics Branch

For FY 85, there will be a continuing level of activity in developing and applying computational finite-difference algorithms for the solution of the nonlinear unsteady fluid flow equations (figure 61). A major effort will be the continued development and application of the three-dimensional transonic small perturbation code, XTRAN3S. Correlation of XTRAN3S calculations with the large body of unsteady pressure measurements obtained at Langley will point directions for future improvements. The code will be modified to enable calculations for more realistic geometrical configurations beginning with the ability to treat multiple surfaces such as a wing and canard. Improved viscous boundary layer models will be incorporated and evaluated as well as the addition of second order terms to the equation to resolve nonuniqueness problems for cases with strong shocks. An unsteady full potential equation code incorporating monotone flux biased differencing will come to fruition. Development of a time accurate Euler equation code will be initiated as well as applications of a two-dimensional Navier-Stokes code. The linear supersonic integral equation method will be further developed and evaluated as a replacement for the Mach box algorithm. In addition, further developments of the rotor unsteady lifting surface theory will be pursued as well as parameter variation applications studies of this theory.

In parallel with the development of computational methods, the unsteady pressure measurement program will continue with a retest of the DAST ARW-2 wing, analysis of data obtained in the 0.3M TCT cryogenic test and fabrication work for the clipped delta wing/canard interference test planned for FY 86. The ARW-2 retest will provide further information leading to an understanding of novel transonic instabilities (shock induced and limit cycle flutter). The flutter suppression system contained in the ARW-2 wing will be modified to demonstrate the viability of active controls in suppressing such instabilities. Finally, fabrication of a novel 2-D pitching and plunging flutter mount system will provide an apparatus with which the effect of airfoil shape and geometry variations upon flutter may be ascertained.

Selected highlights of proposed FY 85 milestones are listed below and are shown by figures 62 through 65.

Theory Development:

- Development and Application of XTRAN3S
- Unsteady Full Potential Calculations Using a Flux Differencing Method
- Supersonic Integral Equation Code

Unsteady Pressure Measurements:

- Investigation of DAST ARW-2 Transonic Instability Boundary

Each highlight is accompanied by descriptive material.

Aeroservoelasticity Branch

There are several efforts planned for FY 85 in each of the three major areas of analysis methods, control law synthesis methods, and applications and validations as summarized in figure 66. Analyses will continue to define the static aeroelastic stability and control characteristics of the DAST ARW-2. There will be a continued development of optimal sensitivity analysis for both analysis and control law synthesis. Steady-state response and singular value constraints will be incorporated into our control law optimization methods. In the applications and validations area, a comparison of predicted and wind-tunnel data for ARW-2 will be undertaken. There is also a plan to apply the advanced control law optimization methods to integrated active controls for a supersonic-cruise fighter configuration. A joint LaRC-DFRF plan for flight testing the ARW-2 will be implemented. Reporting will begin on results of the spanwise gradient measurements of atmospheric turbulence.

Selected highlights of proposed FY 85 milestones are listed below and are shown by figures 67 through 69.

Control Laws Synthesis Methods:

- Parameter Sensitivity Methods in Control Law Synthesis

Applications and Validations:

- DAST Program
- DAST Flight Simulation Activities

Each highlight is accompanied by descriptive material.

## Aerothermal Loads Branch

For FY 85, there will be a continuing level of activity in all three disciplines as summarized in figure 70.

Thermal Loads - The major thrusts of the thermal loads research effort for FY 85 consist of five specific tasks: 1) results of mass addition film cooling tests of a large 12.5 degree cone will be analyzed to determine the cooling effectiveness of both forward facing and tangential coolant ejection; 2) document the results obtained from tests of the CSTA and continue correlation studies with both finite difference and finite element CFD codes; 3) Initiate tests using CSTA to study the effects of large pressure gradients along curved surfaces on gap heating; 4) Analyze experimental results from a wind tunnel model with shallow spherical protuberances that simulate thermally bowed metallic TPS tiles and a model that simulates a chordwise gap formed between adjacent wing elevons tested in the 8' HTT; and 5) Test various simulated arrays of thermally bowed metallic thermal protection systems to obtain heating and pressure loads at Mach 7 in laminar and turbulent boundary layers.

Integrated Analysis. - The major analytical thrust for the ALB analysis effort in Fy 85, which complements the thermal loads experimental effort, is the prediction of aerothermal loads. This effort includes continued application of finite difference solutions to complex flow configurations and development of finite element technology for aerothermal load prediction with the long-range goal of developing an integrated flow-thermal-structural analysis capability. Spectral techniques will be considered in early CY 85.

Facilities Operations and Development. - The facilities effort involves the safe and efficient operation and the expansion of the test capabilities of the six high energy facilities of the Aerothermal Loads Branch--the 8' High Temperature Tunnel (8' HTT), 1' x 3' High Enthalpy Aerothermal Tunnel (1' x 3' HEAT), the 7" High Temperature Tunnel (7" HTT), and the 1, 5, and 20 MW Aerothermal Arc Tunnels.

A major thrust will be the verification testing in the 7" HTT of techniques for providing alternate Mach numbers (4, 4.5, and 5) and oxygen enrichment of the methane air combustion products test stream. This effort is in support of the modification (FY 85 Coff) of the 8' HTT which will make it a unique national research facility for testing air-breathing propulsion systems for very high speed aircraft and missiles.

During FY 85 the Curved Surface Superalloy TPS, a Rene' 41 hot tankage structure, and three aerothermal loads models will be tested in the 8' HTT.

Flow in the test section of the 1' x 3' High Enthalpy Aerothermal Tunnel will be surveyed/calibrated, a new test section panel holder will be checked out and the oxygen enrichment system will be refurbished to permit calibration of high enthalpy flows in early FY 85.

Selected highlights of proposed FY 85 milestones are listed below and are shown by figures 71 through 73.

### Thermal Loads:

- Quilted Tile Array Simulating Thermally Bowed Metallic TPS
- Chine Tile-Gap Heating Model

Facilities Operation and Development:

- Modifications to Upgrade the Langley 8' High Temperature Tunnel (Oxygen Enrichment and Alternate Mach Number)

Each highlight is accompanied by descriptive material.

Thermal Structures Branch

There are several major research activities for FY 85 which collectively represent a concerted thrust to advance the state of the art in thermal structures (figure 74). System studies will continue to identify technology mods in the area of structures and materials. Emphasis will be on the AOTV class of vehicles. A scramjet strut will be delivered and tests initiated to validate the pin-fin cooling concept for propulsion structures. Airframe structural concepts will emphasize advanced composite material structural concepts which use advanced fabrication procedures which allow unique structural arrangements. Thermal protection system (TPS) concepts verifications will continue with the analysis and testing of curved superalloy panel arrays. Testing and data analysis will be completed on the C/C TPS panel joint configuration. Analytical methods for design analysis will be enhanced by improving the SPAR Thermal Analyzer. Analysis research will also address integrated thermal-structural analysis, to include improvement of error analysis and adaptive-grid techniques, algorithms, and interactive graphics. The integrated conceptual design analysis programs already developed will be documented.

Selected highlights of proposed FY 85 milestones are listed below and are shown by figures 75 through 78.

Structural Systems Studies:

- Structural System Study of Military Aerospace Vehicles

Concept Development:

- Curved Metallic TPS
- Analysis and Test of Stiffened Carbon-Carbon Compression Panels

Analytical Methods and Applications:

- Thermal/Structural Analysis and Optimization

CONCLUDING REMARKS

This publication documents the FY 1984 accomplishments, research and technology highlights, and FY 1985 plans for the Loads and Aeroelasticity Division.

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LANGLEY RESEARCH CENTER

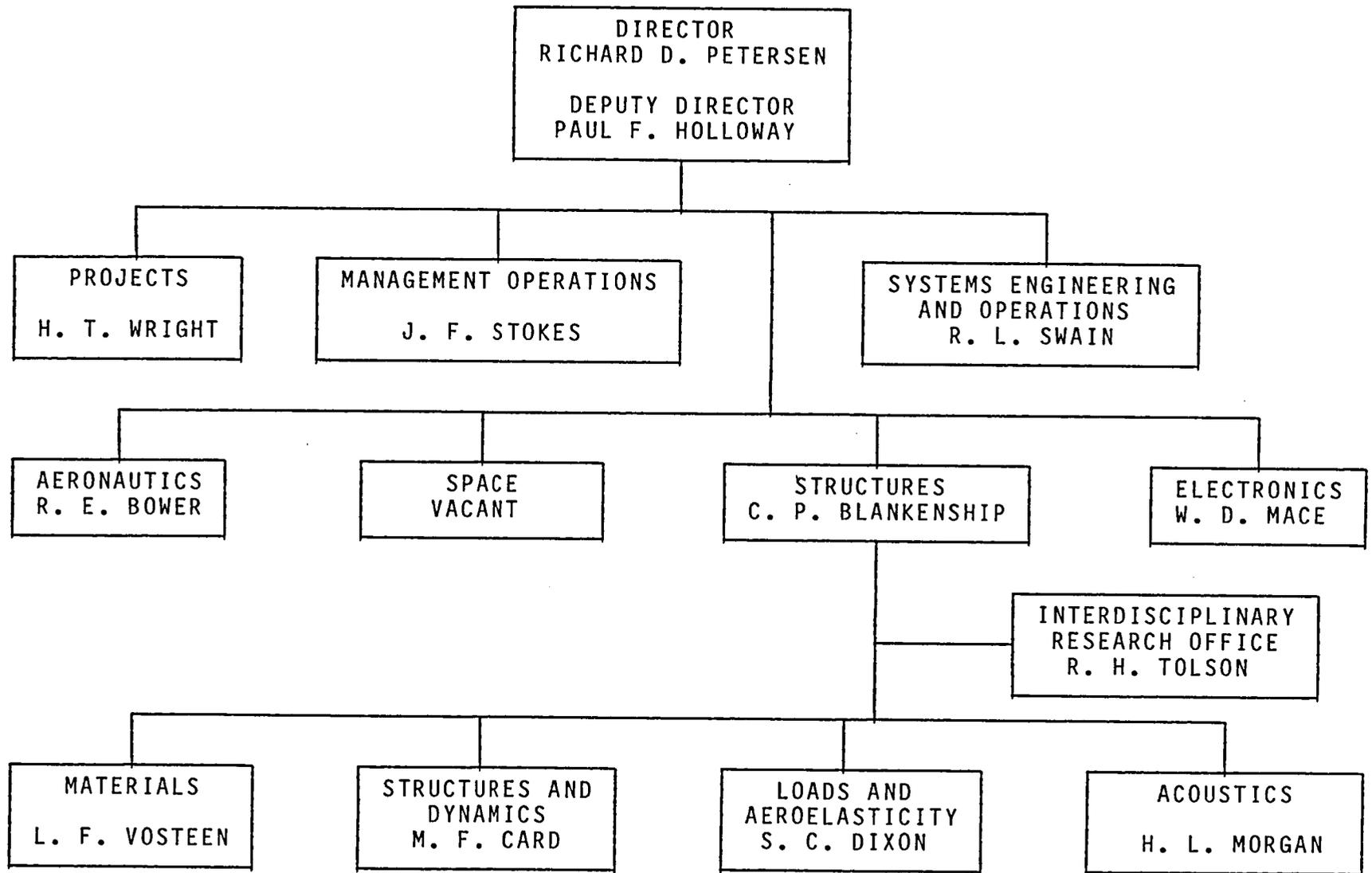


Figure 1.

LOADS AND AEROELASTICITY DIVISION

5

CHIEF: SIDNEY C. DIXON  
ASSISTANT CHIEF: PERRY W. HANSON  
CHIEF SCIENTIST: E. CARSON YATES, JR.  
TECHNICAL ASSISTANT: JAMES E. GARDNER

CONFIGURATION AEROELASTICITY BRANCH

13 + 7A

HEAD: ROBERT DOGGETT  
HDQ. DETAIL: RODNEY RICKETTS

- AIRCRAFT AEROELASTICITY
- ROTORCRAFT AEROELASTICITY
- ROTORCRAFT STRUCTURAL DYNAMICS

UNSTEADY AERODYNAMICS BRANCH

12

HEAD: JOHN EDWARDS

- THEORY DEVELOPMENT
- UNSTEADY PRESSURE EXPERIMENTS

AEROSERVOELASTICITY BRANCH

9

HEAD: PERRY HANSON, ACTING  
ASST: JERRY NEWSOM

- ANALYSIS METHODS
- CONTROL LAWS SYNTHESIS METHODS
- APPLICATIONS AND VALIDATIONS

AEROTHERMAL LOADS BRANCH

17

HEAD: ALLAN WIETING  
ASST: NEALE KELLY

- THERMAL LOADS
- INTEGRATED THERMAL-FLUID ANALYSIS
- FACILITIES OPERATIONS AND DEVELOPMENT

TOTAL NASA GS - 68

TOTAL ARMY GS - 7

THERMAL STRUCTURES BRANCH

12

HEAD: DONALD RUMMLER

- STRUCTURAL SYSTEMS STUDIES
- CONCEPT DEVELOPMENT
- ANALYICAL METHODS AND APPLICATIONS

Figure 2.

LOADS AND AEROELASTICITY DIVISION

LONG-RANGE THRUSTS

AERONAUTICS

- 0 PREDICTION AND CONTROL OF AEROELASTIC STABILITY AND RESPONSE
- 0 LOADS, (MATERIALS), AND STRUCTURES FOR HIGH-SPEED VEHICLES

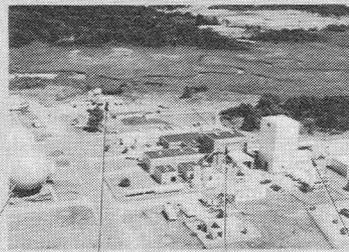
SPACE

- 0 LOADS, (MATERIALS), AND STRUCTURES FOR SPACE TRANSPORTATION SYSTEMS

Figure 3.

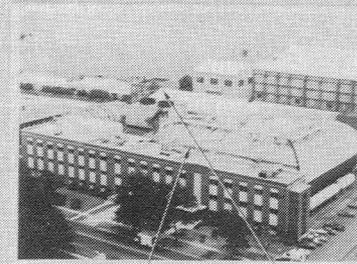
# LOADS AND AEROELASTICITY DIVISION

## AEROTHERMAL LOADS COMPLEX



### FACILITIES

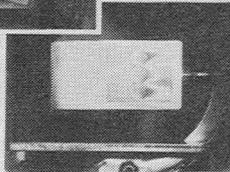
## TRANSONIC DYNAMICS TUNNEL



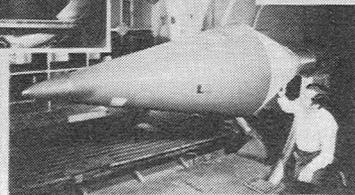
### ARC JETS



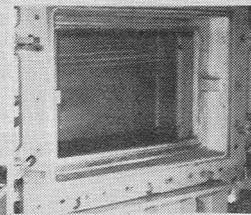
7" HTT



8' HTT



1' X 3' HEAT



### HOVER FACILITY

TDT

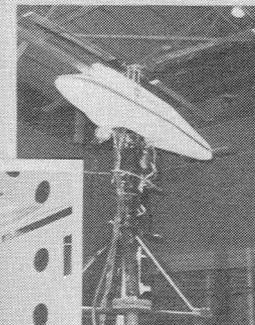
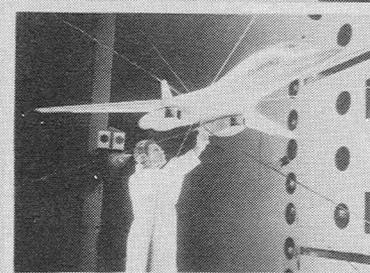
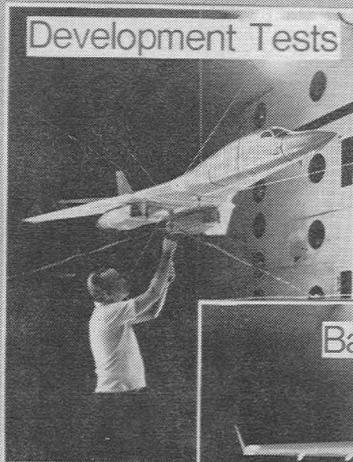


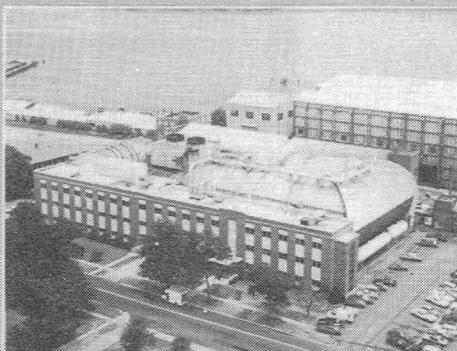
Figure 4.

## CONFIGURATION AEROELASTICITY

### Aircraft



### Transonic Dynamics Tunnel



### Rotorcraft

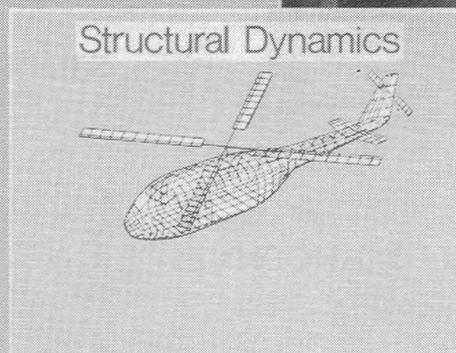
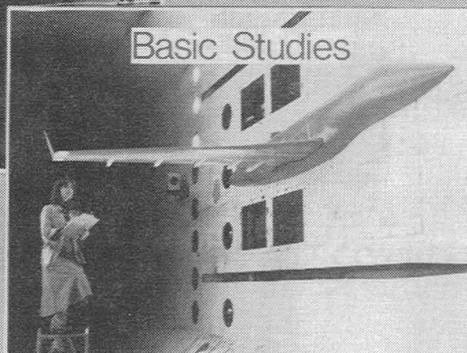


Figure 5.

CONFIGURATION AEROELASTICITY  
FIVE YEAR PLAN

DISCIPLINE	FY 84	FY 85	FY 86	FY 87	FY 88	EXPECTED RESULTS	
<u>AIRCRAFT AEROELASTICITY</u>	DCP FLT TEST	AERO TAILORING				ACTIVE/PASSIVE CONTROL OF AERO-ELASTIC RESPONSE	
		ACTIVE CONTROLS					
		SIO/SCW/WINGLETS/ARROW					DATA BASE, NEW CONCEPTS/CONFIGS
		MILITARY/CIVIL FLUTTER CLEARANCE					FLUTTER-FREE DESIGNS
		TEST TECHNIQUES ..... TDT IMPROVEMENTS					
<u>ROTORCRAFT AEROELASTICITY</u>	HIGHER HARMONIC CONTROL					REDUCED VIBRATION THROUGH ACTIVE CONTROL	
	PARAM TIP		HIGH SPEED	NODALIZED			
	AEROELASTICALLY OPTIMIZED ROTOR					ROTOR DESIGN FOR MINIMUM VIBRATION	
	JVX	VAR SPEED	NEW ROTOR CONCEPTS EVALUATIONS			NEW ROTOR CHARACTERISTICS EXPLORED	
	HINGELESS		BEARINGLESS				
	<u>ROTORCRAFT STRUCTURAL DYNAMICS</u>	CH-47D	BASIC MODELING EXERCISES, AH-64, UH-60 ...				SUPERIOR FEM CAPABILITY
		FOREFRONT TECHNOLOGIES			INTEGRATED ROTOR/AIRFRAME ANALYTICAL METHOD		
BLADE CORRELATIONS							
AH-1G CORRELATION							
APPROACH		MODEL	EVALUATION			ROTOR MATH MODEL ADAPTED FOR AIRFRAME DESIGN ANALYSIS	

Figure 6.

## SPANWISE CURVATURE RAISES FLUTTER DYNAMIC PRESSURE

José A. Rivera  
Configuration Aeroelasticity Branch  
Extension 2661

RTOP 505-33-43

Research Objective - The Navy is designing curved fins for use on a new missile. These fins, which are curved in the spanwise direction, fold around the missile body so it can fit into existing submarine missile storage tubes. Because there is no information in the literature on the effects of spanwise curvature on flutter, the Navy sought NASA Langley's help. To initiate research in this area and thus provide the needed data, an experimental/analytical study of a generic wing planform has been conducted to determine the effect of spanwise curvature on flutter.

Approach - A series of rectangular planform wings (figure 7(b)) of aspect ratio 1.5 were flutter tested in the NASA Langley Transonic Dynamics Tunnel (TDT). The only difference between models in the series was in curvature which ranged from zero (flat) to 1.05. (Curvature as used here is the reciprocal of the radius of curvature.) Each model had a NACA 65-A010 airfoil section. Flutter analyses were conducted for correlation with the experimental results by using structural finite element methods and planar subsonic lifting surface theory. Displacements perpendicular to the curved surfaces were used in calculating the unsteady aerodynamic loads with the planar theory.

Accomplishment Description - Measured and calculated results obtained are shown in the figure as the variation of dynamic pressure at flutter with wing curvature. All results are for a Mach number of 0.7. These results show that curvature increases the flutter speed for wings with an airfoil cross section. Although the reason for this increase was not determined specifically during this study, the increase is believed to be caused by a change in character of the vibration mode shapes as curvature is increased. The analytical results are in good agreement with the experimental results. The kernel function analysis gave conservative results for the uncurved wing and became slightly unconservative as curvature increased, whereas the doublet lattice results were conservative throughout.

Future Plans - The results will be published in a NASA TP.

Figure 7(a).

## SPANWISE CURVATURE RAISES FLUTTER DYNAMIC PRESSURE

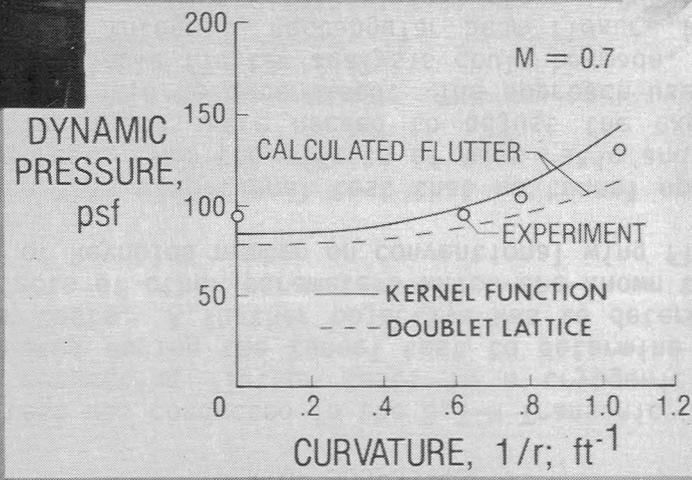
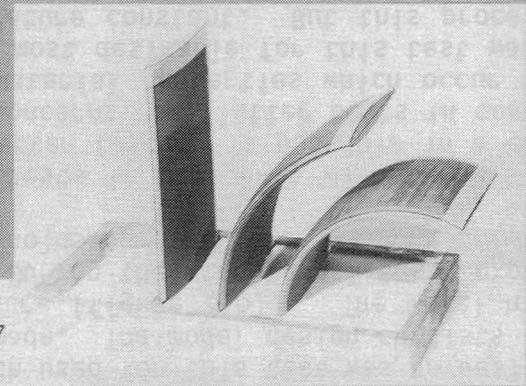
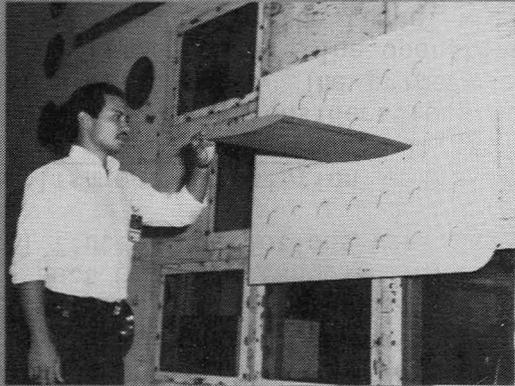


Figure 7(b).

## FLUTTER TESTING TECHNIQUES DEVELOPED FOR USE IN CRYOGENIC WIND TUNNELS

Stanley R. Cole  
Configuration Aeroelasticity Branch  
Extension 2661

RTOP 505-33-43

Research Objective - A flutter test was conducted in the 0.3-M Transonic Cryogenic Tunnel at NASA Langley to explore the feasibility of conducting flutter tests in a cryogenic wind tunnel. Several tunnel operating procedures were evaluated during the tunnel test to determine the more appropriate procedure for conducting cryogenic flutter tests. A further objective was to determine if Reynolds number effects could be separated from the effects of other parameters which are known to influence flutter and, if so, how significant are the effects of Reynolds number on conventional wing flutter.

Approach - It was realized before this wind-tunnel test that no tunnel operating procedure could completely separate Reynolds number effects from the effects of mass ratio and temperature. Therefore, reliable analytical flutter prediction trends were needed to adjust the experimental results so that the Reynolds number effects on flutter could be determined. The approach used for this test was to design a simple flutter model for which a reliable flutter analysis could be made. The model design consists of a relatively rigid wing mounted on an integral, rectangular beam flexure (figure 8(b)). The model had a 1.5 aspect ratio rectangular wing with a NACA 64A010 airfoil shape. During the wind-tunnel test, experimental flutter predictions were made for several Mach numbers and Reynolds numbers.

Accomplishment Description - The feasibility of conducting flutter tests in cryogenic wind tunnels was examined through this test. It has been found that, while useful flutter testing is possible in a cryogenic tunnel, many considerations must be made which are not usual concerns for flutter tests in conventional wind tunnels. The primary concern is the large changes in material properties which occur with changing temperature. The operating procedure that was found to be most desirable for this test was to increase pressure in the tunnel while holding Mach number and temperature constant. But this procedure required that the Reynolds number varied as the flutter condition was approached. Analytical trends obtained through flutter analysis were used to adjust the experimental results for the effects of mass ratio and temperature. In this manner, the effects of Reynolds number on flutter were found to be small for this model. This was not unexpected for the airfoil used in these tests. A slight decrease in the flutter dynamic pressure was found to occur with increasing Reynolds No. over the range of 5 to  $20 \times 10^6$ .

Future Plans - Results of the cryogenic flutter model test will be documented in a paper at the 1985 AIAA SDM Conference.

Figure 8(a).

REYNOLDS NUMBER EFFECTS ON FLUTTER SHOWN TO BE SMALL IN 0.3 - M  
TRANSONIC CRYOGENIC TUNNEL

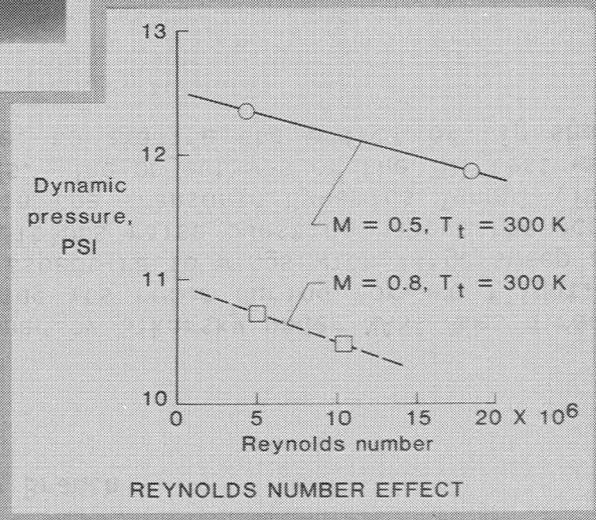
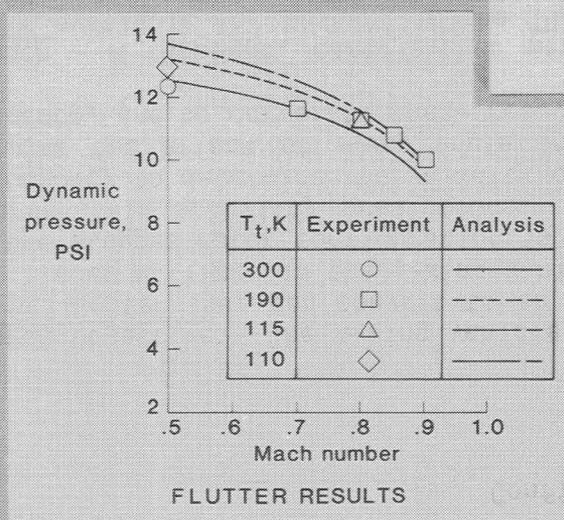
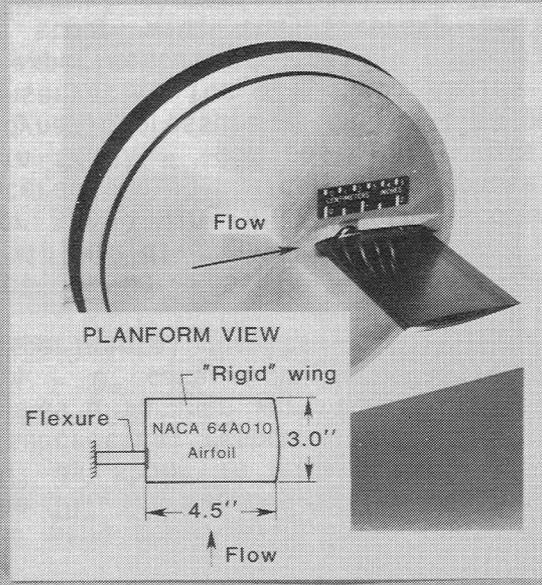


Figure 8(b).

# X-WING AEROELASTIC DIVERGENCE ANALYTICAL METHODS VALIDATED BY TDT TEST RESULTS

Stanley R. Cole  
Configuration Aeroelasticity Branch  
Extension 2661

RTOP 532-06-13

Research Objective - The X-Wing vehicle, currently being designed by Sikorsky under NASA Ames direction, is a helicopter that can convert to an airplane mode by stopping its rotor during forward flight. The rotor blades utilized in the X-Wing vehicle are unusually high aspect ratio wings with large sweep angles when the stopped rotor configuration is considered. These conditions raise questions about the accuracy of aeroelastic stability predictions. A test was conducted in the Transonic Dynamics Tunnel (TDT) at NASA Langley to determine the validity of the analytical methods. The objectives of the TDT test were to determine the effect of sweep angle on aeroelastic divergence, to examine the effect of tip shape on divergence, and to correlate these results with analysis.

Approach - A straight, cantilevered wing model with an aspect ratio similar to the X-Wing design was readily available for testing in the TDT. A support system was designed to allow the model to be set at  $\Lambda = 0^\circ, -15^\circ, -30^\circ, -45^\circ, \text{ or } -60^\circ$  as shown in figure 9(b). At each sweep angle, the divergence instability was predicted using a subcritical response technique. Two tip shapes (shown in the figure) were tested in the  $\Lambda = -45^\circ$  position as a further verification of the analytical methods. These two tip shapes would be identical in the  $\Lambda = 0^\circ$  case. An aeroelastic divergence analysis of the model was conducted for comparison with the experimental results.

Accomplishment Description - Predictions were obtained for each sweep angle with the rectangular tip. The high-aspect ratio model had a minimum divergence point near  $\Lambda = -45^\circ$ . This is significant because the most critical configuration for the X-Wing will be the stopped rotor position rather than other azimuth angles which will be encountered during the conversion phase of flight. The second tip, which is parallel to the flow, was tested in the  $\Lambda = -45^\circ$  position. This tip configuration resulted in a 14 percent increase in the divergence dynamic pressure which was also predicted by analysis. The analytical predictions for this model were conservative for all sweep angles except  $\Lambda = 0^\circ$ . At zero sweep, the analytical result agreed with the experimental prediction. The analytical methods used in this study were also used in a comparative study with the Sikorsky analysis of the actual X-Wing design configuration. Based on this analytical comparison and the TDT test data, it is concluded that the X-Wing design results are conservative predictions of static divergence speeds.

Future Plans - The X-Wing model test is complete. Results of the study will be documented in a NASA Technical Memorandum for use by the NASA X-Wing Project Office.

Figure 9(a).

# X-WING AEROELASTIC DIVERGENCE ANALYTICAL METHODS VALIDATED BY TDT TEST RESULTS

## Test Configurations

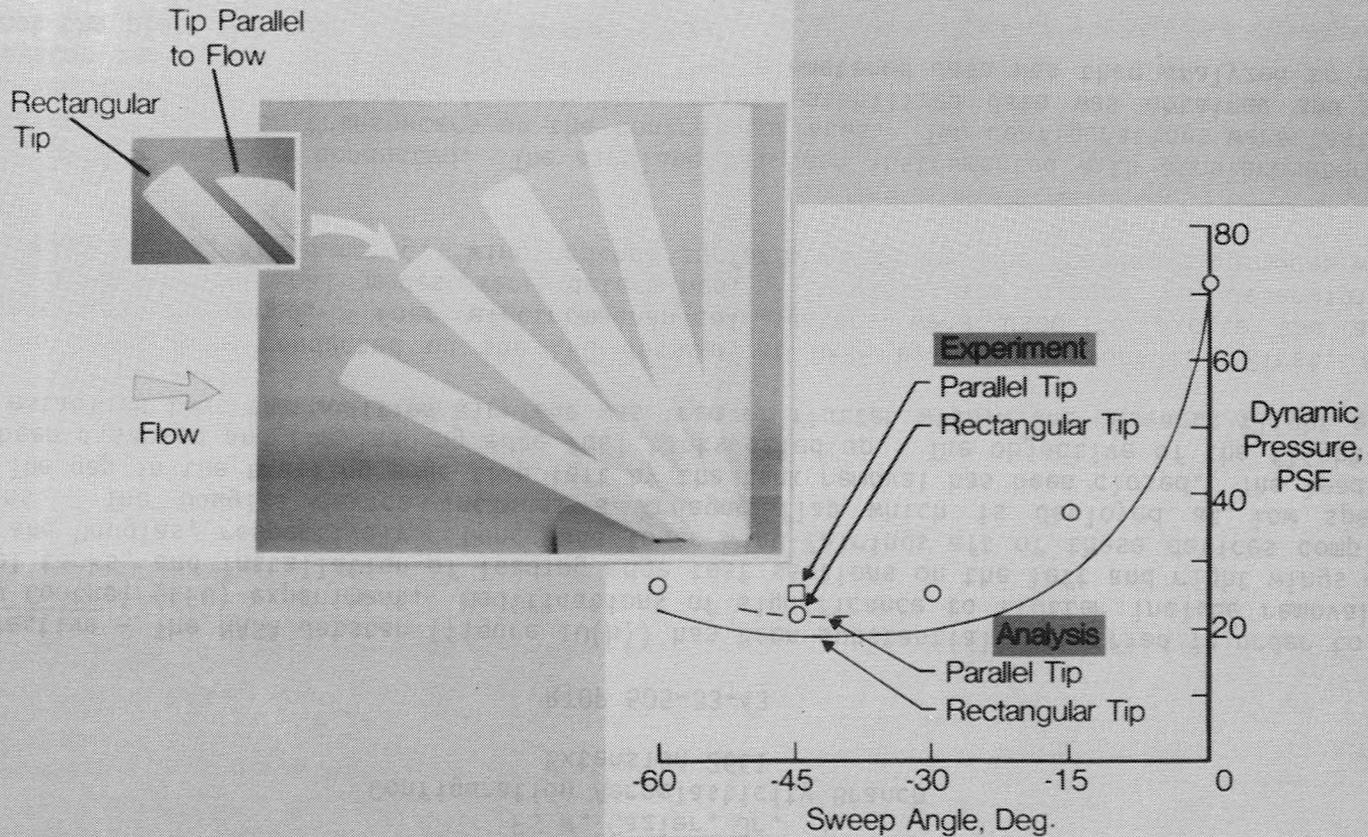


Figure 9(b).

## FLIGHT TEST SHOWS LAMINAR FLOW CONTROL RESEARCH AIRCRAFT FREE OF FLUTTER

F. W. Cazier, Jr.  
Configuration Aeroelasticity Branch  
Extension 2661

RTOP 505-33-43

Research Objective - The NASA Jetstar (figure 10(b)) has been substantially modified in order to fly the Laminar Flow Control (LFC) experiment. Modifications of significance to flutter include removal of the external fuel tanks, and installation of leading edge test sections on the left and right wings designed by Lockheed and Douglas, respectively. Upper and lower wing fairings aft of these devices complete the test sections. The Douglas device includes a Krueger flap which is deployed at low speeds and altitudes. The gap in the trailing edge flap left by the tank removal has been closed. The leading edge flaps have been disabled and the leading edge fuel tanks dried up. The objective of the flight flutter test was to establish that the modified airplane was free of flutter within the research flight envelope.

Approach - Two tests were conducted on the LFC Jetstar at DFRF by LaRC personnel. First, a ground vibration test was performed. Four electromechanical shakers were used to excite the airplane. Fundamental aircraft structural modes were determined. Emphasis was placed on determining wing frequencies, mode shapes, and damping. Wing control surface modes, and the Krueger-flap modes were also determined.

A flight flutter test was then conducted. The airplane had been instrumented with accelerometers on the lifting surfaces and position transducers on the control surfaces. Two configurations were tested: the Krueger flap deployed and retracted. At each test point, stabilized data was obtained and then the aileron, elevator and rudder were manually pulsed. The telemetered data was then analyzed to determine whether or not the plane was near a flutter instability.

Accomplishment Description - The ground vibration test mode shapes and frequencies agreed well with predicted values used in flutter analyses. In the subsequent flight tests, the aircraft was shown to be flutter free over its research flight envelope. This envelope encompassed all flight test conditions for the laminar flow experiments plus additional margin for overspeed conditions. Both the Krueger flap retracted and extended envelopes were cleared over their respective envelopes.

Future Plans - With the airplane being cleared of flutter, the laminar flow control experiments proper can now be conducted. The two leading edge test sections are now being checked.

Figure 10(a).

## FLIGHT TEST SHOWS LAMINAR FLOW CONTROL RESEARCH AIRCRAFT FREE OF FLUTTER

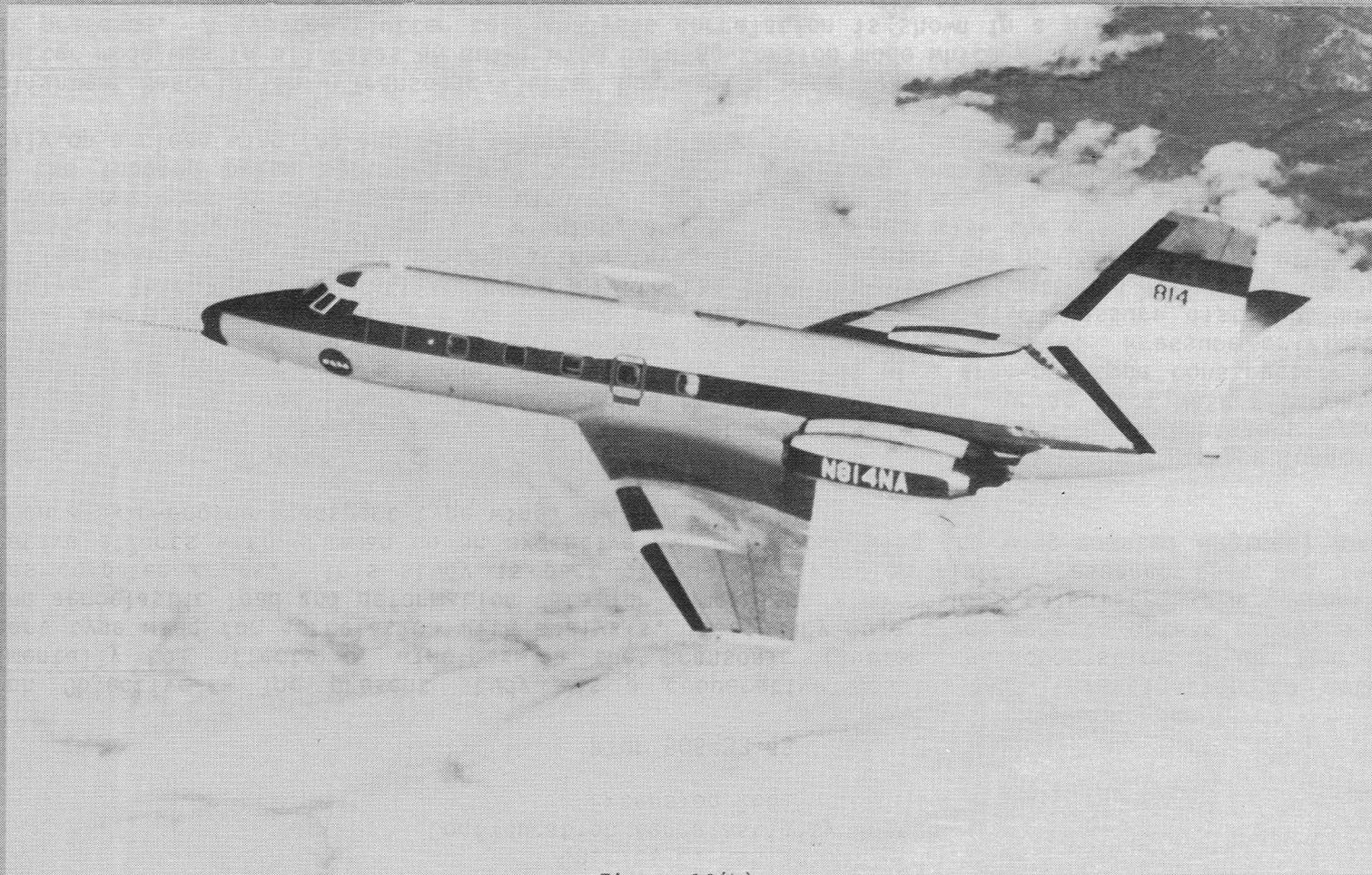


Figure 10(b).

# FLUTTER OF FOUR-ENGINE TRANSPORT WING WITH WINGLET PREDICTABLE BY ANALYSIS

Charles L. Ruhlin  
Configuration Aeroelasticity Branch  
Extension 2661

RTOP 505-33-43

Research Objective - The present study was a cooperative Douglas/NASA investigation to determine experimentally the effects of winglets on the transonic flutter characteristics of a four-engine transport type wing for correlation with analysis. Secondary objectives were to obtain steady pressure data and aeroelastic load and deformation data for comparison with results calculated from current fluid dynamics computer codes. This study is part of on-going winglet flutter research that has included cooperative efforts with Grumman on an executive jet transport wing (no wing-mounted engines) and with Boeing on a twin-engine transport type wing.

Approach - Transonic flutter tests were conducted jointly in the Langley Transonic Dynamics Tunnel (TDT) using a Douglas built, .08-size semispan flutter model of an advanced four-engine transport type wing (figure 11(b)). The model was cantilever-mounted through a soft roll spring to a NASA five-component aerodynamic force balance on the tunnel sidewall. The wing was of a spar-pod type construction with a supercritical airfoil section and flexibly mounted simulated engine nacelles. Pressure orifices were located at the 0.69 semispan station to measure the chordwise steady static pressure distribution via a scannivalve. Thin tufts were attached sparsely to the undersides of the outboard four wing segments to detect flow separation. Flutter configurations investigated included the nominal wing and engines with (1) a basic winglet, (2) a lightweight winglet, and (3) a wing tip with the winglet replaced by a boom having the same mass as the lightweight winglet. The basic winglet was also flutter tested on the wing having the inboard pylon softened in pitch. Aerodynamic load and pressure measurements were made primarily on a clean wing (no engines) with wing tip boom configuration.

Accomplishment Description - Transonic flutter boundaries were measured for the flutter configurations. The flutter mode was in all cases an outer wing bending-torsion mode which was strongly influenced by the winglet presence. A typical flutter test-analysis correlation is shown in a plot of normalized flutter speed against Mach number  $M$  for the nominal wing and engines with the basic winglet (see figure). It can be seen that the theory agreed well with test results at the critical  $M$  (2.8), but was somewhat low at the lower  $M$ . The analysis employed doublet lattice aerodynamics corrected by a scalar (based on experimental data) at each  $M$ . Overall the measured static pressures, aeroelastic load and deformation data agreed reasonably with the analytically predicted results. The experiments also provided useful guides for aerodynamic analysis improvements.

Future Plans - There are no plans for additional testing of this model.

Figure 11(a).

# FLUTTER OF FOUR-ENGINE TRANSPORT WING WITH WINGLET PREDICTABLE BY ANALYSIS

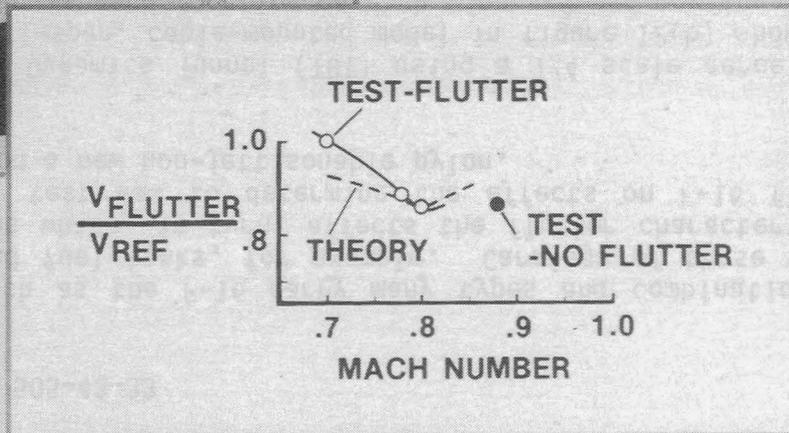
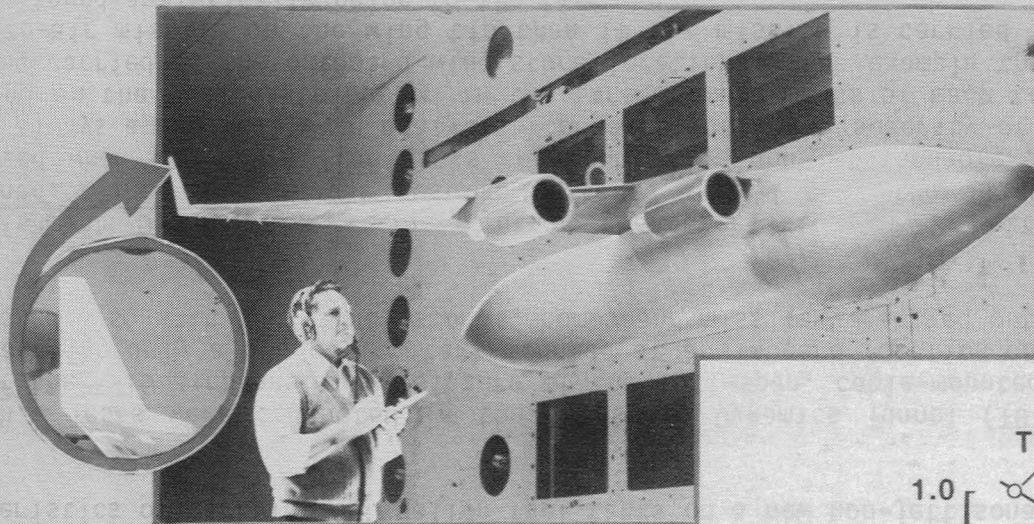


Figure 11(b).

## EFFECTS OF 600 GALLON FUEL TANKS ON F-16 FLUTTER CHARACTERISTICS STUDIED IN TDT

Moses G. Farmer and Frank W. Cazier, Jr.  
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Extension 2661

RTOP 505-43-33

Research Objective - Modern fighter airplanes such as the F-16 carry many types and combinations of external wing-mounted stores; bombs, missiles, and fuel tanks, for example. Carriage of these stores changes the dynamic characteristics of the airplane which, in turn, affects the flutter characteristics of the airplane. The objective of the current test was to determine the effects on F-16 flutter characteristics of carrying 600 gallon fuel tanks on a new non-jettisonable pylon.

Approach.- Tests were conducted in the Transonic Dynamics Tunnel (TDT) using a 1/4 scale aeroelastic model of the F-16 airplane. The picture of the full-span, cable-mounted model in figure 12(b) shows the model carrying only a 600 gallon tank mounted at an inboard location on each wing. Other configurations were tested also with additional stores, bombs and missiles, carried outboard on the wings.

Accomplishment Description - The flutter characteristics of 27 external store configurations were determined. The experimental results obtained showed that the flight envelope for the airplane is restricted when the 600 gallon tanks are mounted on non-jettisonable pylons. The experimental results did not always agree well with analytical predictions. The severity of the flight restriction was found to depend on the fuel distribution in the three compartments of each tank and also on what other stores are being carried at the outboard wing store locations. For example the restriction is more severe with an air-to-air missile on the wing tip than if the missile is carried under the wing. A similar result had been found analytically prior to the test.

Future Plans - These test results will be used by General Dynamics together with analytical and flight test results to establish flight operation procedures and restrictions for the aircraft.

Figure 12(a).

## EFFECTS OF 600 GALLON FUEL TANKS ON F-16 FLUTTER CHARACTERISTICS STUDIED IN TDT



Figure 12(b).

EFFECTS OF NEW MULTI-PURPOSE PYLONS ON F-16 FLUTTER CHARACTERISTICS STUDIED IN TDT

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RTOP 505-43-33

Research Objective - Modern fighter airplanes such as the F-16 are required to carry stores on pylons mounted under the wings of the aircraft. In an attempt to improve the operating efficiency of the F-16, a new multi-purpose pylon is being developed which will enable several bombs or missiles to be quickly loaded onto an airplane, thereby, reducing turn-around time between combat missions. The multi-purpose pylons (MPPS), which are larger and heavier than previous pylons, can carry stores on two side-by-side launcher rails. This current study was conducted to determine the effects of MPPS on F-16 flutter characteristics.

Approach - A test was conducted in the NASA Transonic Dynamics Tunnel (TDT) using a 1/4 scale aeroelastic model of the F-16 airplane. The model was mounted on the two-cable mount system. The wing span of the model is about 7.5 feet. Figure 13(b) shows a view looking downstream and up underneath the right wing of the model. An air-to-air missile can be seen on the wing tip. The aft portion of a fuel tank can be seen near the wing root. An MPP, which is located at the wing midspan, has an MK-84 bomb attached to its outer rail. For other configurations, different combinations of stores were installed on the MPPS and at the other store locations on each wing. A total of 21 configurations were tested.

Accomplishment Description - The experimental results showed that the airplane flight envelope will be restricted by flutter if MPPS like the ones tested are used on the F-16. This result was consistent with those obtained by analysis. In an attempt to reduce the flight restrictions, the location and stiffness of attachment points between the stores and the MPPS were varied. The changes, however, had little effect on the flutter characteristics.

Future Plans - No further wing-tunnel tests are currently planned. General Dynamics, the MPPS designer, is currently using the wind-tunnel test data and analytical results to develop modifications to the MPPS design that will provide satisfactory flutter characteristics.

Figure 13(a).

## EFFECTS OF NEW MULTI-PURPOSE PYLONS ON F-16 FLUTTER CHARACTERISTICS STUDIED IN TDT

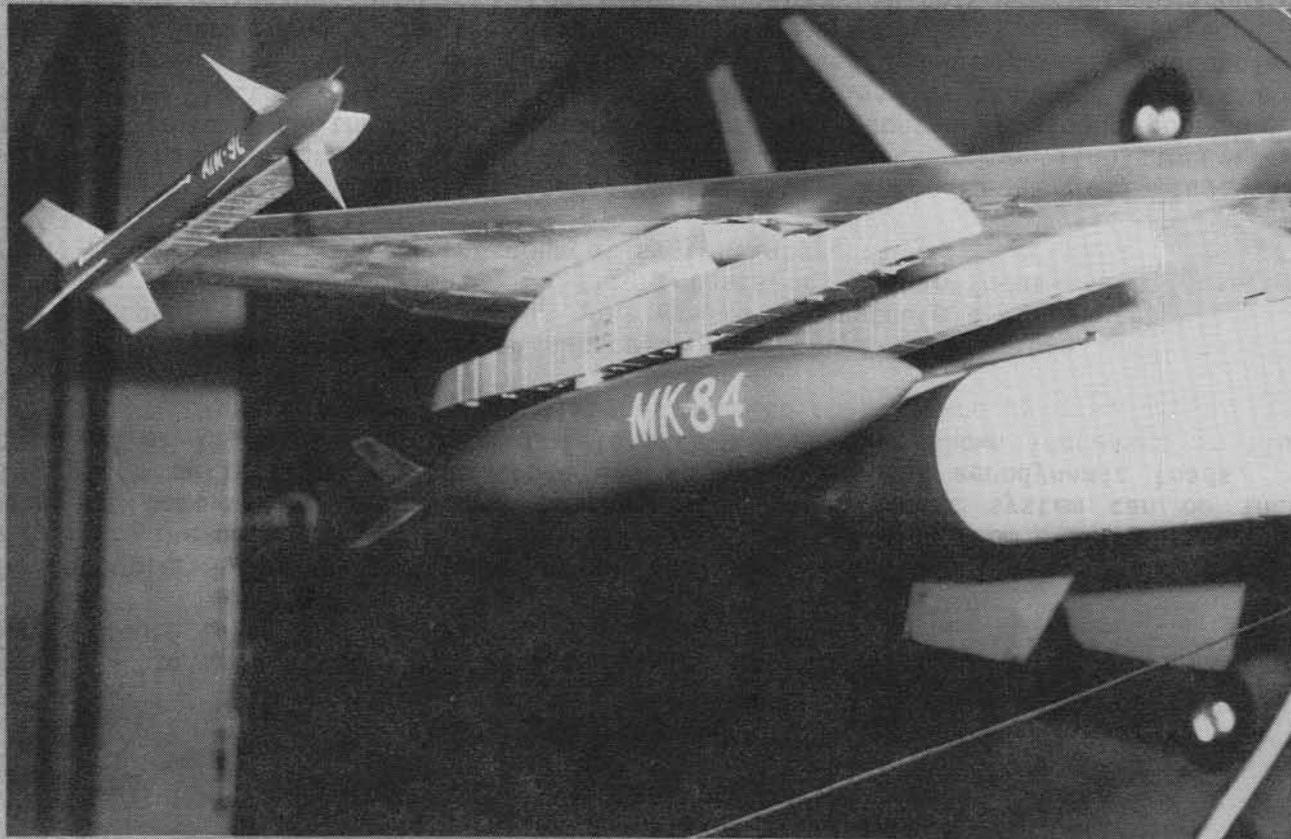


Figure 13(b).

## DECOUPLER PYLON PROGRAM

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RTOP's 505-33-43 and 533-02-73

Research Objective - To demonstrate passive suppression of wing/store flutter on a modern lightweight fighter airplane (figure 14(b)).

Approach - The Decoupler Pylon Program consists of analyses, wind-tunnel tests, and flight tests of a NASA patented pylon. The decoupler pylon dynamically isolates the wing from external store pitch inertia effects by means of soft-spring and damper components. An alignment system can be incorporated to minimize static pitch deflections of the store due to maneuvers and aerodynamic loads. Analyses and wind-tunnel tests of YF-17 and F-16 flutter models with stores have shown increases in flutter dynamic pressure in excess of 100-percent over the same stores mounted on standard pylons.

The flight test program demonstrated flutter suppression on the F-16 with the same store configuration tested in the wind tunnel. The decoupler pylon goal was to demonstrate a 70-percent increase in flutter dynamic pressure over a production pylon. The flight tests bring into focus the effects of turbulence, flight maneuvers, store ejection and flight control system interactions.

Accomplishment Description - The flight test stores configuration includes the following: AIM-9J wingtip missiles, GBU-8 bombs near midspan, and half full 370 gallon fuel tanks. The flights were first made with the 2250-pound GBU-8 mounted on the F-16 production pylon. The continual strong pounding oscillations which characterize the flutter with the mid-wing store mounted on its standard pylon are shown on the right side of the figure. The flights were then repeated with the GBU-8 mounted on decoupler pylons. As can be seen in the figure, the decoupler pylon successfully suppresses the wing/store flutter. No evidence of flutter was detected throughout the flight envelope of this stores configuration. An 84% increase in dynamic pressure over the production pylon was demonstrated by the decoupler pylon. Maneuvers up to 4 g's were performed throughout the flight envelope. Only small pitch deflections of the store were recorded such that the alignment system was seldom needed. It performed well and realigned the store when activated. At the conclusion of the flight test program, one GBU-8 was ejected demonstrating that weapon separation is normal from the decoupler pylon.

Future Plans - The program is complete. All objectives of the program were met.

Figure 14(a).

## DEC OUPLER PYLON SUPPRESSES FLUTTER



### Wingtip Acceleration

M = .9 Altitude = 10,000 feet

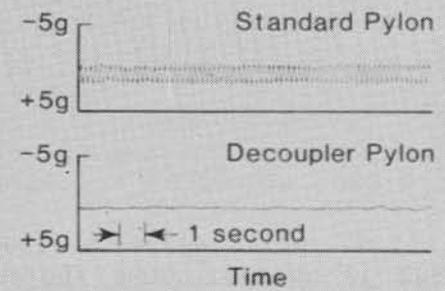


Figure 14(b).

## HIGHER HARMONIC CONTROL TECHNOLOGY TRANSFER AND FLIGHT DEMONSTRATION

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Extension 2661

RTOP 505-42-23

Objective - The objective of the Higher Harmonic Control (HHC) Flight Test Program (a combined program of NASA-LaRC, Army Structures Lab and Hughes Helicopter, Inc., the prime contractor) is to implement on a full scale helicopter for flight research a vibration reduction concept that was shown to be feasible in wind-tunnel studies. The ultimate goal was a program review and technology demonstration of the HHC equipped OH-6A Helicopter for members of the helicopter industry, government researchers and users.

Approach - The Higher Harmonic Control flight test program was initiated in September 1980 after a preliminary design study identified an Army OH-6A (S/N 68-17230) helicopter as the best testbed aircraft available for concept demonstration. The flight test helicopter is shown in figure 15(b). Higher Harmonic Control is achieved by superimposing sinusoidal non-rotating swashplate motions at the blade passage frequency (4 per revolution for a 4-bladed rotor) upon the basic collective and cyclic flight control inputs. The frequency of the inputs is the blade passage frequency because this is the frequency of the loads which are to be suppressed. The amplitude and phase of the HHC inputs are calculated by an onboard computer using an optimum adaptive control law with tri-axial accelerometers at the pilots seat providing vibration information. The HHC concept required the design and construction of high frequency hydraulic actuators and servovalves, and a flightworthy electronic control unit to extract the 4 per rev vibrations from total accelerometer signals. In addition a softness in the OH-6A control system was identified in late 1981 which caused a redesign of the OH-6A basic control system bellcranks and mixer.

Accomplishments - Flight testing of this active control concept began in the summer of 1982 at the U.S. Army, Castle Dome Heliport outside Yuma, AZ and was subsequently moved to the Hughes Helicopter, Mesa, AZ flight test facility. In January of 1983, in light of the success of the flight tests to date, it was decided that the concept was ready for demonstration to members of the government research community, government helicopter users, and selected members of the helicopter industry. To this end, the aircraft was completely renovated, including rebuilding of HHC actuators. On May 10, 1984, a total of approximately 50 personnel from government and industry attended the technology transfer and user demonstration with nine of these persons flying in the aircraft to evaluate the HHC system. Results to date indicate that the HHC system is very effective in reducing 4-per-rev vibration levels. Illustrative results shown on the figure indicate that some system-on vibration levels are only about 10 percent of baseline values (system off) for both level and maneuvering flight conditions.

Future Plans - The planned research flight program, including testing in accelerating maneuvers, will be completed and documented.

Figure 15(a).

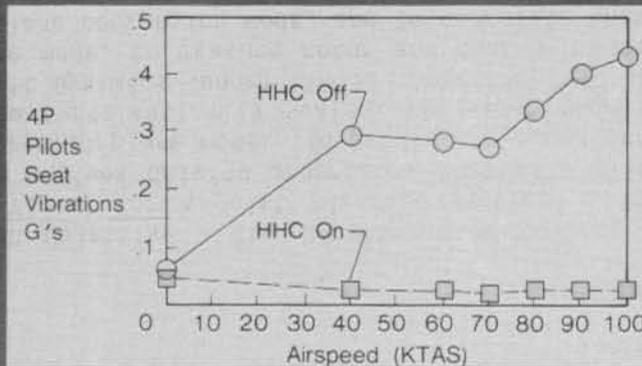
# FLIGHT TEST DEMONSTRATION OF HIGHER HARMONIC CONTROL (HHC) CONCEPT

May 1984

- One Day Briefing
- Current Closed-Loop Software Demonstrated
- About Fifty Government and Industry Attended
- Nine Fly in Aircraft



Level Flight  
Vertical Vibrations



Tri-axial Vibrations  
30° Turn, 80 Knots

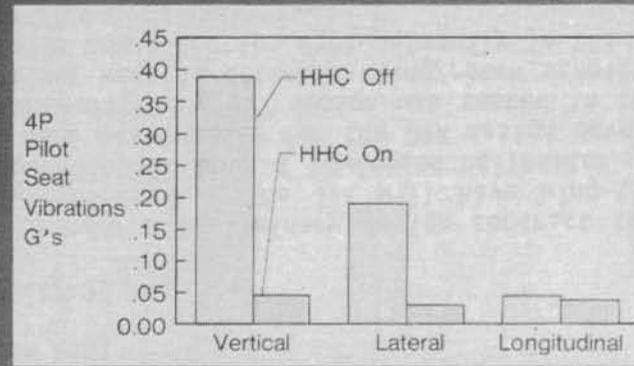


Figure 15(b).

## INITIAL TDT TEST PROVIDES ESSENTIAL DATA BASE FOR JVX PRELIMINARY DESIGN DEVELOPMENT

William T. Yeager, Jr., Charles L. Ruhlin, and Robert G. Cramer, Jr.  
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RTOP 505-42-23

Research Objective - The Department of Defense has awarded a preliminary design contract to Bell and Boeing/Vertol for a Joint Advanced Vertical Lift (JVX) aircraft. The JVX will have wing-tip mounted tilting engines driving proprotors that will allow operation in both a low-speed helicopter mode and a high-speed airplane mode. To provide an experimental data base needed for the JVX design development, a 0.2-size aeroelastically scaled, semi-span model of a preliminary JVX design was tested in the Langley Transonic Dynamics Tunnel (TDT). Specific test objectives were to determine wing/rotor stability in the airplane mode, to measure rotor and control system loads and vibration data primarily in the helicopter to airplane conversion mode, and to correlate these results with analysis.

Approach - The test was conducted by a team of NASA-Army-Bell-Boeing/Vertol personnel for a period of eight weeks (Feb.-Apr. 1984). The model as shown in figure 16(b) consists of a scaled, cantilevered wing and pylon/rotor system that could be operated with the rotor either powered or windmilling. During the test, the pylon and rotor could be remotely tilted from the helicopter mode to the airplane mode. The model was tested in both air (low Mach number operation) and freon (high Mach number operation) at densities corresponding to altitudes from sea level to 15,000 feet. Model parameters tested included: 1) pylon to wing locking (on and off downstop), 2) rotor RPM, 3) wing aerodynamics, 4) wing spar stiffness, 5) rotor pitch-flap coupling, and 6) rotor control system stiffness. Sub-critical damping data were obtained by exciting the model in the wing beam, chord, and torsion modes. The system damping was then extracted from the model response to this excitation.

Accomplishment Description - Wing beam mode instabilities were critical (lowest flutter speed) for most model configurations tested although some wing chord mode instability data were also obtained. Rotor and fixed system loads were measured for the helicopter to airplane conversion corridor. The measured damping results were compared with pre-test predictions made by Bell and Boeing/Vertol using two analyses, DYN4, developed in-house by Bell and CAMRAD, developed at NASA-Ames by W. Johnson and used by Boeing/Vertol. Because the experimental aeroelastic instabilities occurred at scaled speeds considerably below those predicted by either analysis, the major impact of this test was to cause a detailed re-examination of, and changes to, the inputs and degrees of freedom used in both DYN4 and CAMRAD. The figure shows both the pre-test and post-test analytical results from CAMRAD compared to test data for an airplane configuration with windmilling rotor at both 85% and 100% of scaled operational rotor RPM. The agreement between the current post-test analytical results and the test data is considered satisfactory.

Future Plans - A second series of tests were conducted in June/July. The existing model was modified to be more representative of the latest JVX design. The objectives of this second series of tests are similar to those of the first, and were met. The results are being analyzed.

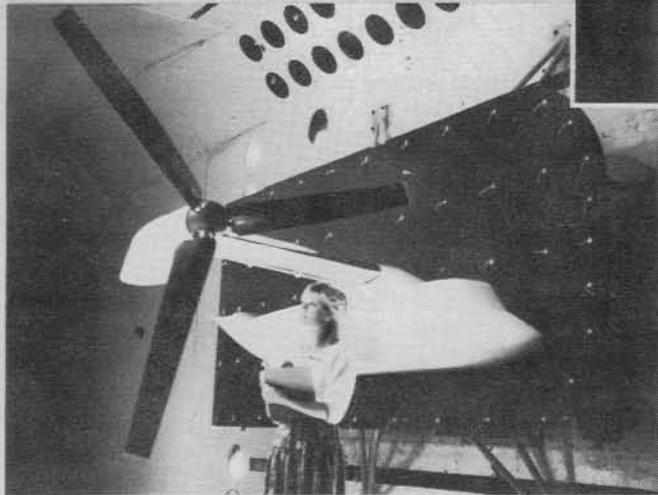
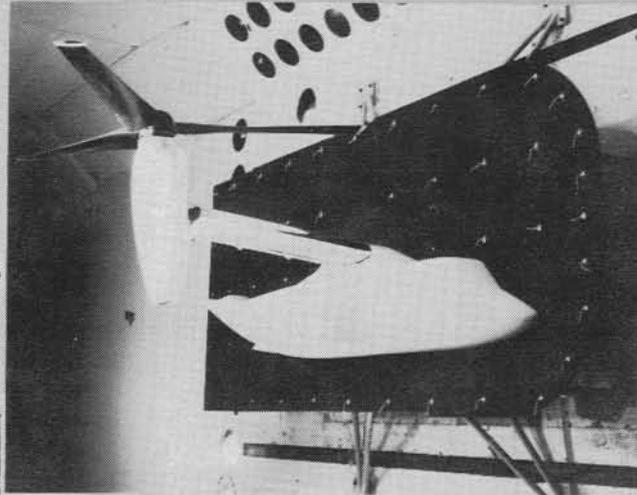
Figure 16(a).

# INITIAL TDT TEST PROVIDES ESSENTIAL DATA BASE FOR JVX PRELIMINARY DESIGN DEVELOPMENT

JVX

Helicopter  
Low Speeds

Airplane  
High Speeds



Airplane Mode Instabilities

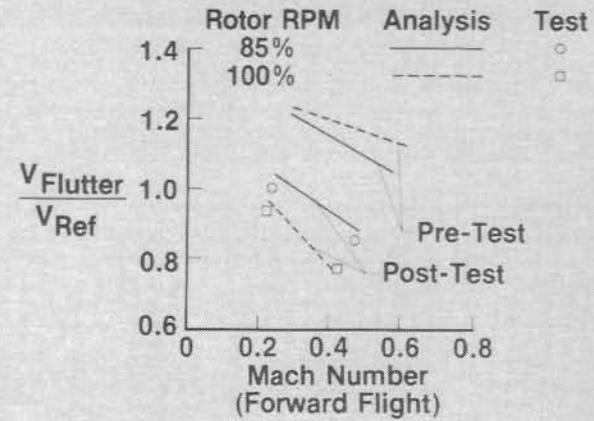


Figure 16(b).



## LANGLEY ANALYSIS PREDICTS MEASURED STABILITY CHARACTERISTICS OF JVX MODEL

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Extension 2661

RTOP 532-06-13

Objective - The JVX is the multi-mission, multi-service tiltrotor aircraft which is currently under preliminary design by a Bell-Boeing Vertol team. To provide the data base needed for the JVX design development, a .2-size, aeroelastically scaled, semi-span model of first a preliminary design and then an updated version were tested in two separate entries into the Langley Transonic Dynamics Tunnel (TDT). A key objective was to determine aeroelastic stability of the rotor/wing system in the high-speed airplane mode of flight (figure 17(c)) and to correlate these results with analysis. During the first test, the instabilities occurred at airspeeds considerably below those predicted by either contractor each using a different stability analysis code. Because no immediate explanation could be found for these discrepancies, an independent analysis by Langley using a different stability analysis code was performed to provide insight into the lack of correlation.

Approach - The Langley studies were carried out using PASTA (Proprotor Aeroelastic Stability Analysis) which is a modified version of a computer program originally developed here about 14 years ago to support tiltrotor tests in the TDT. The equations underlying the PASTA code are linear and are based on a rather simple mathematical model of a rotor having a gimbaled hub. The rotor is assumed to be in axial flow (airplane mode) and to be windmilling (nonthrusting). Quasi-steady strip-theory aerodynamics are employed for the blade airloading; the wing aerodynamic loading is assumed to be zero. A modal representation is employed for the wing structure. Calculations were made for a wide variety of configurations of the gimbaled-hub version of the model. Parameters varied included pylon-to-wing locking (on and off downstop), rotor RPM, blade pitch-flap coupling, hub flapping restraint, and wing and blade stiffness distributions. In addition, PASTA was used to analyze several configurations of the model with an updated hub design which had offset flapping hinges in addition to the gimbal.

Figure 17(a).

## LANGLEY ANALYSIS PREDICTS MEASURED STABILITY CHARACTERISTICS OF JVX MODEL

(Continued)

Accomplishment Description - Illustrative results are presented in the figure as the variations with airspeed of calculated and measured damping and frequency of the three lowest wing modes (beam, chord, and torsion) which are of importance to stability of the rotor/wing system. In addition, calculated results for the blade lag mode are also shown. The wing beam mode (primarily wing vertical bending) is seen to be critical (lowest flutter speed) for the case shown. This mode was critical for most of the configurations analyzed. There was, however, one configuration which exhibited an instability in a wing chord mode (mainly wing fore-and-aft bending). The wing torsion mode was not critical for any of the configurations studied. The peaks in the calculated damping curves for the wing chord mode at about 65 knots and the wing beam mode at about 85 knots are due to coupling of the blade lag (inplane bending) mode first with the chord mode and then with the beam mode as velocity is increased, as can be seen by inspection of the plot showing the variation of the modal frequencies with airspeed. The degree of correlation shown here is typical of that obtained for all other configurations of the model with the gimbaled hub. Further, using a physically-based approximation as an expedient, the PASTA analysis, although not directly applicable to the model with the updated hub, was also applied to those configurations with surprisingly good results. A rather intensive dialog was maintained between Langley and the contractors during these studies which has led to a better understanding of why the contractor analyses did not better predict the experimental results.

Future Plans: Results will be documented in a NASA report.

# LANGLEY ANALYSIS PREDICTS MEASURED STABILITY CHARACTERISTICS OF JVX MODEL

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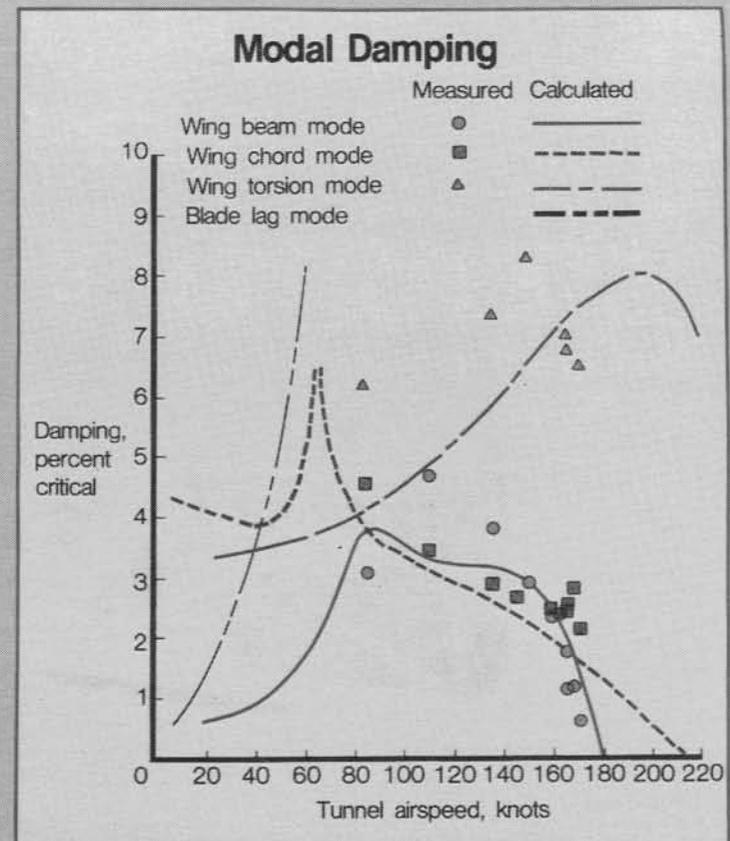
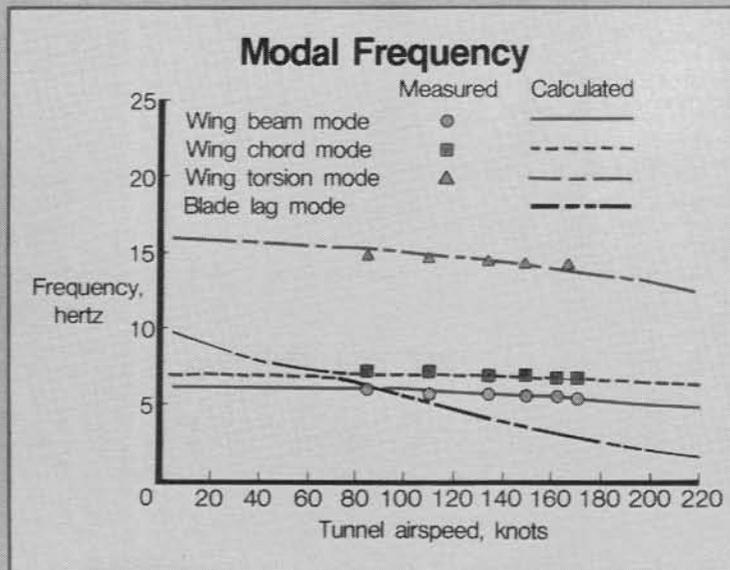


Figure 17(c).

# UNSTEADY AERODYNAMICS

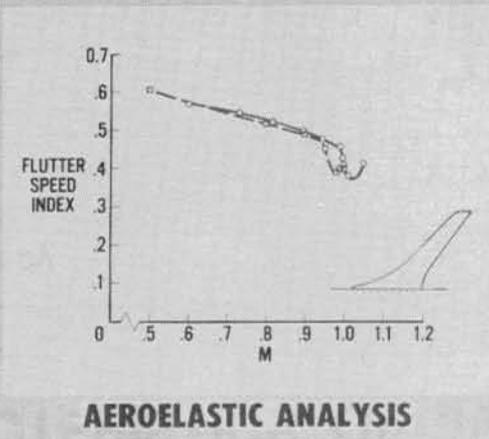
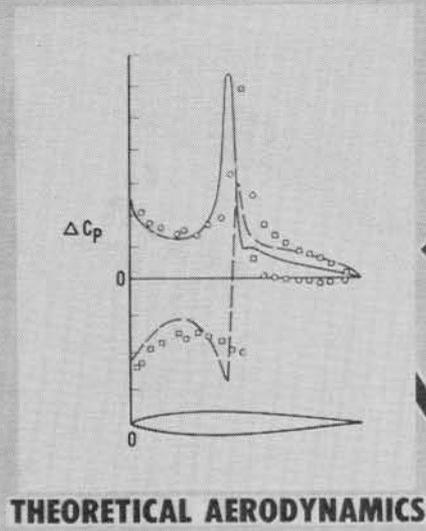


Figure 18.

UNSTEADY AERODYNAMICS  
5-YEAR PLAN

DISCIPLINE	FY 84	FY 85	FY 86	FY 87	FY 88	EXPECTED RESULTS	
THEORY DEVELOPMENT	SUB/SUPERSONIC COMPLETE CONFIG.					<ul style="list-style-type: none"> <li>- COMPLEX GEOMETRY CAPABILITY</li> <li>- BENCHMARK CALCULATIONS</li> <li>- EFFICIENT DESIGN TOOLS</li> <li>- VALIDATED AERODYNAMIC &amp; AEROELASTIC ANALYSIS</li> </ul>	
	TSP	WING/CANARD/CONTROLS/BODY					
		FULL POTENTIAL & EULER					
	VISCOUS EFFECTS				NAVIER-STOKES		
	STRONG SHOCK, SEP.						
		VORTEX FLOWS			DESIGN METHODS		
		ROTOR UNSTEADY AERO					
		NONLINEAR AEROELASTIC ANALYSIS 2-D/3-D/CONTROLS					
EXPERIMENT		2-D FLUTTER		3-D FLUTTER		<ul style="list-style-type: none"> <li>- REFERENCE PRESSURE DATA</li> </ul>	
	LASER VAPOR SCREEN						
		UNSTEADY PRESSURE MEASUREMENTS - TDT/TCT/NTF					
		DAST ARW-2/TDT			BUFFET-GUST RESPONSE		
					COMPLEX CONFIGS.		
		CANARD/WING, HI $\alpha$ , VORTEX FLOW					
		HI RN - OSCILLATING CONTROLS					
	2D CRYO/TCT		ROTOR LOADS, POSITION				

Figure 19.

GENERALIZED UNSTEADY AIRLOAD CAPABILITY ALLOWS ASSESSMENT  
OF PADE APPROXIMATIONS FOR NONHARMONIC MOTIONS

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Unsteady Aerodynamics Branch  
Extension 4236

RTOP 505-33-43

Research Objective - The objective of this research is to develop the capability to predict unsteady aerodynamic forces for arbitrary values of complex frequency, appropriate to growing and decaying oscillatory motions of aeroelastic structures. This capability is needed for more accurate evaluation of active control designs, especially for gust response and dynamic loads analyses. Also, the capability will allow accurate assessment of the commonly used Padé-approximant method of representing these air loads in nonharmonic motions.

Approach - The existing FAST computer program for flutter analysis was modified to allow complex values of the reduced frequency in the expression for the kernel function. Also, the traditional V-g equations of dynamic equilibrium were modified so that consistent eigensolutions are obtained for all speeds and not just the flutter speed as is the case in the V-g analysis.

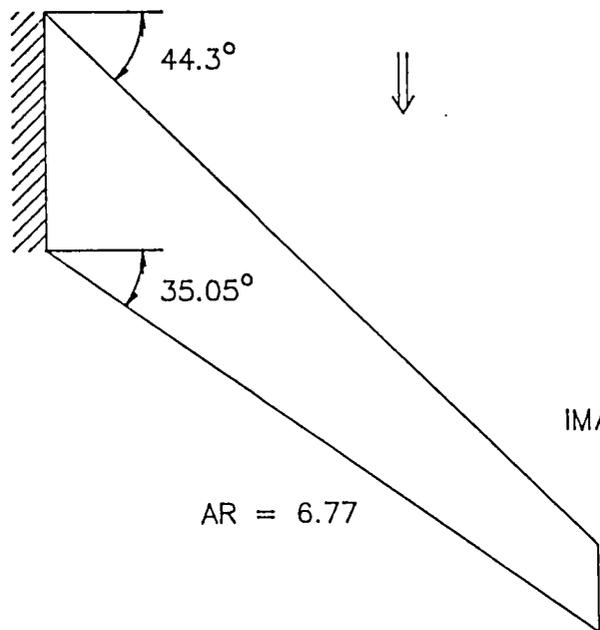
Accomplishment - Figure 20(b) shows at the right a comparison of the first-torsion-mode complex generalized force for the example planform at  $M = 0.8$ , and for reduced frequency  $k$  in the range 0 to 1. Excellent agreement of the Padé approximation with the exact (kernel function) results is shown for harmonic motion ( $\zeta = 0$ ), and for growing motion ( $\zeta$  negative). The agreement is almost as good for motion decaying with  $\zeta = 0.1$ , but for the rapid decay,  $\zeta = 0.577$ , the good agreement extends only to about  $k = 0.25$ . For higher  $k$  the Padé approximation departs from the exact curve. This comparison is from NASA TP 2292, which also features good approach-to-flutter predictions for the DAST ARW-1 flutter test vehicle.

Future Plans - The new generalized subsonic kernel function can be applied directly in analyses of gust response and of active controls for flutter suppression. Alternatively it can be used to assess the validity of aerodynamic approximations already in use for these purposes.

Figure 20(a).

GENERALIZED UNSTEADY AIR LOAD CAPABILITY ALLOWS ASSESSMENT  
 OF PADE APPROXIMATIONS FOR NONHARMONIC MOTIONS.  
 MACH = 0.8 ,  $0 \leq k \leq 1.0$  , FIRST TORSION MODE.

51



NASA

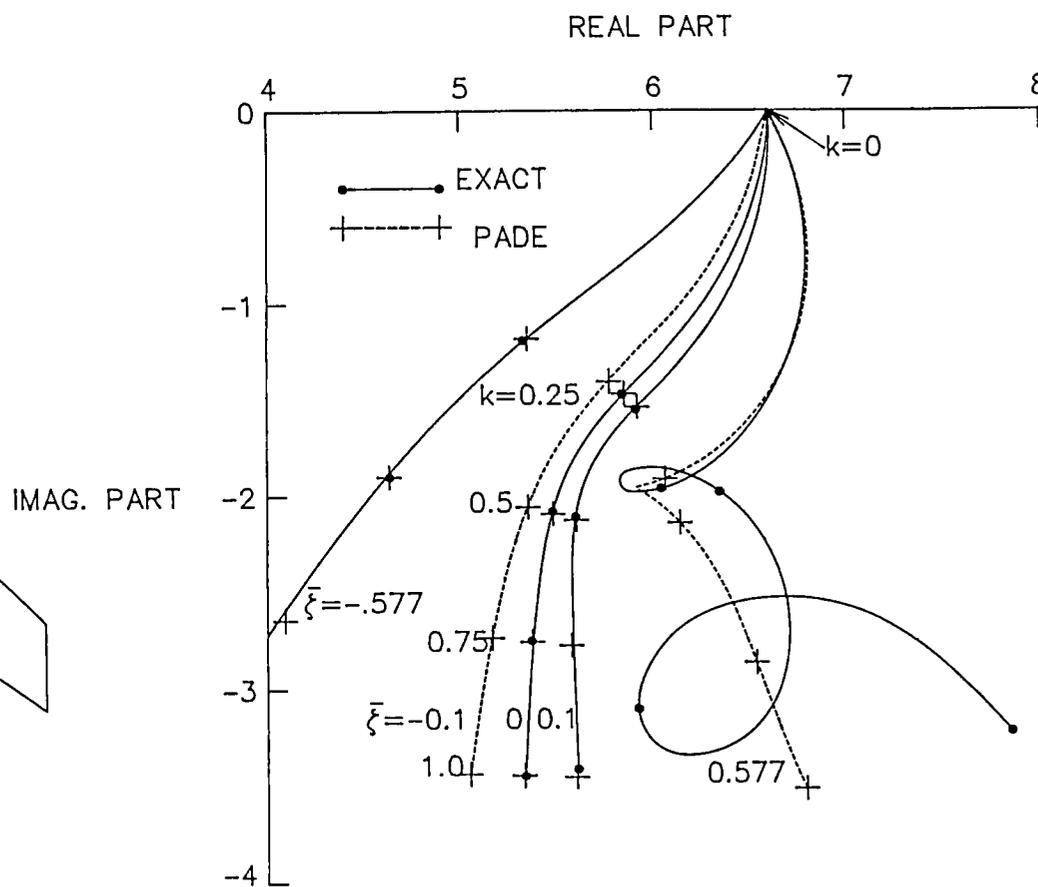


Figure 20(b).

EFFICIENT ACCURATE VISCOUS BOUNDARY LAYER MODEL COUPLED WITH XTRAN2L CODE

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Extension 4236

RTOP 505-33-43

Research Objective - The objective of this effort is to improve the accuracy of numerical predictions of unsteady transonic flow phenomena by the accurate modeling of viscous effects in unsteady finite-difference transonic computer codes.

Approach - An existing quasi-steady integral boundary layer method has been coupled with unsteady, inviscid, transonic computer code XTRAN2L developed at Langley. The coupling procedure is derived from the LTRAN2 viscous code developed by Donald Rizzetta for Ames Research Center. Several modifications to Rizzetta's procedure have been made; the most important being the ability to iterate the viscous-inviscid solutions at each time step and the inclusion of an explicit coupling technique which greatly accelerates convergence.

52 Accomplishment Description - Viscous calculations have been made for several airfoils under both steady and unsteady flow conditions. The unsteady conditions investigated include harmonic oscillations as well as transient pulses for obtaining unsteady airload frequency response functions from a single transient calculation. Figure 21(b) shows results which have been obtained for an NACA 64A010 airfoil. The overall good agreement shown between the viscous calculations and experimental results is typical for the airfoils studied.

Future Plans - The techniques will be incorporated into the unsteady 3-D transonic code XTRAN3S. The 2-D code will be systematically applied to the AGARD computational test cases.

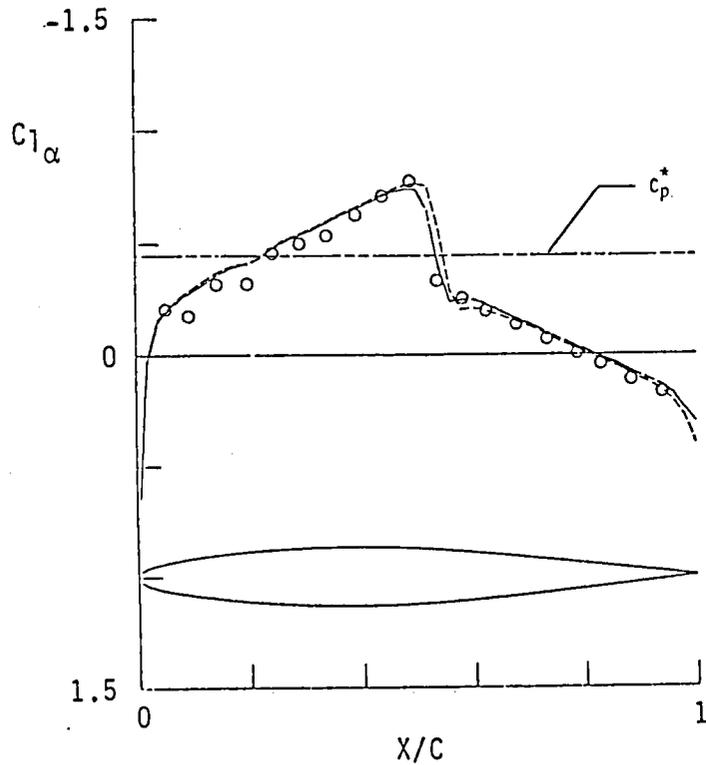
Figure 21(a).

EFFICIENT, ACCURATE VISCOUS BOUNDARY LAYER MODEL COUPLED WITH XTRAN2L CODE

NACA 64A010,  $M = 0.8$ ,  $\alpha = 0$  deg.

UPPER SURFACE STEADY PRESSURES

- - - - - INVISCID  
 ——— VISCOUS  
 O EXPERIMENT



UNSTEADY LIFT COEFFICIENT

——— REAL/INVISCID  
 - - - - - IMAG/INVISCID  
 ——— REAL/VISCOUS  
 - - - - - IMAG/VISCOUS  
 O REAL/EXPERIMENT  
 □ IMAG/EXPERIMENT

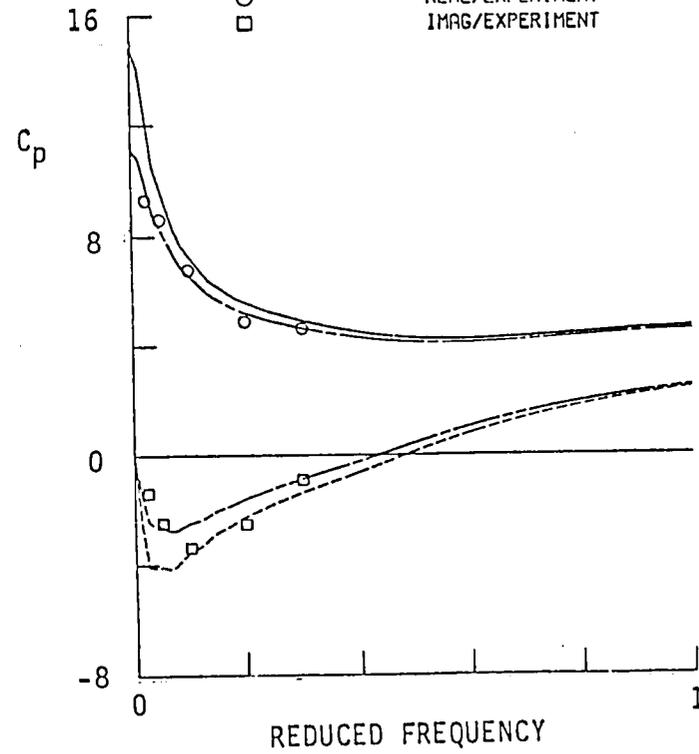


Figure 21(b).

# INVISCID TRANSONIC CODE PREDICTS DYNAMIC AIRFOIL LIFT FOR TRANSIENT RAMPING MOTION

David A. Seidel  
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Extension 4236

RTOP 505-33-43

Research Objective - The objective of this research is to assess the ability of inviscid unsteady aerodynamic computer codes to accurately predict unsteady pressures for large amplitude structural motions. Comparisons were made with experimental pressures from oscillating models.

Approach - The computer code XTRAN2L solves the two-dimensional transonic small disturbance equation. The code allows unsteady pressures and airloads to be calculated for prescribed airfoil pitching and plunging motions. For this study, calculations were made to compare with experiments made on a NACA0012 airfoil undergoing a transient ramping motion in pitch angle from 0 to 15 degrees pitch angle. At these conditions which are typical of some helicopter blade operating conditions, flow separation occurs at the higher angles.

Accomplishment Description - The cases shown in figure 22(b) are AGARD Standard Aeroelastic Configuration Computational Test Cases. They were defined and tested to examine the conditions of dynamic stall at scaled time rates similar to those of a typical helicopter application. The pitch motions are at nominally constant pitching rates with the case on the right having the faster rate or smaller nondimensional time,  $\tau_{max}$ , to achieve the maximum deflection. The figure shows the predicted and experimental lift coefficients for the two cases. For the slower pitch transient shown on the left the experimental data indicates that the flow stays attached up to a pitch angle of  $\alpha = 9$  deg. and a lift coefficient of 0.9. The faster pitch transient allows the flow to remain attached to  $\alpha = 11$  deg. and  $C_l = 1.1$ . In both cases, the inviscid XTRAN2L results accurately predict the airfoil lift coefficient up to the point of dynamic stall. This ability of an inviscid small disturbance code to provide reasonably accurate transient airloads at such large displacements had not been anticipated.

Future Plans - The XTRAN2L code has been modified to allow the simulation of an attached boundary layer coupled with the outer inviscid potential flow. This will enable more accurate calculations (particularly of the moment coefficient) for large amplitude dynamic motions up to stall onset. To allow treatment of cases including flow separation, improved viscous boundary layer modeling algorithms are being developed.

Figure 22(a).

INVISCID TRANSONIC CODE PREDICTS DYNAMIC AIRFOIL LIFT FOR TRANSIENT RAMPING MOTION

- o XTRAN2L CODE
- o NACA 0012, M = 0.6

55

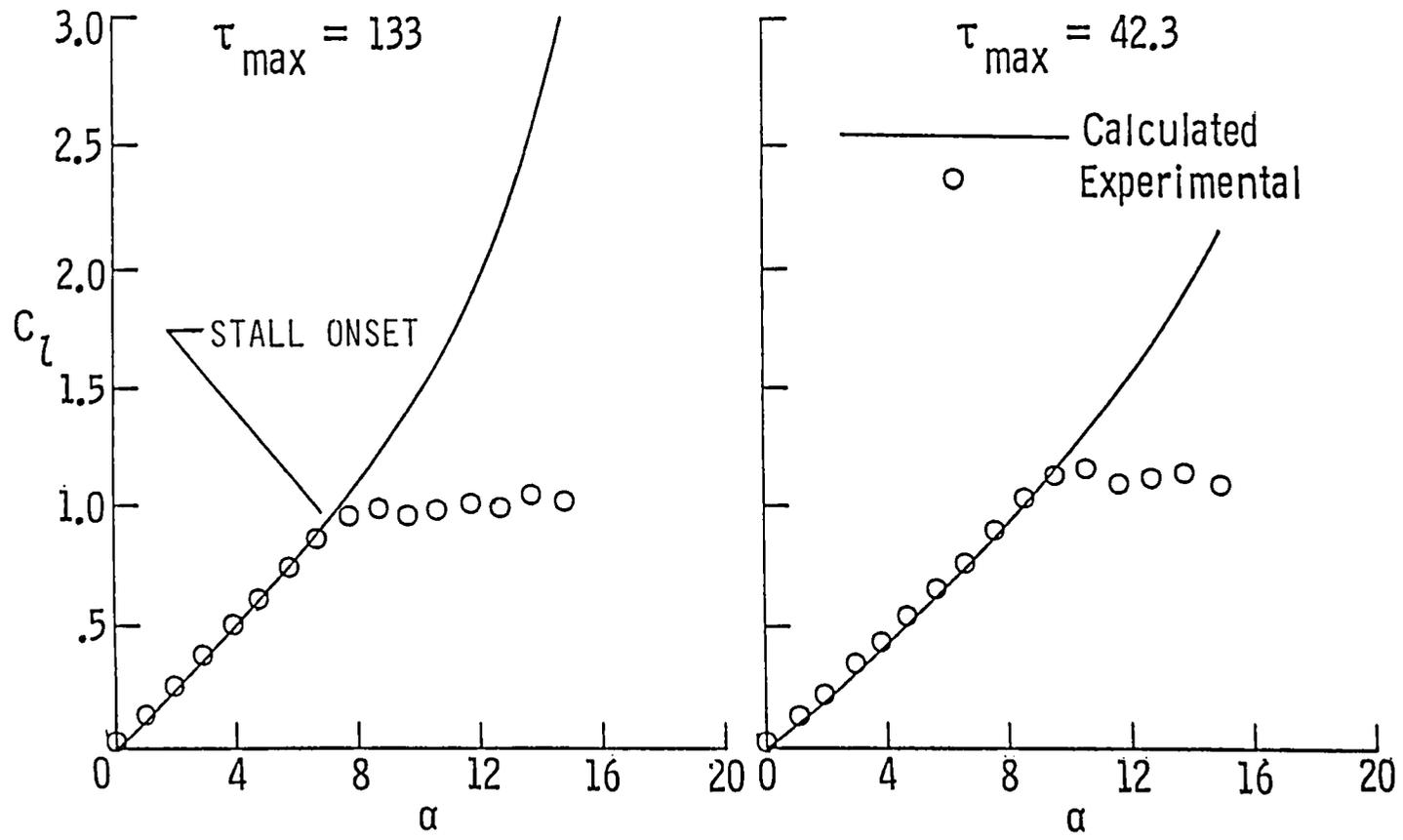


Figure 22(b).

## XTRAN2L EXTENDED TO MULTIPLE SURFACE CONFIGURATION

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RTOP 505-33-43

Research Objective - The objective of this research was to extend the XTRAN2L transonic code to allow the treatment of 2-D multiple lifting surfaces such as closely-coupled canard-wing configurations.

Approach - The two-dimensional finite-difference code XTRAN2L provides a time-marching solution to the nonlinear, small-disturbance, potential equation for transonic flow. The alternating-direction implicit (ADI) solution procedure of XTRAN2L has been extended to admit an additional lifting surface. The program is now capable of computing unsteady transonic flowfields about 2-D interfering airfoil configurations.

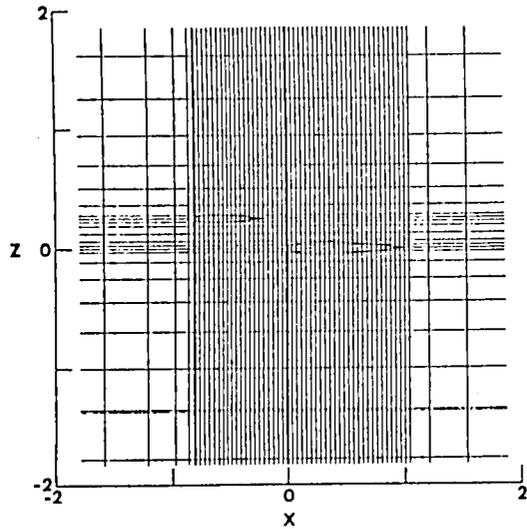
56 Accomplishment Description - To demonstrate the XTRAN2L multiple surface capability, selected steady results are presented for a closely-coupled configuration as shown in figure 23(b). In this example, the leading airfoil was placed one chordlength upstream of the trailing airfoil (measured from midchord to midchord in units of trailing airfoil chordlength) and one-quarter chordlength above. Mean angles of attack for both airfoils were  $\alpha = 1^\circ$  and the freestream Mach number was  $M = 0.5$ . The lifting pressure coefficient,  $\Delta C_p$ , is plotted for both isolated and interfering configurations. In the upper-right of the figure, comparisons of lifting pressures calculated with the XTRAN2L code and with an independent vortex lattice code are given for the case of flat plate airfoils. The good agreement shown verifies the XTRAN2L code modifications. The distance between the isolated  $\Delta C_p$  curves and the interfering  $\Delta C_p$  curves represents the aerodynamic interference between the two lifting surfaces. For the configuration shown, the leading airfoil produces a downwash on the trailing airfoil thus decreasing its  $\Delta C_p$  and lift. Conversely, the trailing airfoil induces an upwash on the leading airfoil which increases its  $\Delta C_p$  and lift. In the lower-right of the figure, XTRAN2L pressure distributions for NACA 0010 airfoils at  $M = 0.5$  further demonstrate the aerodynamic coupling between the airfoils. Mach number contour lines for this case clearly illustrate the strong interference effects.

Future Plans - This work is a preliminary effort to assess the incorporation of the multiple lifting surface capability into the three-dimensional transonic code XTRAN3S. The multiple surface computational capability will permit the assessment of interference effects on transonic unsteady airloads and flutter.

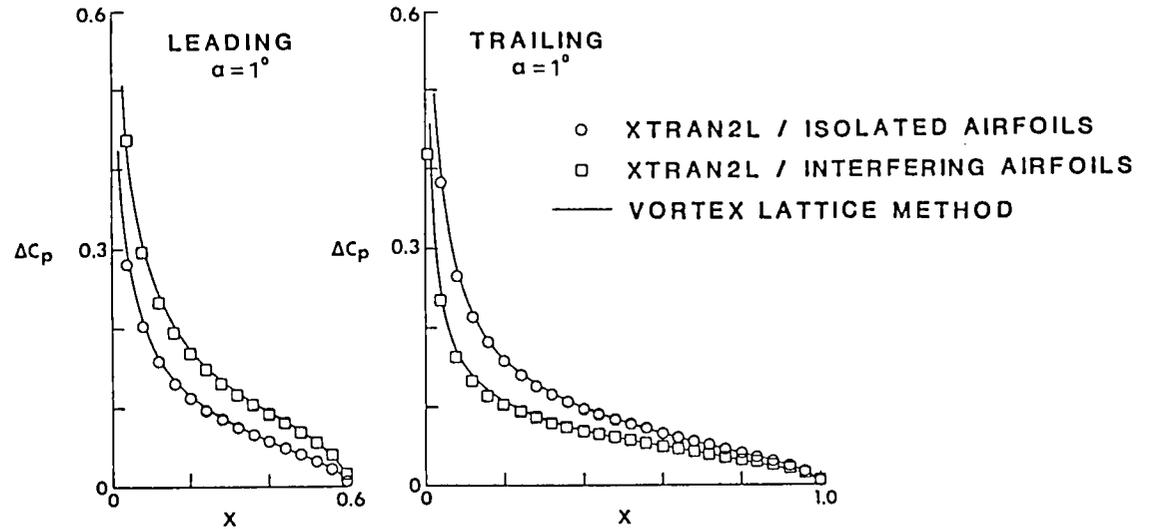
Figure 23(a).

# XTRAN2L EXTENDED TO MULTIPLE SURFACE CONFIGURATION

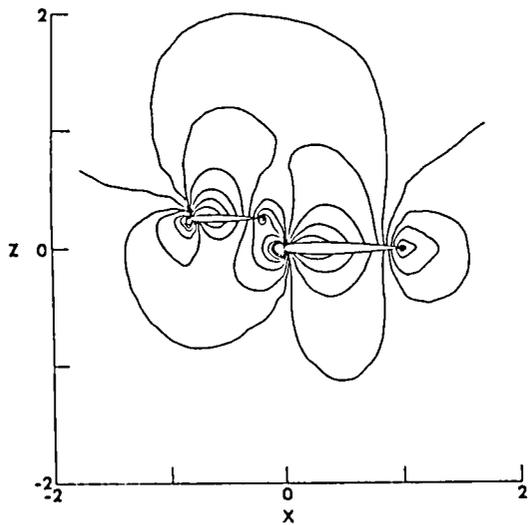
## ● GRID



## ● FLAT PLATE AIRFOILS, M = 0.5



## ● MACH CONTOURS



## ● NACA 0010 AIRFOILS, M = 0.5

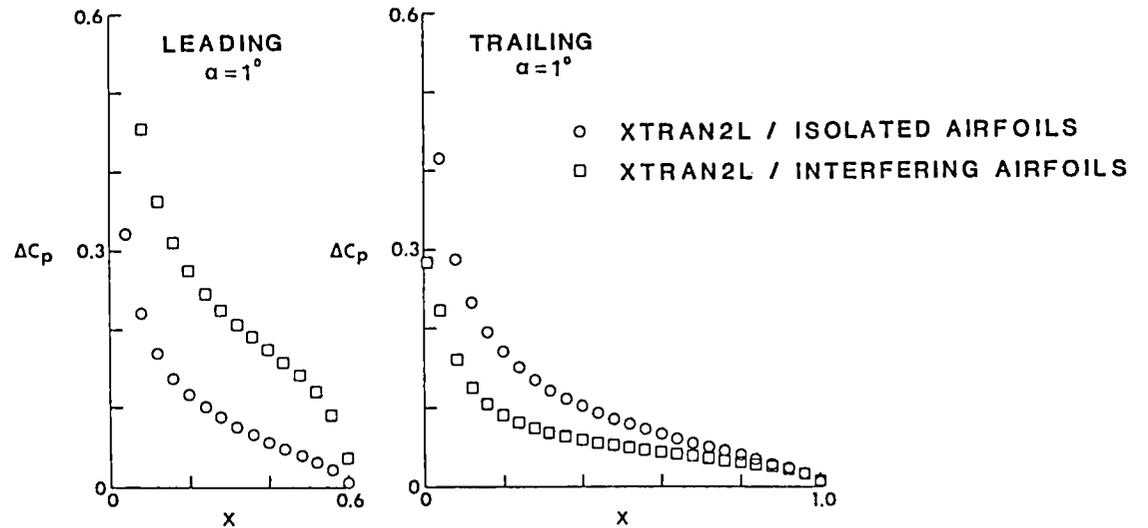


Figure 23(b).

# AIRFOIL SHAPE, THICKNESS, CAMBER, AND ANGLE OF ATTACK EFFECTS ON TRANSONIC UNSTEADY AIRLOADS

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Unsteady Aerodynamics Branch  
Extension 4236

RTOP 505-33-43

Research Objective - The objective of this study is to investigate the effects of airfoil shape, thickness, camber and angle of attack on transonic unsteady airloads as calculated by a finite-difference algorithm.

Approach - The two-dimensional, finite difference code XTRAN2L is used to determine the aerodynamic forces. This code provides a time-marching solution to the nonlinear, small-disturbance potential equation for transonic flow. The harmonic airloads for airfoil plunging and pitching motions are determined using the pulse transient method available in XTRAN2L. Shape effects were investigated by examining the pressure distributions, shock locations, and unsteady airloads for three 10% thick airfoils of different shape: NACA 0010, NACA 64A010, and parabolic arc. Thickness effects were determined by considering a family of three airfoils with different thickness: NACA 0008, NACA 0010, and NACA 0012. Angle of attack and camber effects were studied by either including mean angle of attack or by adding a simple parabolic camber distribution to the originally symmetric airfoils.

Accomplishment Description - Figure 24(b) shows the effects of shape and thickness on the lift coefficient due to airfoil pitching  $C_{L\alpha}$ . The results shown on the left indicate that airfoils of different shape yield transonic unsteady airloads that have similar trends with frequency even though their steady shock locations are very different. Results from the thickness study, shown on the right, indicate that these airfoils also yield transonic unsteady airloads that have similar trends with frequency. Here, the steady shock locations for the three airfoils were matched by scaling Mach number using the steady transonic similarity relation. In general, unsteady lift results for either shape or thickness effects compare better than moment results. Detailed comparisons of unsteady airloads show similarities in effects due to shape, thickness, camber, or mean angle of attack. These comparisons offer insight into how to limit the number of transonic unsteady aerodynamic calculations for changes in airfoil geometry or angle of attack, thereby reducing computer costs.

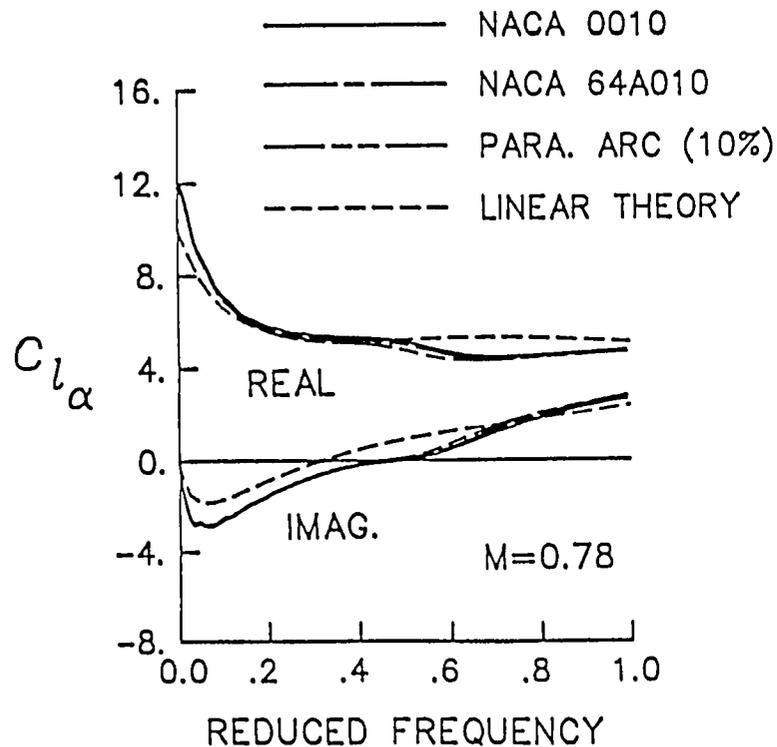
Future Plans - This work is part of a continuing effort to investigate the unsteady aerodynamic and aeroelastic behavior of airfoils in transonic flow as calculated by finite-difference algorithms.

Figure 24(a).

# AIRFOIL SHAPE AND THICKNESS EFFECTS ON TRANSONIC UNSTEADY AIRLOADS DETERMINED

## SHAPE EFFECTS

- STEADY SHOCK LOCATION NOT MATCHED
- SIMILAR TRENDS WITH FREQUENCY



## THICKNESS EFFECTS

- STEADY SHOCK LOCATION MATCHED (TRANSONIC SIMILARITY SCALING)
- SIMILAR TRENDS WITH FREQUENCY

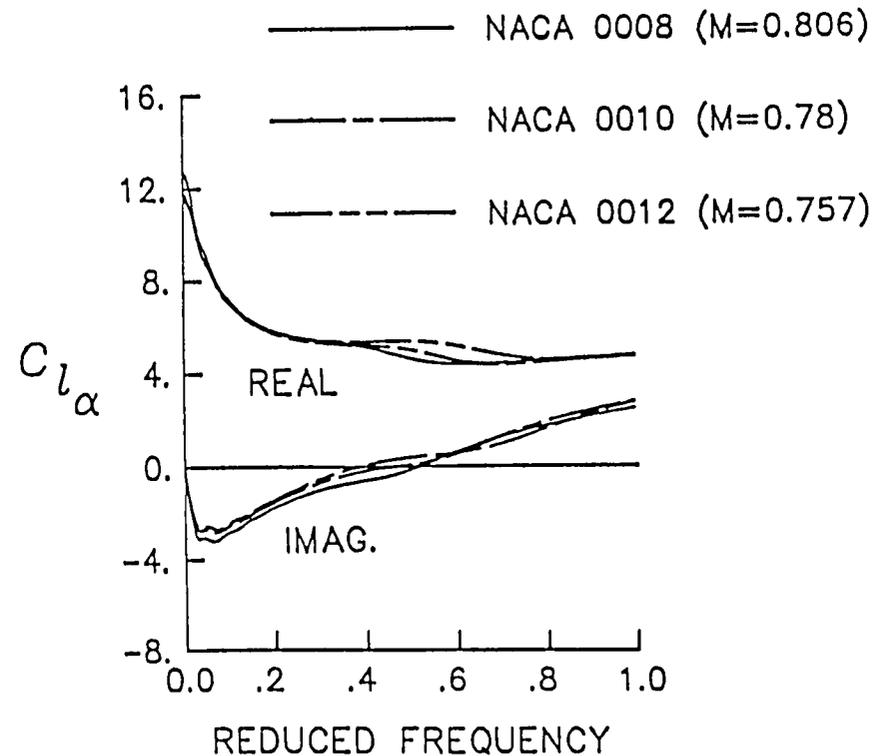


Figure 24(b).

## TRANSONIC CODE USED FOR AIRFOIL ACTIVE FLUTTER CONTROL

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Extension 4236

RTOP 505-33-43

Research Objective - The objective of this study was to demonstrate the use of a transonic code for the study of active flutter control. The active control investigated was the feedback of a velocity transducer signal,  $\xi_s$ , to a 25% chord trailing edge control surface with rotation  $\beta_c$ .

Approach - Transonic aeroelastic stability and response analyses were performed for NACA 64A006 and NACA 64A010 conventional airfoils and an MBB A-3 supercritical airfoil. Stability analyses were performed using a state-space aeroelastic model termed the Padé model. The model was formulated using a Padé approximation of the unsteady aerodynamic forces calculated by the transonic small-disturbance code LTRAN2-NLR. Time-marching response analyses were also performed by coupling the structural equations of motion to the LTRAN2-NLR code for simultaneous time-integration. A modal identification technique was applied to the time-marching response curves to estimate damping and frequency of the aeroelastic modes for verification of Padé model s-plane eigenvalues.

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Accomplishment Description - Open-loop stability and response analyses were performed to determine the behavior of the aeroelastic modes as a function of flight speed. For all of the cases considered, steady shocks were relatively weak and were located in the range of 50% to 60% chord. Padé model flight speed root-loci for the MBB A-3 airfoil at  $M = 0.765$ , for example, are shown in the left part of figure 25(b). Torsion and aileron modes are both stable; with increasing flight speed the bending dominated root-locus becomes the flutter mode. (Data to the right of the  $\sigma/\omega_\alpha = 0$  line indicates instability). Open-loop time-marching solutions shown on the right verify the Padé model predictions. Simple constant gain control laws were studied to determine the behavior of the aeroelastic modes as a function of control gain. As shown in the right part of the figure, velocity feedback  $\xi_s$  with positive control gains  $K_y$  stabilized the bending dominated flutter pole and increased torsion mode damping. In general, the Padé model eigenvalues compared well with the time-marching modal damping and frequency estimates. Therefore, locally linear aeroelastic modeling was found to be applicable to active control of 2-D airfoils in transonic flow with mild shocks. The study also represents the first application of a transonic code to active flutter control.

Future Plans - The effects of viscous modeling on transonic aeroelasticity and active control are currently being examined using the LaRC transonic code XTRAN2L.

Figure 25(a).

# TRANSONIC CODE USED FOR AIRFOIL ACTIVE FLUTTER CONTROL

- □ ◇ PADÉ MODEL SOLUTION
- ■ ◆ TIME-MARCHING SOLUTION

OPEN-LOOP FLIGHT SPEED ROOT-LOCI

VELOCITY FEEDBACK GAIN ROOT-LOCI  
AT THE OPEN-LOOP FLUTTER SPEED

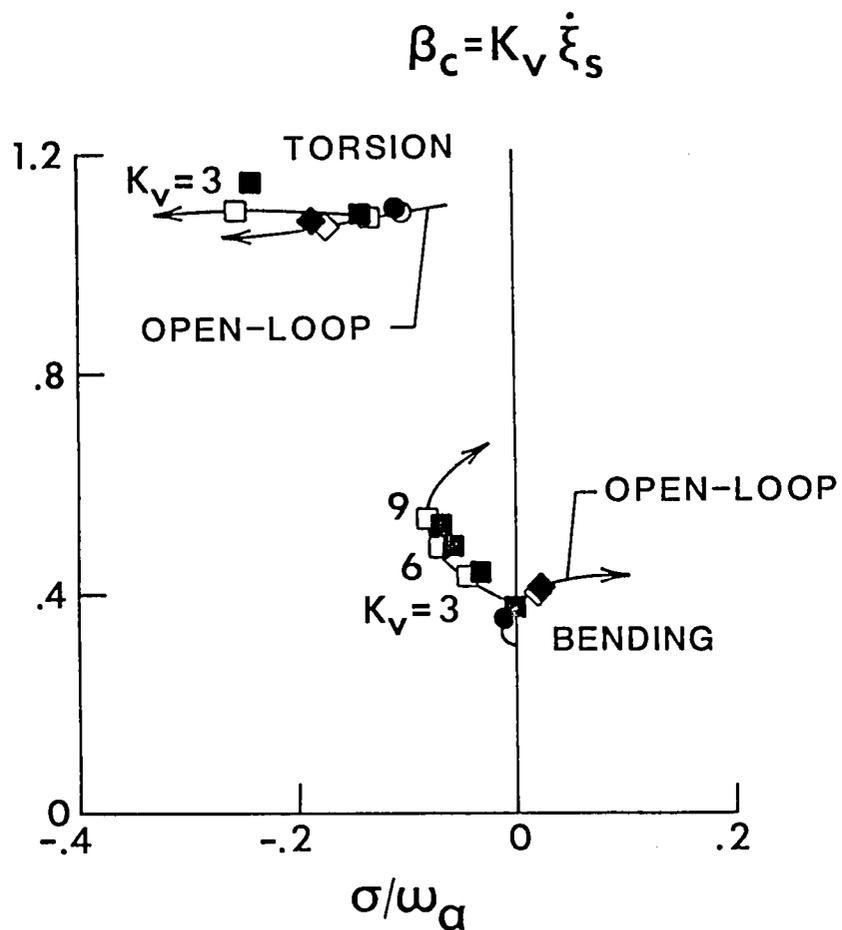
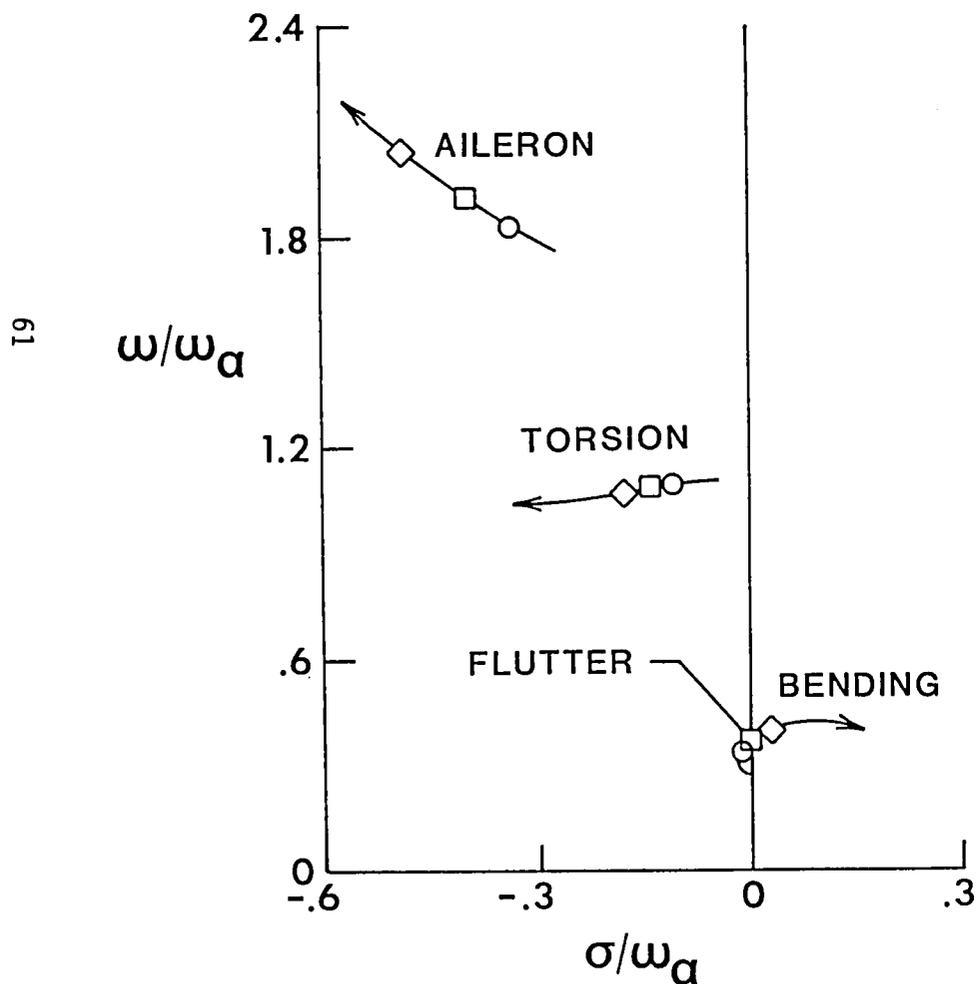


Figure 25(b).

# NON-ISENTROPIC UNSTEADY TRANSONIC SMALL DISTURBANCE THEORY

Dennis Fuglsang and Marc Williams  
Purdue University

RTOP 505-33-43

Research Objective - The objective of this effort was to correct certain deficiencies that exist in classical transonic small disturbance (TSD) potential theory. While TSD is a useful, efficient approximation to the full Euler equations of inviscid flow, the solutions obtained from it sometimes agree very poorly with the corresponding Euler solutions (usually cases involving strong shocks). Also in certain Mach number and angle of attack regions, multiple (nonunique) solutions are obtained for a given flow field. Several simple modifications to TSD theory were introduced in order to more accurately model the Euler equations and to resolve the nonuniqueness problem (apparently not present in the Euler equations).

Approach - Modifications to the TSD equation are derived from a formal asymptotic development of the Euler equations including the effects of shock generated entropy. To first order, this results in the classical TSD equation while the second order terms provide the above noted modifications. This second order theory involves; a) a new streamwise flux formulation satisfying the exact Prandtl relation for shock jump conditions, b) the inclusion of shock generated entropy in the pressure coefficient evaluation and c) the convection of that entropy in the wake boundary condition. The modifications are implemented in the two dimensional finite difference code XTRAN2L.

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Accomplishment Description - Steady and unsteady calculations were made for the NACA0012 airfoil using both the modified and unmodified small disturbance codes. The calculations indicate that the modifications resolve the nonuniqueness problem in static and dynamic cases, with the solutions giving good agreement with Euler code results. The modified flux and pressure coefficient evaluation place the shock in the correct position (within one grid cell of the Euler solution) with the correct strength, in terms of velocity (Prandtl relation is captured by the numerical scheme) and pressure. The convection of entropy in the wake constrains the shock motion to realistic locations (agreeing well with Euler results) for lifting flow cases while consistently modeling the entropy jump in the flow. Figure 26(b) clearly shows the effects of the modified theory for the steady pressure distribution for a case in the middle of the nonuniqueness region. Similar agreement with the Euler code is obtained for cases outside the nonuniqueness region for the NACA0012 airfoil, and for NACA64A010 and NLR7301 airfoil cases. The modified theory requires only minor coding changes in existing small disturbance algorithms and entails little increase in computational cost.

Future Plans - The modified second order theory will be extended to an existing three dimensional transonic small disturbance code to enable unsteady aerodynamic calculations on wings in strong transonic flows. Also, it is anticipated that the second order theory, when coupled with a viscous boundary layer capability, will improve agreement with experimental unsteady pressure distributions.

Figure 26(a).

# NON-ISENTROPIC SMALL DISTURBANCE THEORY RESOLVES NONUNIQUENESS PROBLEMS

- o MODIFIED XTRAN2L CODE
- o NACA 0012 AIRFOIL
- o  $M = 0.84$ ,  $\alpha = 0.25$  deg.

UPPER SURFACE

LOWER SURFACE

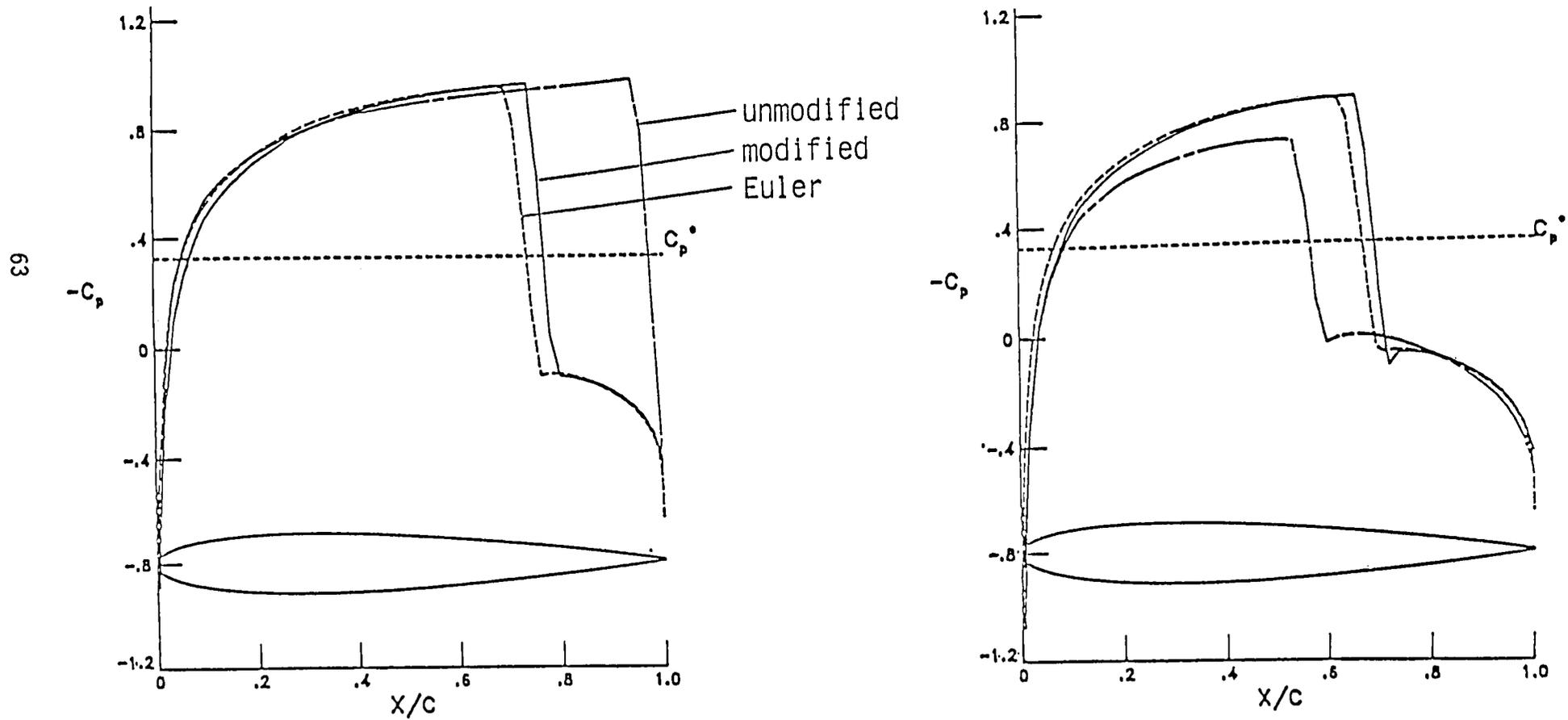


Figure 26(b).

# NONREFLECTING FAR-FIELD BOUNDARY CONDITIONS IMPROVE UNSTEADY AIRLOAD CALCULATIONS

Woodrow Whitlow, Jr.  
Unsteady Aerodynamics Branch  
Extension 4236

RTOP 505-33-43

Research Objective - The objective of this research effort is to increase the accuracy and efficiency of finite difference methods for calculating transonic unsteady aerodynamic loads on wings.

Approach - The original version of the XTRAN3S computer program for three dimensional transonic small perturbation unsteady aerodynamic analysis incorporated steady state conditions at the boundaries of the computational domain. This caused disturbances incident on the boundaries to be reflected back into the computational region. The reflected disturbances can cause errors in the calculated unsteady aerodynamic loading. In this research effort, nonreflecting (characteristic) far-field boundary conditions, which simulate outgoing waves at the boundaries, were developed. These conditions absorb most of the disturbances incident on the boundaries.

Accomplishment Description - The boundary conditions were implemented in XTRAN3S and tested by calculating the unsteady forces on a flat plate rectangular wing with a pulse in root angle of attack  $\alpha$ . The lift response  $C_l$  and frequency response function for the unsteady lift curve slope  $C_{l\alpha}$  were calculated with and without the nonreflecting boundary conditions. The calculations were made for a free stream Mach number,  $M$ , of 0.85. The lift response, which should return smoothly to zero after an initial transient, is at the left of figure 27(b). Use of the steady state far-field conditions results in oscillations in calculated lift well after the initial transient. When the nonreflecting conditions are used, the lift returns smoothly to zero. In the frequency response function, shown at the right of the figure, the XTRAN3S results are compared with those obtained with an exact kernel function method. Using the original far-field conditions causes spurious oscillations in the frequency response. When the new conditions are used, the oscillations are eliminated and good agreement with the kernel function method is obtained.

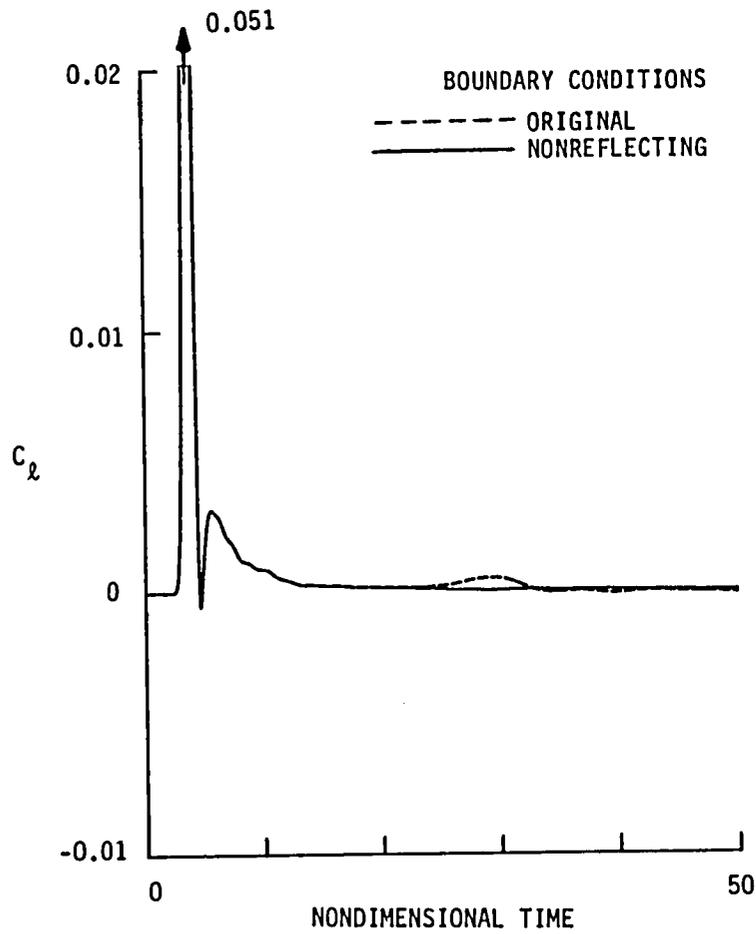
Future Plans - Future plans include systematic studies to determine how close the far-field boundaries may be placed with no loss in accuracy of the computed results. This could lead to fewer required grid points and thus a more efficient computational method.

Figure 27(a).

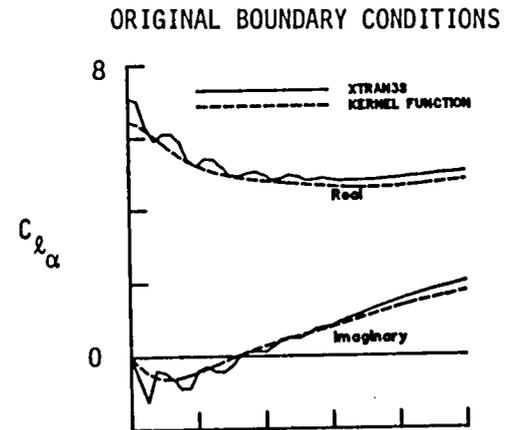
# NONREFLECTING FAR-FIELD BOUNDARY CONDITIONS IMPROVE UNSTEADY AIRLOAD CALCULATIONS

RECTANGULAR FLAT PLATE WING,  $M = 0.85$

## LIFT RESPONSE



## FREQUENCY RESPONSE



## NONREFLECTING BOUNDARY CONDITIONS

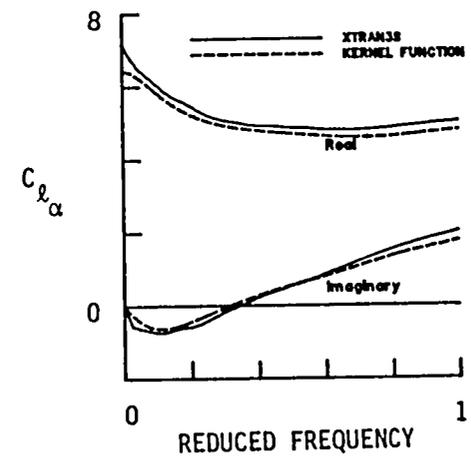


Figure 27(b).

# LIFTING SURFACE THEORY FOR A HELICOPTER ROTOR IN FORWARD FLIGHT

Hsiang Tai and Harry L. Runyan  
Unsteady Aerodynamics Branch  
Extension 4236

RTOP 505-33-43

Research Objective - To develop a three dimensional compressible, time dependent lifting surface theory for a helicopter rotor in forward flight.

Approach - The concept of the acceleration potential is followed in the present work. This technique involves the solution of the integral equation

$$w = \int_A LK \, dA$$

where A is the area of the rotor and w is the time dependent downwash which is made equal to the surface motion in such a manner that the surface is essentially impenetrable. The aerodynamic loading, L, is the unknown and represents the desired pressure distribution on the rotor. The kernel, K, represents the downwash produced by a pressure doublet of unit strength on the rotor surface. The kernel is a complex function and involves an intergration in the time domain. The basic formulation is based on an inertial coordinate system as illustrated at the top of figure 28(b).

Accomplishment Description - The integral equation has been successfully solved, including the evaluation of a high order singularity. The computer program is now operational and includes the following features (1) compressibility (2) multiblade capability (3) time dependence for a vibrating rotor. Examples of the results are shown in the attached figure, where the difference in the thrust coefficient is given for the compressible and incompressible cases. The thrust coefficient for a one bladed rotor and a two bladed rotor is also shown. Note the large interference effect shown by the two bladed case from  $\psi = 250^\circ$  to  $\psi = 60^\circ$ .

Future Plans - Plans for the following year include: 1) conduct systematic parametric studies 2) correlation with experiment and other analytical techniques 3) extension of the multibladed case to include more chordwise sections 4) calculation the whole velocity flow field.

Figure 28(a).

UNSTEADY LIFTING SURFACE THEORY FOR ROTORS PREDICTS DETAILED BLADE LOADS

$$\mu = 0.17, \theta_B = 0.1, \alpha_R = 0.05, M_{tip} = 0.54$$

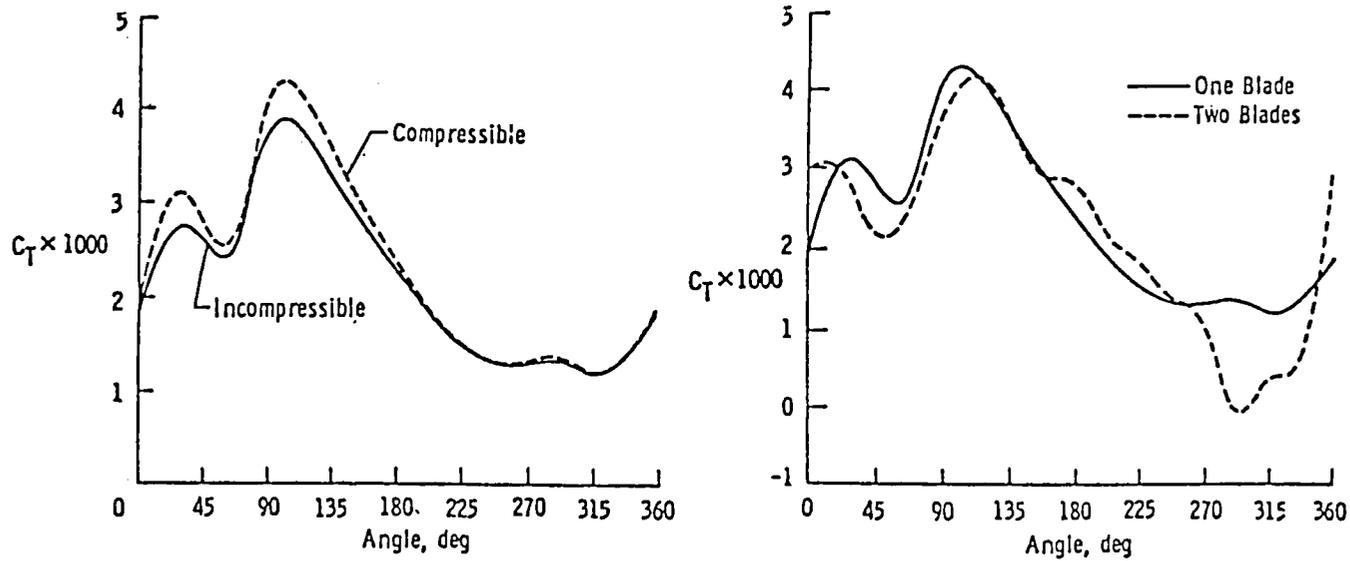
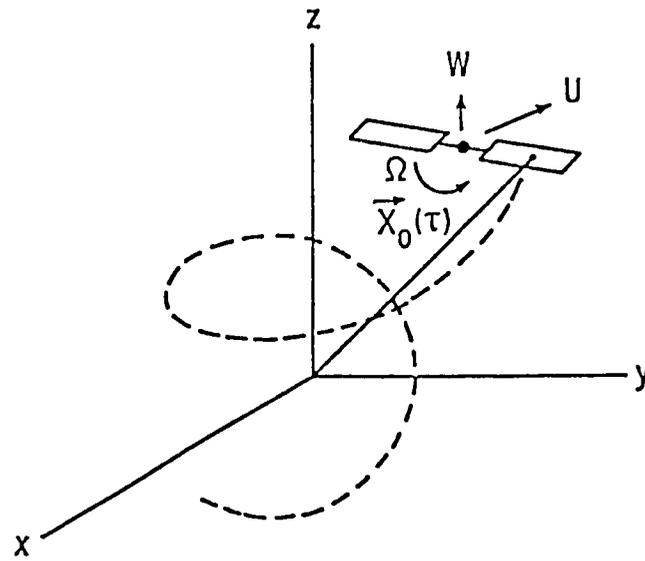


Figure 28(b).

## REYNOLDS NUMBER EFFECTS ON UNSTEADY PRESSURE STUDIED IN 0.3M CRYO TUNNEL

Robert W. Hess and David A. Seidel  
Unsteady Aerodynamics Branch  
Extension 4236

RTOP 505-33-43

Research Objective - This effort had two objectives. The first was to develop and test experimental techniques for measuring unsteady pressures in a cryogenic environment. The second objective was to measure the effects of Reynolds number on the unsteady pressure distribution of an oscillating, two dimensional, supercritical, airfoil with free transition.

Approach - The airfoil tested, SC(2)0714, is a well documented 14 percent thick supercritical airfoil. A series of preliminary investigations were conducted to develop and test the instrumentation system. Transducer installation integrity was established for the final configuration by repeated cryo cycling of a simulated transducer subject to a realistic pressure differential. The oscillating drive system was designed to accomodate thermal expansion and contraction experienced by the tunnel test section during operation (stagnation temperature ranged from 120 to 340 deg. K and pressures from 1 to 6 atm.).

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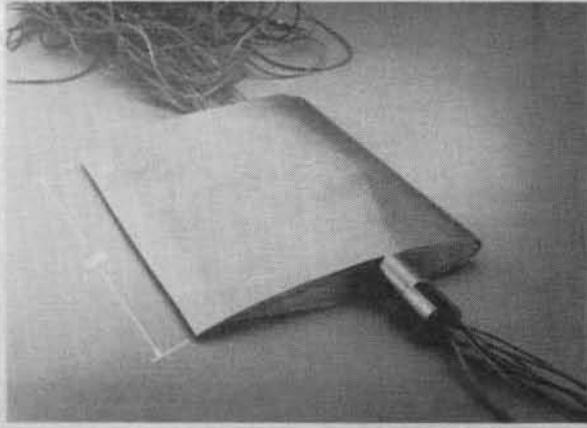
Accomplishment Description - The test conditions are shown in figure 29(b). Measurements were made over a range of Reynolds numbers, based on a chord length of 6 inch., from 6 million to 35 million. Mach number was varied at Reynolds numbers,  $R_n$ , of 15 and 30 million. For each data set, static pressure measurements were made from -2.5 to 2.5 deg. in 0.5 deg. increments. Unsteady pressure measurements were taken at mean agles of attack of from -2 to +2 degrees in 1 deg. increments. For those tunnel conditions where both frequency and amplitude were varied, frequency was varied from 5 Hz to a maximum of 60Hz and amplitude was varied from 0.25 deg. to a maximum of 1 deg. The uncorrected static pressure distributions shown in the figure indicate laminar flow and a weak shock at  $R_n$  of 6 million and a stronger shock in a turbulent boundary layer at a  $R_n$  of 30 million. Unsteady pressure measurements were made at static angles of attack to obtain data which will be used to locate transition. Steady and unsteady pressures were surveyed in the wake and static tunnel floor and wall pressure measurements were made for flow correction calculations.

Future Plans - The data is being reduced and will be used to evaluate viscous and inviscid coupling procedures for unsteady B.L. codes.

Figure 29(a).

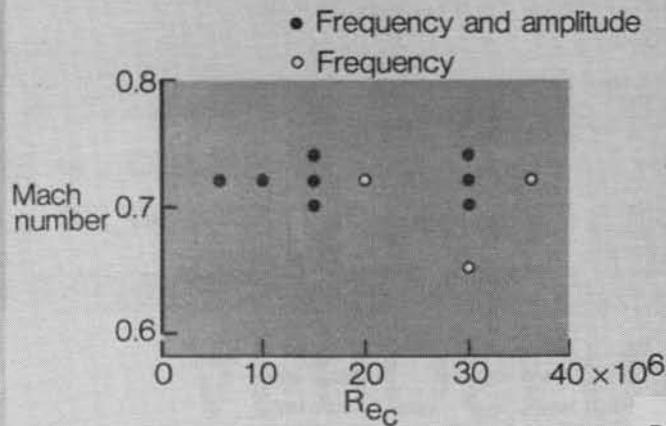
## REYNOLDS NUMBER EFFECTS ON UNSTEADY PRESSURE STUDIED IN 0.3 M CRYO TUNNEL

14 percent supercritical airfoil



- Experimental techniques developed to measure unsteady pressures in a cryo environment
- 1st oscillating airfoil pressures measured in cryo tunnel at High Reynolds Number

### Test conditions



### Steady pressure distributions $M=0.72, \alpha=1.5$ deg

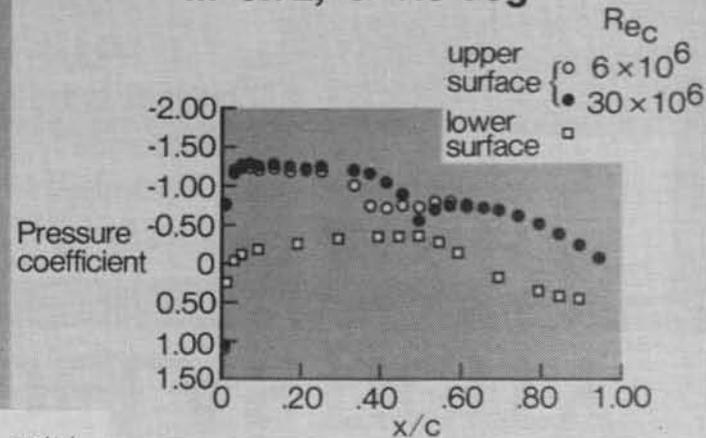


Figure 29(b).

# AEROSERVOELASTICITY

- Classical
- Optimal

Synthesis Methods

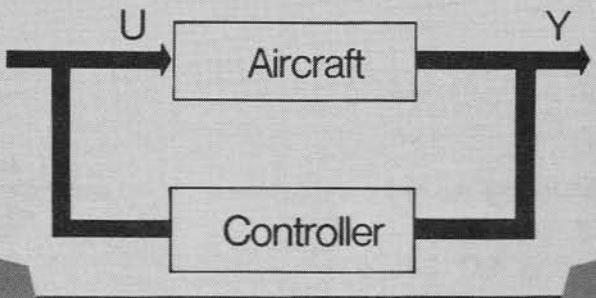
Analysis Methods

- Singular Values
- Stability

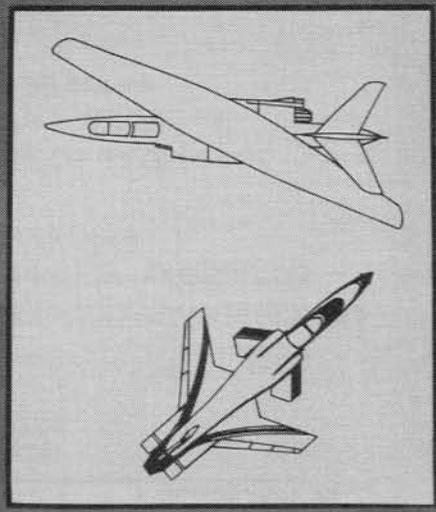
Validation



Methodology For Aeroservoelastic Interactions



Applications



Active Control Technology

Figure 30.

AEROSERVOELASTICITY  
5-YEAR PLAN

DISCIPLINE	FY-84	FY-85	FY-86	FY-87	FY-88	EXPECTED RESULTS
ANALYSIS METHODS	EMPIRICAL CORRECTIONS		STATIC AEROELASTICITY LINEAR METHODS		STATIC AEROELASTICITY NONLINEAR METHODS	NEW AND IMPROVED AEROSERVOELASTIC METHODS FOR ANALYSIS AND SYNTHESIS TO ALLOW ACTIVE CONTROLS INTEGRATION INTO AIRCRAFT OPTIMIZATION
	OPTIMAL SENSITIVITY ANALYSIS					
	CONTROL LAWS SYNTHESIS METHODS	SIMULTANEOUS STRUCTURAL/CONTROL DESIGN (STATIC)			STATIC & DYNAMIC	
SIMULTANEOUS STRUCTURAL/CONTROL DESIGN (DYNAMIC)						
ADVANCED CONTROL LAW DESIGN						
APPLICATIONS AND VALIDATIONS	DAST ARW-2					
	AGENCY FLIGHT PROGRAMS, i.e. OBLIQUE, X-WING					

Figure 31.

## A NEW FORMULATION OF AIRPLANE DYNAMIC LOADS EQUATIONS

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Extension 3323

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Kentron International  
865-1010

RTOP 505-33-43

Research Objective - Historically, the equations of motion (EOM) and the dynamic loads equations (DLE) of flexible airplanes have been formulated in the frequency domain. Within the past decade a technique for reformulating the EOM in the time domain has become well known and widely used. The purpose of the present work is to reformulate the DLE in the time domain in a manner analogous to the EOM reformulation.

Approach - The solution of the EOM is a vector of generalized coordinates. The DLE are obtained using the summation of forces method, in which this generalized coordinator vector ( $q$ ), its second derivative ( $\ddot{q}$ ), and a disturbance quantity (such as gust velocity,  $w_g$  or control deflection,  $\delta$ ) each multiply coefficient matrices. The coefficient matrix for  $\ddot{q}$  contains inertia terms and each element of this matrix is a real constant. The coefficient matrix for  $\dot{q}$  contains aerodynamic terms due to airplane motion; for  $w_g$  or  $\delta$ , aerodynamic terms due to disturbance. Each element of these aerodynamic matrices is complex and, in addition, is a tabular function of reduced frequency,  $k$ . In the new formulation of airplane DLE, these tabular functions are each converted to an approximating rational polynomial in the Laplace variable,  $s$ . Once this conversion has been made, it is a routine step to transform the DLE (and, by an analogous manner, the EOM) into a state space formulation in the time domain. This state space formulation makes available a wide range of computational tools for performing multivariable control system analysis and synthesis tasks.

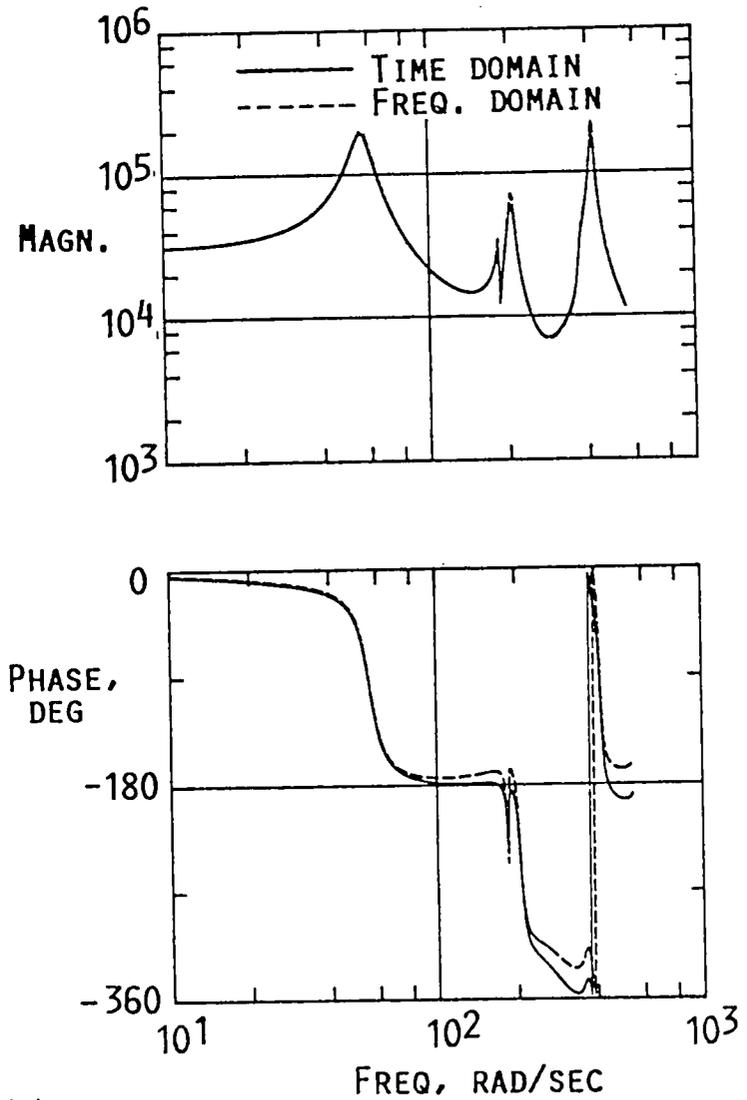
Accomplishment Description - The sketch on the left side of figure 32(b) illustrates the approximating rational polynomial and corresponding tabular values for an element of one of the aerodynamic matrices for DAST ARW-2. A comparison of dynamic load frequency response functions is presented on the right side of the figure. They both represent wing root bending moment on ARW-2 due to oscillating the outboard control surface on the wing and they both contain inertia, motion-aerodynamic, and disturbance-aerodynamic components. One frequency response function was computed using the frequency domain formulation of the DLE; the other with the time-domain reformulation. There is excellent agreement in the magnitude comparison and very good agreement in the phase angle comparison.

Future Plans - There are plans to use this new formulation in computing gust loads on DAST ARW-2. In addition, an AIAA paper and a NASA TM are proposed.

Figure 32(a).

# NEW FORMULATION OF AIRPLANE DYNAMIC LOADS EQUATIONS

## BENDING MOMENT FRF



AERO LOAD COEF.  
DUE TO  
AIRPLANE MOTION

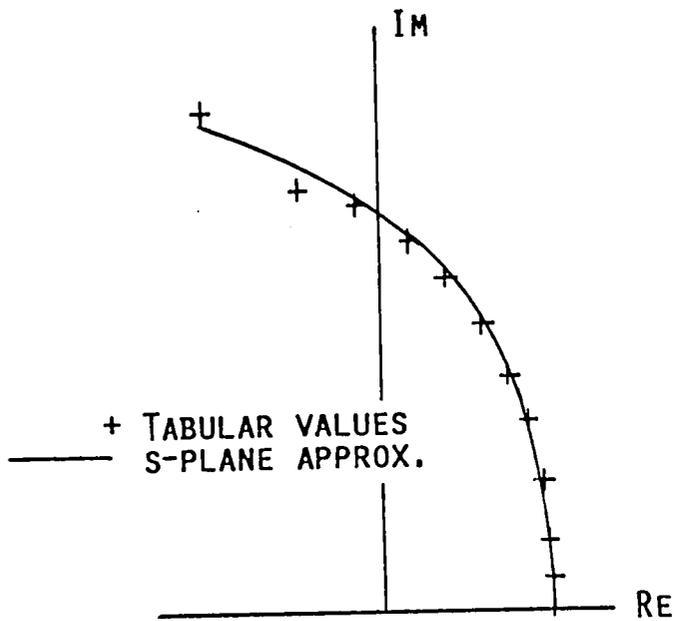


Figure 32(b).



# A METHOD TO STABILIZE LINEAR SYSTEMS USING EIGENVALUE GRADIENT INFORMATION

Carol D. Wieseman  
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Extension 4232

RTOP 505-33-43

Research Objective - Most optimal control law synthesis techniques require an initial control law that results in a stable closed-loop system. The methods that are presently used to determine such a control law are largely trial and error and can therefore be time consuming, especially for open-loop unstable systems. The objective of this research is to develop a method that is mathematically based and overcomes the difficulties associated with the trial and error approaches.

Approach - A stable system is one in which all of the real parts of the system eigenvalues are less than zero. The problem is therefore to determine the control law parameters (gains, poles, zeros, etc.) that make all the real parts of the eigenvalues negative. The method that has been developed to solve this problem employs a gradient-based optimization algorithm to determine a search direction for the control law parameters that will move all the eigenvalues into the left half of the complex plane. An important element in employing the optimization algorithm is the calculation of the gradients. The present research included the development of analytical expressions for the gradients of the eigenvalues with respect to control law design variables. Initially, a direct constrained optimization method (method of feasible directions) was used where the real parts of the eigenvalues were the constraints. Due to tracking problems of the eigenvalues and discontinuities in the eigenvalue gradients, this approach was abandoned. An indirect constrained optimization method (a penalty function approach) was adopted where the objective function (Kreisselmeier Steinhauser) was a penalty function of all the real parts of the eigenvalues. The K.S. function has the characteristic that the positive real parts of the eigenvalues dominate the value of the objective function. Figure 33(b) shows a flow chart of the overall method.

Accomplishment Description - This method was applied to the design of an active flutter suppression system for the DAST ARW-1 wind tunnel model. Values for the control law design variables which would stabilize the system were determined using this method starting at several arbitrary values of the design variables. Figure 33(c) shows the results of a numerical example. The control law was chosen to be represented by a transfer function with a third order numerator and a fourth order denominator. The gain of the control law was chosen to be the design variable. A history of the gain, the objective function and eigenvalues are shown in the figure. The method was successful in moving the unstable eigenvalue at approximately 350 rad/sec into the left-half plane in five iterations.

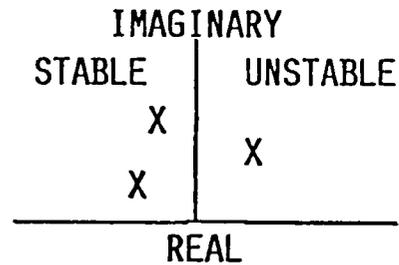
Future Plans - Future plans include deriving the remaining gradients of the eigenvalues with respect to any parameters which make up the control system or the augmented system matrices. In addition, this method will be reported in a proposed NASA TP.

Figure 33(a).

METHOD DEVELOPED TO DETERMINE A CONTROL LAW  
 THAT STABILIZES AN UNSTABLE SYSTEM

NEW METHOD: EMPLOY OPTIMIZATION

GIVEN:



PROBLEM: FIND CONTROL LAW PARAMETERS  
 THAT MAKE ALL REAL PARTS OF  
 EIGENVALUES NEGATIVE

PAST METHODS: TRIAL AND ERROR

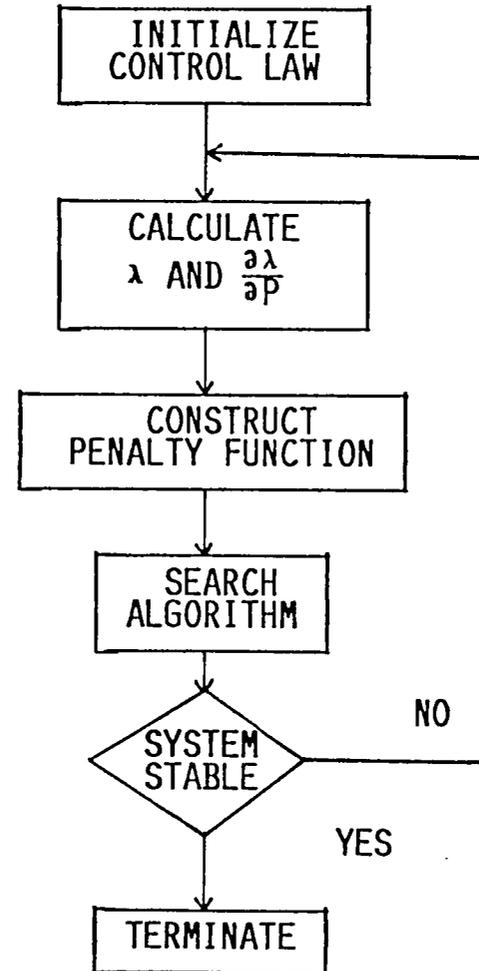


Figure 33(b).

METHOD DEVELOPED TO DETERMINE A CONTROL LAW  
 THAT STABILIZES AN UNSTABLE SYSTEM

EXAMPLE

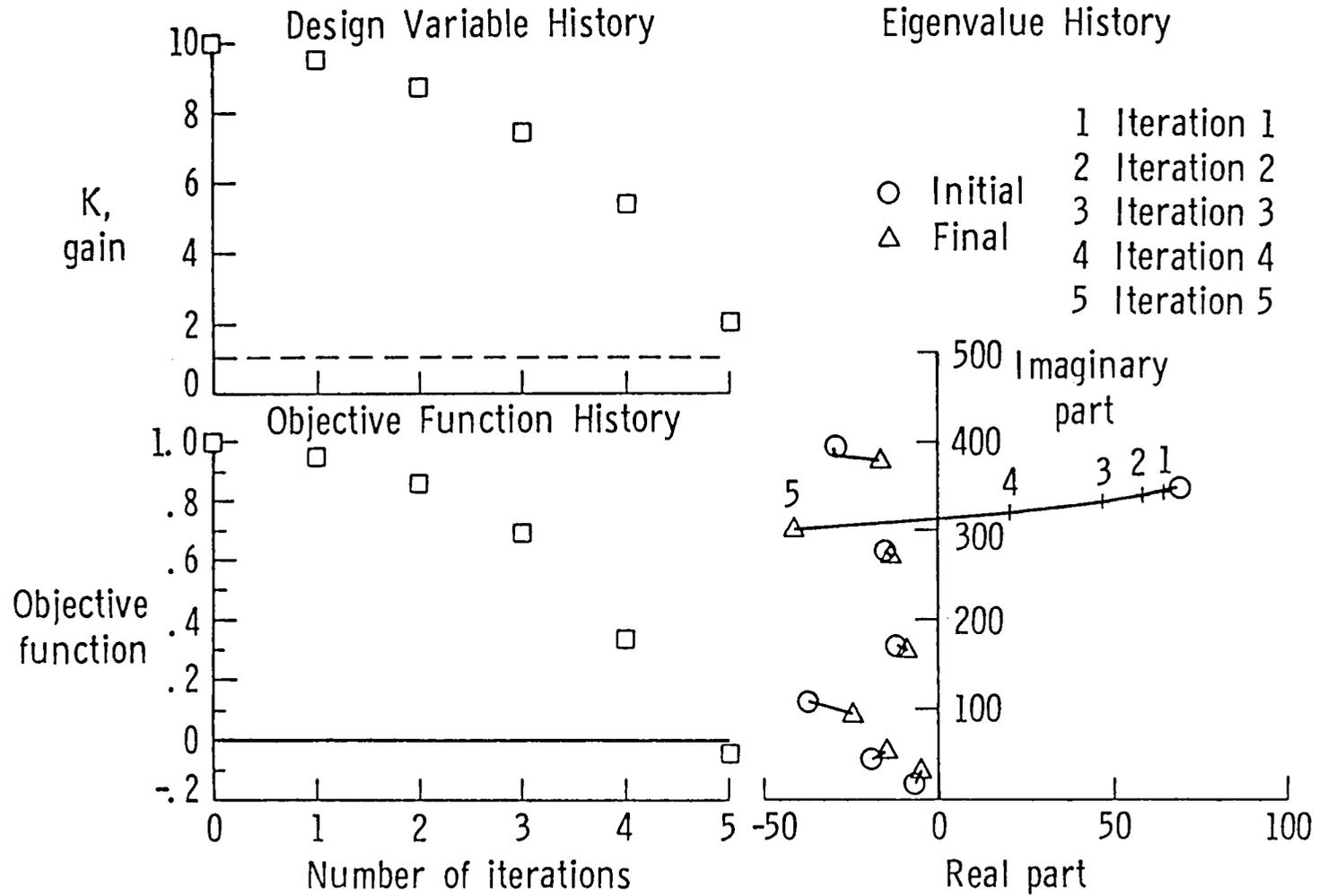


Figure 33(c).



FLEXSTAB RESULTS USED TO MODIFY WIND TUNNEL RIGID MODEL DATA FOR  
PREDICTION OF FLEXIBLE AIRPLANE PERFORMANCE

C. V. Eckstrom  
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Extension 3834

RTOP 505-33-43

Research Objective - Basic stability and control data are available for the DAST program flight vehicle with the Aeroelastic Research Wing (ARW-2) from measurements on a model tested in a wind tunnel through a range of Mach numbers. The wind tunnel test model was essentially a rigid structure with the wing in the desired shape for the design cruise conditions (Mach number of 0.80 and  $C_L = .53$ ). The research objective is to obtain stability and control data for the flight research wing, which is quite flexible, by modifying the wind tunnel data to account for: (1) the difference between the wind tunnel model design cruise wing shape and the flight wing fabrication shape, and (2) the shape changes that occur to the flexible flight wing because of aerodynamic loadings.

Approach - A Flexible Airplane Analysis Computer program called FLEXSTAB is being used as the analysis tool. To run FLEXSTAB it is necessary to input data describing the DAST ARW-2 wing and aircraft geometric and structural characteristics and the desired flight conditions of Mach number and dynamic pressure. Because the FLEXSTAB program uses a linear aerodynamic analysis method it is unable to accurately predict the surface pressure distribution for a supercritical airfoil like that used on ARW-2. To improve accuracy, corrections to the computed wing surface pressures are made, based on limited wind tunnel measurements, using input provisions provided by FLEXSTAB. In spite of this, the FLEXSTAB predicted aircraft stability and control characteristics do not match wind tunnel test data sufficiently well to be used directly. The procedure is to obtain FLEXSTAB results for: (1) rigid models at both the design cruise shape and the fabrication shape, and (2) for a flexible model (initially in the fabrication shape) at various levels of flight dynamic pressure. These FLEXSTAB calculations provide useable results in the form of: incremental changes in angle of attack at zero lift, lift curve slope changes as a function of dynamic pressure, incremental changes in pitching moment at zero lift, and incremental shifts in aerodynamic center location that are then applied to the measured wind tunnel data to establish the stability characteristics of the research wing.

Figure 34(a).

FLEXSTAB RESULTS USED TO MODIFY WIND TUNNEL RIGID MODEL DATA FOR  
PREDICTION OF FLEXIBLE AIRPLANE PERFORMANCE

Continued

Accomplishment Description - The FLEXSTAB program has been used as described to predict incremental changes in stability and control characteristics for the DAST ARW-2 vehicle and research wing as a function of fabrication shape and aerodynamic loading. A procedure has also been established for modifying the wind tunnel data based on the predicted incremental changes from FLEXSTAB. Results for the Mach number 0.80 case are presented on figure 34(c) for dynamic pressure variations from zero (rigid case) to 800 psf. The difference between the wind tunnel rigid model data and the prediction for the flexible airplane at  $q=0$  psf is a result of the flexible airplane fabrication shape being different than that of the wind tunnel model. The changes in slope that occur for the flexible airplane data are from predicted changes in wing shape (twist and bending) that result from aerodynamic loading. The changes in lift and pitching moment coefficients are significant and must be planned for in the development of the control system for the aircraft.

Future Plans - The procedure developed and reported on here is for modification of wind tunnel model data obtained with the horizontal tail surfaces on. For the DAST wind tunnel model, data are available for both tail-on and tail-off configurations, which allows determination of downwash at the tail. Consequently activities are in progress to develop a procedure using FLEXSTAB results to also modify the downwash characteristics. Flexibility effects will then be determined for the DAST ARW-2 using the more detailed procedure which will include modifications to the downwash characteristics.

FLEXSTAB RESULTS USED TO MODIFY WIND TUNNEL RIGID MODEL DATA  
FOR PREDICTION OF FLEXIBLE AIRPLANE PERFORMANCE

(MACH NUMBER = 0.80)

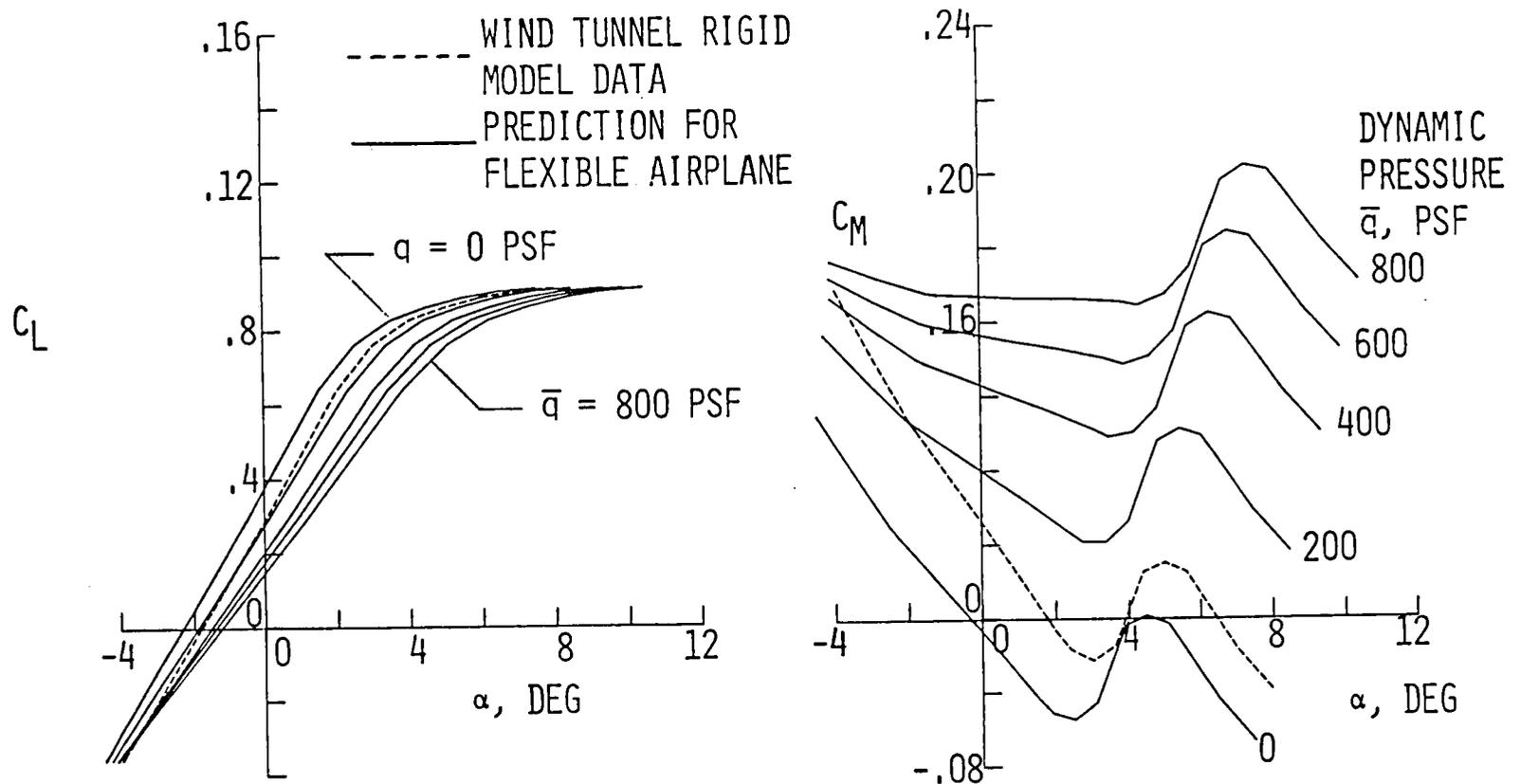


Figure 34(c).

# WING SURFACE PRESSURES MEASURED DURING FLIGHT TEST DEFINE SHOCK LOCATION

C. V. Eckstrom  
Aeroservoelasticity Branch  
Extension 3834

RTOP 505-33-43

Research Objective - Surface pressure measurements were made during flight testing of the DAST program Aeroelastic Research Wing (ARW-1) to obtain pressure distributions for a flexible supercritical wing through a range of off design flight test conditions. The research objective is to evaluate the data obtained to define changes in pressure distribution as a result of variations in angle of attack, Mach number, and dynamic pressure.

Approach - The data acquisition system used provided for the continuous transmission of flight test data to the ground station where it was recorded on magnetic tape. As a result, large quantities of data are available from which measured wing surface pressure coefficients for any orifice location can be presented as a function of angle of attack at fixed Mach numbers or as a function of Mach number for fixed angles of attack. By using both of these relationships, it is possible to fairly accurately define the progress of the shock wave across any given orifice location as a function of angle of attack and Mach number.

Accomplishment Description - Surface pressure measurements obtained from several chordwise orifice locations on the wing upper surface at an inboard station ( $\eta = .345$ ) are presented as a function of Mach number in figure 35(b). The data presented are for an aircraft angle of attack of two degrees. The recompression shock wave is considered to be at a particular orifice location when the surface pressure gradient as a function of Mach number has a nearly vertical gradient. The higher or more negative values of surface pressure coefficient indicate supersonic flow and the lower values subsonic flow. As the Mach number increases the shock wave moves aft. At a Mach number of 0.84 the shock is at  $x/c = 0.20$  but moves back as far as  $x/c = .61$  at a Mach number of 0.91. Of special interest is how well the measurements define the significant changes in surface pressure which occur as the increasing Mach number brings the shock wave back toward and across each orifice location.

Future Plans - Data available from the referenced flight tests, including the surface pressure data presented here, will be reported in a proposed NASA TM.

Figure 35(a).

WING SURFACE PRESSURES MEASURED DURING FLIGHT TEST DEFINE SHOCK LOCATION

( $n = .345$ , UPPER SURFACE,  $\alpha = 2^\circ$ )

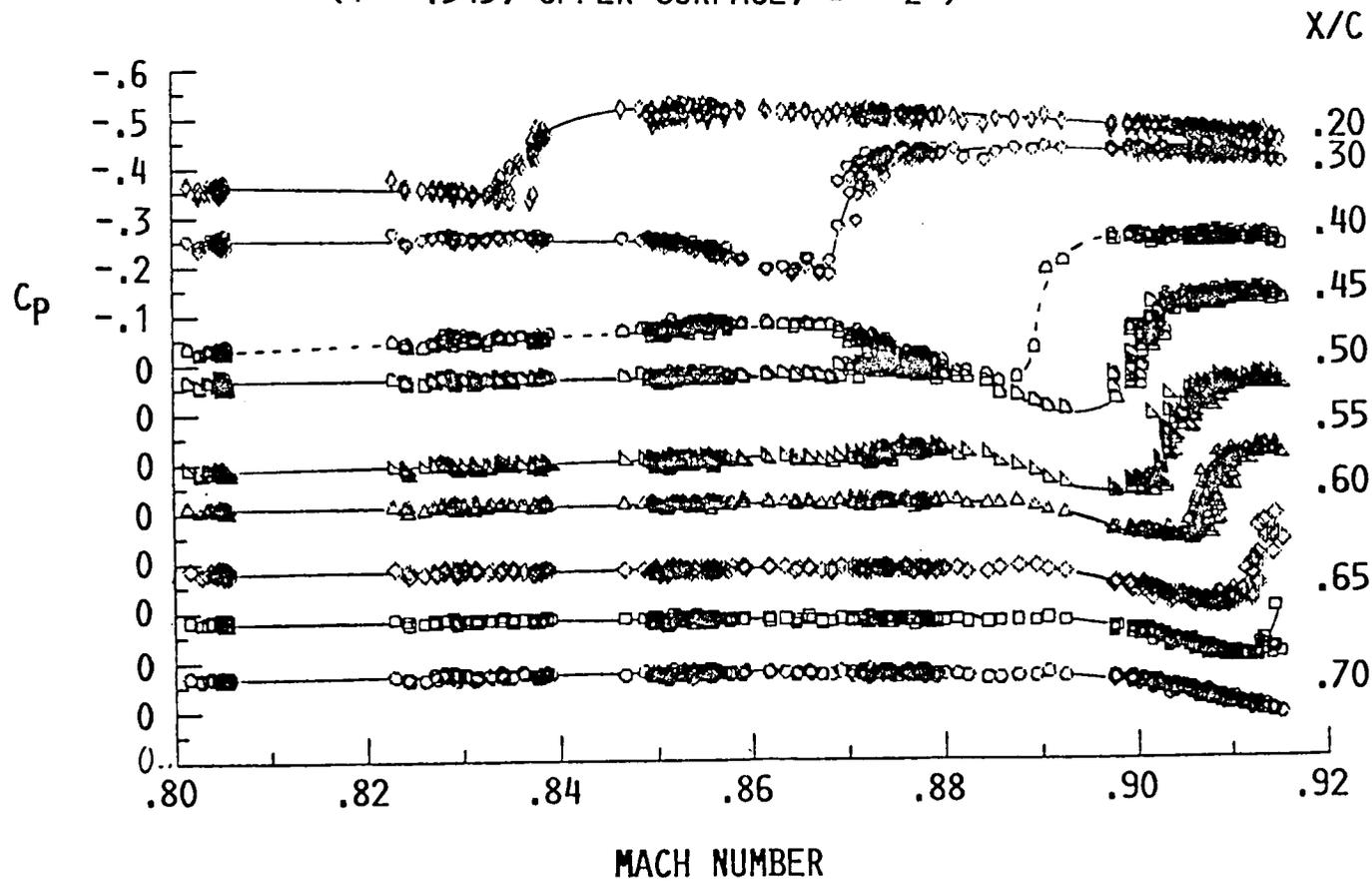


Figure 35(b).

### AEROTHERMAL LOADS

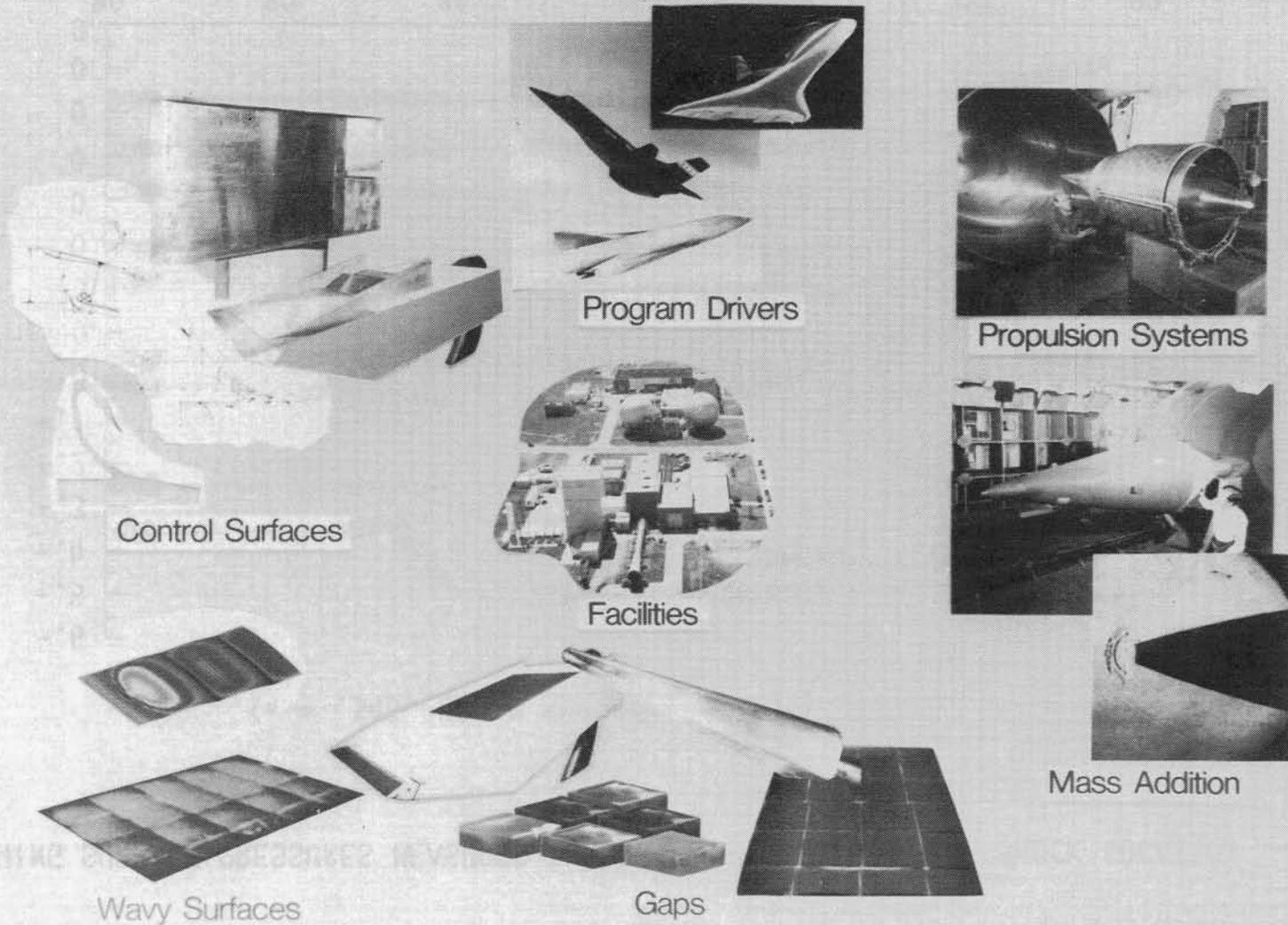


Figure 36.

AEROTHERMAL LOADS  
5-YEAR PLAN

DISCIPLINE	FY 84	FY 85	FY 86	FY 87	FY 88	EXPECTED RESULTS	
THERMAL LOADS	SPHERICAL PROTUBERANCE		QUILTED SURFACE		CYL. PROTUB.	DETAILED THERMAL DESIGN LOADS	
	SPLIT ELEVON		CHINE GAP HEATING				
	SHOCK-SHOCK INTERACTION						
	8' HTT BOUNDARY LAYER CHARACTERIZATION		2D & 3D COMPRESSION CORNER HEATING		AXIAL CORNER HEATING		
	MASS ADDITION FLOW EFFECTS						
INTEGRATED ANALYSIS	3D F. E. EULER CODE		CONTROL OF SPURIOUS NUMERICAL OSCILLATIONS			INTEGRATED FLOW-THERMAL STRUCTURAL ANALYSIS CAPABILITY	
	VALIDATE 3D F.E. NAVIER-STOKES COMP. VISCOUS FLOW CODES						
				ERROR ANALYSIS TECHNIQUES			
	F. E. AND SPECTRAL NONLINEAR ALGORITHMS						
	INT. GRAPHICS		ADAPTIVE GRID & TIME DOMAIN SPLITTING				
FACILITIES & TEST TECHNIQUES	HIGH TEMP. INSTRUMENTATION (TEMPS, HEATING RATES, DISP.)						
	8' HTT APPROACH SECTION		HIGH SPEED DAS		FY85 CF INSTALL. & CHECKOUT		
	O <sub>2</sub> SENSORS & CONTROLS		86 COF REHAB 7" HTT & AAT				
	1x3 HEAT CALIBR.		FEASIBILITY STUDIES FOR INCREASED ENHTALPY & MACH # FOR 8' HTT				
	EVALUATE ALT. M/MIXER CONCEPT						

Figure 37.



## AEROTHERMAL TEST OF SHUTTLE SPLIT-ELEVON MODEL IN 8'HTT

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Extension 3423

RTOP 506-51-23

87 Research Objective - The two elevons on each of the Shuttle wings are split and are separated with a chordwise gap of 6 inches where complex flow interactions are produced that are not amenable to proven analytical techniques. In the original Shuttle design, the specified aerothermal loads allowed the use of HRSI tiles in the gap for thermal protection. However, test results from small-scale laminar test at NASA ARC suggested that the HRSI tiles were not adequate and replacable ablation panels were used on the first five Shuttle flights. When it was clear that the heat load was much lower than expected, the original tile design was reinstated. Flight results indicate that the present design is adequate, but the excessive heating on the windward edge needs to be addressed. Earlier large-scale test results in the Langley 8' HTT indicated that the gap heating was less than the original design level. To provide a detailed aerothermal load distribution with critical test parameters varied, a 1/3 scale model of the Shuttle split elevon as shown in figure 38(b) was mounted to the previously used model for test in the 8' HTT for both laminar and turbulent flow conditions.

Approach - The elevon portion of the existing model was modified to extend and streamline the elevons to chord length of 24 inches and to vary the gap width from 1 to 3 inches. The elevons were designed for a sharp or a rounded windward leading edge to define possible edge effects on the aerothermal loads. Tests were performed in the 8' HTT (figure 33(c)) at a free-stream Mach number of 6.5, a unit Reynolds number range from 0.4 to  $1.5 \times 10^6$ , and a total temperature of 3500°R. The model wing angle was varied from 0 to 10 degrees and the elevon angles were varied from 0 to 20 degrees.

Accomplishment Description - The basic data from all 31 test runs have been reduced and preliminary data plots have been generated. Generally, the present results agree with the low heating level in the chordwise gap indicated in the earlier tests. The peak heating in the gap is driven by the elevon windward surface pressure and is not directly related to whether the wing boundary layer is laminar or turbulent.

Future Plans - Results will be documented in a NASA TP and be available for comparison with analytical solutions when appropriate computer codes are identified to handle these flow details.

Figure 38(a).

SHUTTLE SPLIT ELEVON GAP HEATING MODEL IN 8' HTT

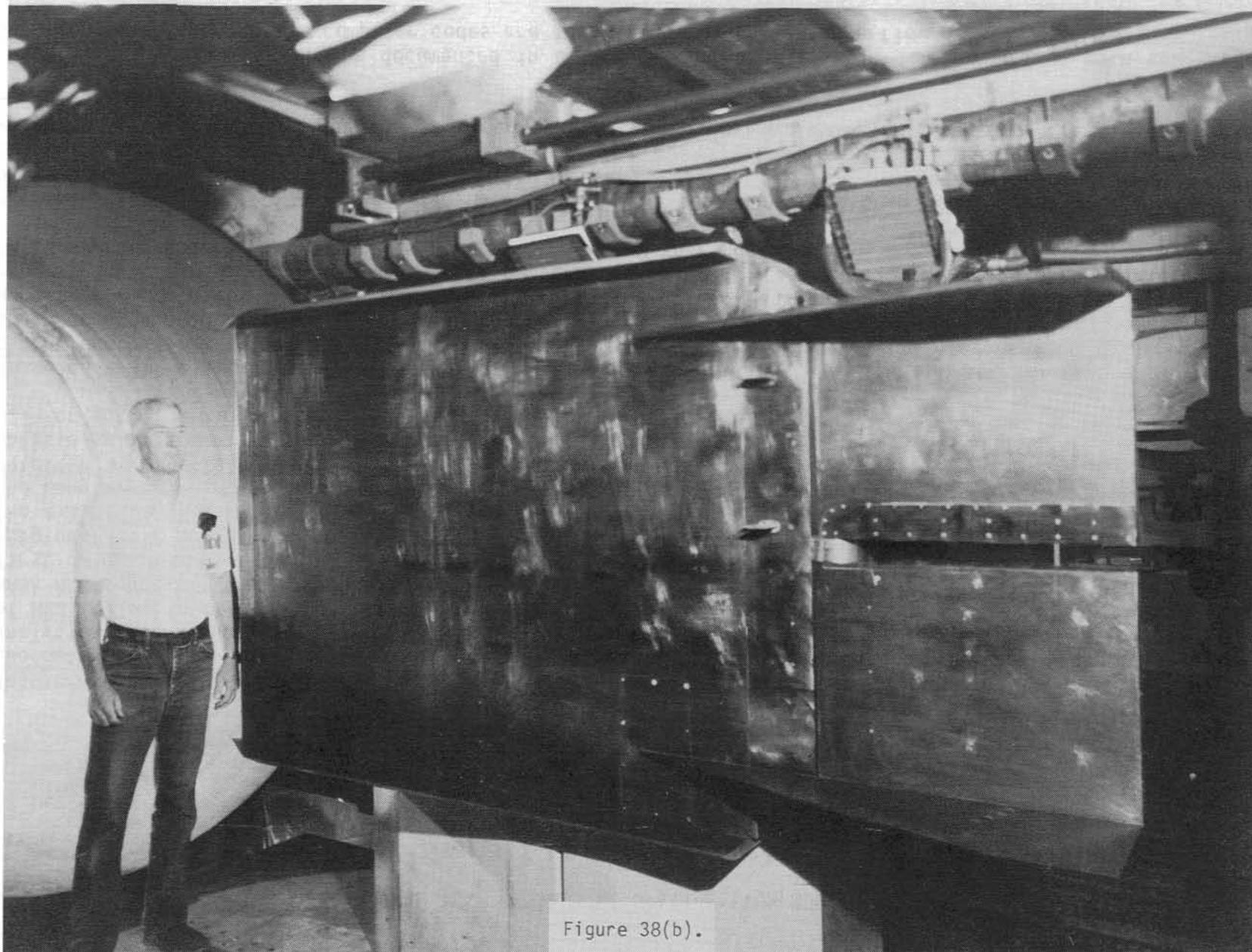
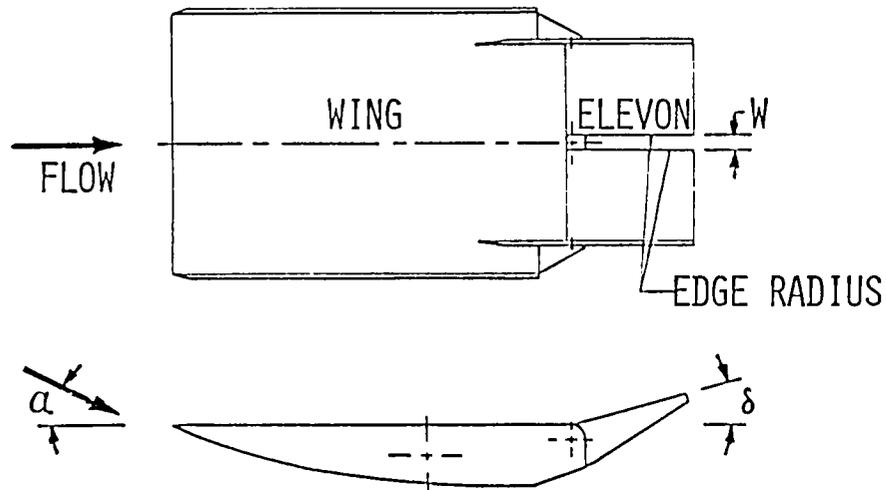


Figure 38(b).

# SHUTTLE SPLIT ELEVON GAP HEATING TEST IN 8' HTT COMPLETED

8' HTT 1/3 SCALE MODEL SCHEMATIC



TEST CONDITIONS

MACH NO.	6.5
RE/FT x 10 <sup>-6</sup>	0.4 - 1.5
T <sub>T</sub>	3500°R
q	2.5 - 10 PSI
α	0, 5, 10 DEG.
δ	0, 5, 10, 20, DEG.
W	1, 2, 3, IN.
EDGE RAD.	0, 0.25 IN.
TEST RUNS	14 LAMINAR 17 TURBULENT

Figure 38(c).

## AEROTHERMAL TESTS OF SPHERICAL DOME PROTUBERANCE MODELS IN 8'HTT COMPLETED

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Aerothermal Loads Branch  
Extension 3423

RTOP 506-51-23

Research Objectives - Metallic TPS panels on high speed vehicles are subject to thermal distortions when they experience large through-the-thickness temperature gradients. The panels, anchored at the corners, bow up and form a spherical dome protuberance into the flow field. Although the protuberance height of the panels is expected to be less than the local boundary layer thickness, the complex interaction of the high-speed flow field and the bowed surface will effect the local and global aerothermal loads to the vehicle. An experimental aerothermal study is required to complement current analytical studies of this problem.

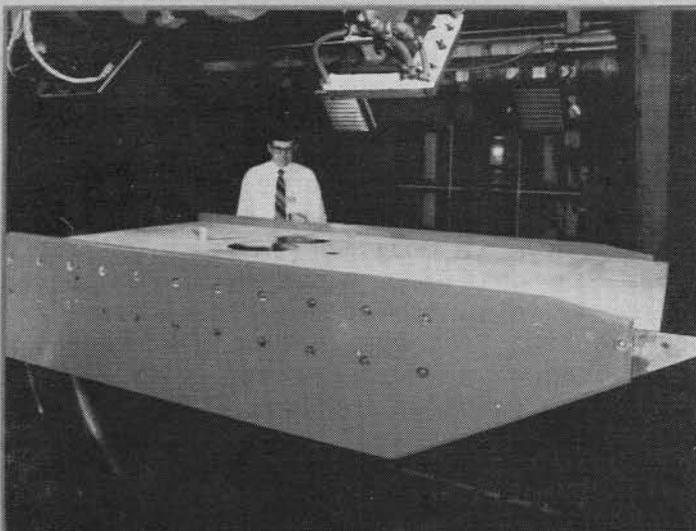
Approach - An array of spherical dome protuberance models were designed and fabricated to fit the 8' HTT panel holder (figure 39(b)), which provides two-dimensional laminar or turbulent boundary layer flow. The dome dimensions were varied for diameters of 7, 14, and 28 inches and heights of 0.05, .1, .2, .4, and .8 inches. The design included three types of domes: metallic pressure instrumented, metallic thin-wall heat-transfer, and ceramic domes to obtain surface temperatures approaching equilibrium from infrared photography. Aerothermal tests were performed at a free-stream Mach number of 6.5, a unit Reynolds number of  $0.4 \times 10^6$  per foot, and a total temperature of 3500°R.

Accomplishment Description - Preliminary data analysis of the 37 test runs have been completed. Results include surface pressure and heating rate distributions comparing various test parameter effects, pressure and heating rate contour plots, and boundary layer profiles. These results agree qualitatively with the 3D Navier-Stokes results for laminar boundary layer flow obtained earlier.

Future Plans - Results will be documented in a NASA TP including a comparison of the Navier-Stokes solution of the lamiiar flow condition. Similar tests of multiple domes (quilted pattern) to simulate thermally bowed TPS array will be performed in FY 85.

Figure 39(a).

## AEROTHERMAL TESTS OF SPHERICAL DOME PROTUBERANCE MODELS IN 8'HTT COMPLETED



### Test Conditions

Mach no.	6.5
$Re/ft \times 10^{-6}$	0.4
$T_t$	3500 R
$q$	2.5 psi
$\delta$	0.81 in.- Laminar 1.42 in. -Turbulent
Test Runs	19- Laminar 18- Turbulent

### Dome Dimensions

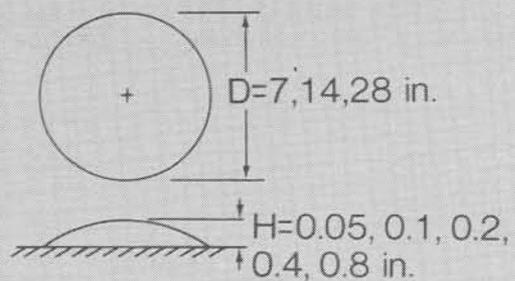


Figure 39(b).

## FINITE ELEMENT MULTIPLE TIME DOMAIN ALGORITHM

Kim S. Bey  
Aerothermal Loads Branch  
Extension 4441

RTOP 506-51-23

Research Objective - Compressible flow solutions typically contain regions of large gradients and discontinuities. A finite element model of the solution domain will contain small elements in these high gradient regions and larger elements in regions where the solution is smooth. For computational stability, the critical time step used to march the solution is governed by the smallest element. This results in a time step that is smaller than necessary for the larger elements. An algorithm which allows different time steps in different regions of the mesh is needed for computational efficiency.

Approach - Dr. Ken Morgan from the University of Wales (NAGW-478) has developed a domain splitting scheme which is accurate for the true transient. The attached figure illustrates the solution procedure. While it is not practical, the illustration is on a one-dimensional mesh for the sake of simplicity.

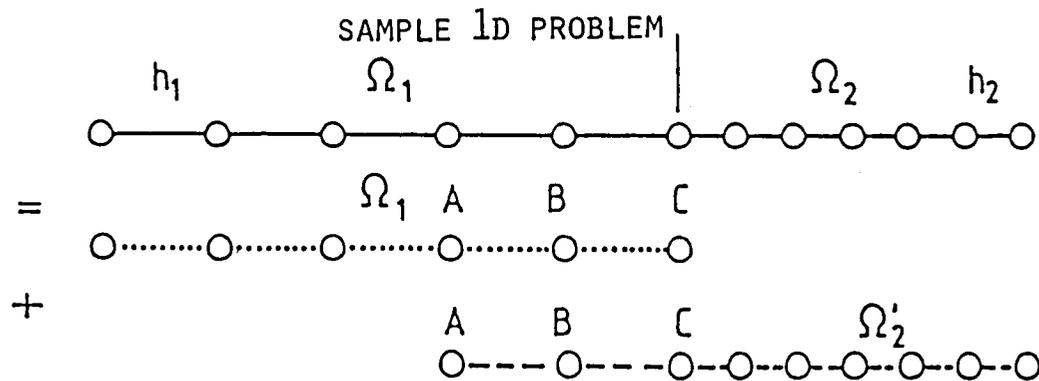
The global solution domain has two subdomains,  $\Omega_1$  and  $\Omega_2$ . The lengths of the elements are  $h_1$  and  $h_2$  respectively. If the lengths are such that  $h_1 = nh_2$  then the allowable time steps are related by  $\Delta t_1 \Delta = n t_2$ . The solution procedure is as follows: (1) Add to  $\Omega_2$  (the smaller timestep domain) two grid points from  $\Omega_1$  (the larger domain) and call the new subdomain  $\Omega'_2$ . (2) Fix the values of the unknowns at the subdomain boundaries (From figure 40(b): fix point C in  $\Omega_1$  and point A in  $\Omega'_2$ ). Advance the solution one timestep  $\Delta t_1$  in  $\Omega_1$  and  $n$  time steps  $\Delta t_2$  in  $\Omega'_2$ . (3) Transfer results to global domain (use values from  $\Omega_1$  for point A, values from  $\Omega'_2$  for point C, and average values from  $\Omega_1$  and  $\Omega'_2$  for point B. All other values are those obtained from  $\Omega_1$  and  $\Omega'_2$  in step 2). The computations for each subdomain are done independently and could be performed in parallel.

Accomplishment Description - This time-domain splitting technique has been implemented in a 2D inviscid finite element algorithm. Several transient and steady state problems were analyzed. Results show that this procedure is as accurate as the single domain scheme for both types of problems. In each of the problems worked, the procedure was twice as fast as with the single domain.

Future Plans - This technique will also be applied to viscous flow problems where smaller elements are used to resolve boundary layers. The implementation in a 3D algorithm will also be investigated.

Figure 40(a).

FINITE ELEMENT MULTIPLE TIME DOMAIN ALGORITHM  
AS ACCURATE AS AND FASTER THAN SINGLE DOMAIN ALGORITHM



SOLUTION PROCEDURE

- SUBDIVIDE DOMAIN
- OVERLAP 3 GRID POINTS (A,B,C) INTO ADJACENT LARGER TIME DOMAIN
- SOLVE SUBDOMAINS SEPARATELY
- TRANSFER RESULTS TO GLOBAL DOMAIN
- AVERAGE INTERIOR(B) OVERLAP RESULTS

INITIAL EVALUATION FOR 2D INVISCID FLOWS

- AS ACCURATE AS SINGLE DOMAIN IN TRANSIENT AND STEADY STATE
- TWICE AS FAST AS SINGLE DOMAIN FOR PROBLEMS SOLVED

Figure 40(b).

## AUTOMATIC FINITE ELEMENT MESH REFINEMENT

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Aerothermal Loads Branch  
Extension 4441

RTOP 506-51-23

Research Objective - Solutions of compressible flow problems are characterized by discontinuities such as shock waves. Numerical techniques must be able to accurately resolve such phenomena. Since the analyst does not know the exact location of these large gradients, the ideal algorithm should automatically refine the mesh in these regions throughout the computation.

Approach - Using the finite element method because of its inherent ability to handle complex geometries and capture shocks, the University of Wales under NASA grant has developed and implemented an automatic mesh refinement technique. This mesh enrichment technique adds elements to the mesh, at certain times in the solution, based on regions of high gradients so as to evenly distribute (or minimize) the solution error.

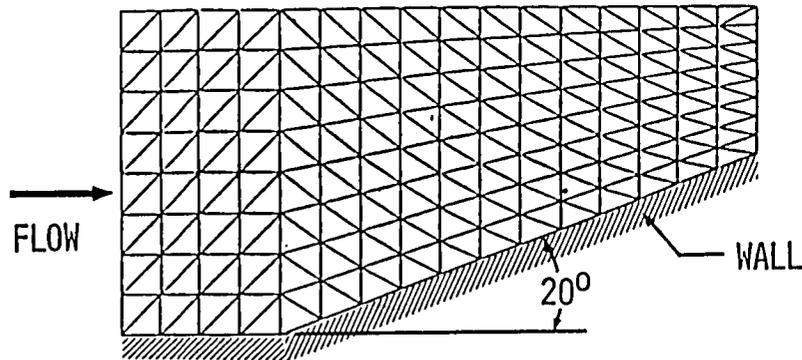
Accomplishment Description - The effectiveness of this technique has been demonstrated for a number of inviscid compressible steady flow problems. The results for flow past a compression corner are shown in figure 41(b). The solution is started with a crude and fairly uniform mesh. After every 100 timesteps the algorithm adds elements in the shock region, so that after 300 timesteps there are many small elements near the shock giving good resolution unobtainable with the original mesh.

Future Plans - The actual coding of this technique is extremely complicated and currently is only applicable to steady-state problems. Also, vectorization of the process is at this point uncertain. Morgan is now investigating the application of mesh enrichment to viscous flow problems in the resolution of 2D boundary layers. He will also consider this technique for solving 3D problems.

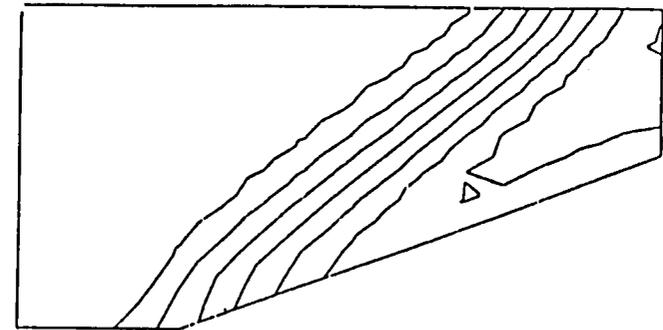
Figure 41(a).

AUTOMATIC GRADIENT SENSITIVE FINITE ELEMENT MESH REFINEMENT ALGORITHM DEMONSTRATED

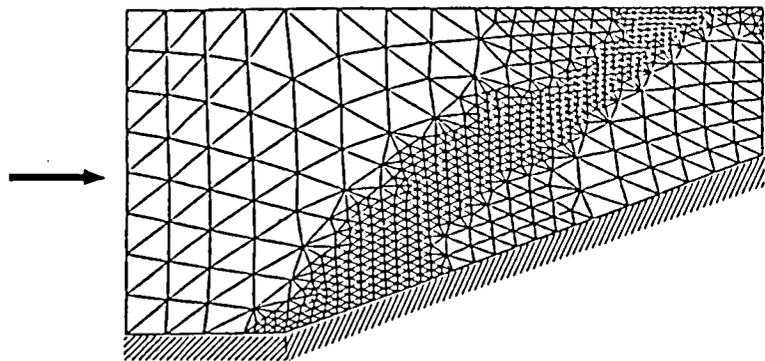
INVISCID SUPERSONIC FLOW PAST A COMPRESSION CORNER



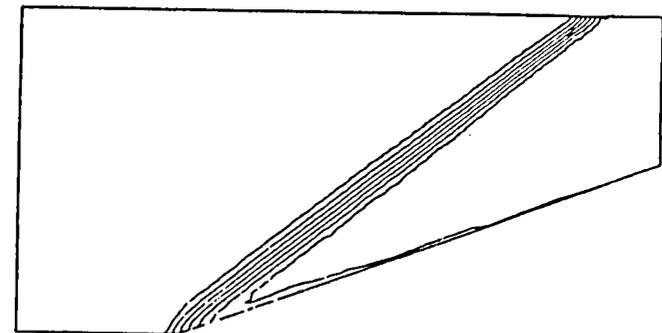
ORIGINAL MESH



DENSITY CONTOURS AFTER 100 TIME STEPS



MESH AFTER SECOND REFINEMENT



DENSITY CONTOURS AFTER 300 TIME STEPS

Figure 41(b).

## FINITE ELEMENT FLOW-THERMAL ANALYSIS

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Extension 4441

RTOP 506-53-33

Research Objective - Develop an integrated flow-thermal structural analysis capability to accurately predict aerodynamic heating and structural response due to high speed flow.

Approach - Finite element methodology has reached a relative maturity for integrated thermal-structural analysis. Our approach is to take advantage of existing capabilities and develop finite element methodology for high speed compressible flow. This development includes using color graphics for model generation and display of results, evolving effective solution algorithms, and validation through comparison with other solution techniques and experimental data.

Accomplishments Description - Figure 42(b) illustrates current finite element flow analysis capabilities. 2D and 3D model generation and results display are done with PATRAN. This required development of software to translate the general PATRAN neutral data file to an input file for the flow analysis code. A vectorized 2D finite element code was developed for evaluating algorithms. The density distribution for flow over a forward facing step, shown in the figure, was obtained with this code for 2D inviscid flow. This demonstrates the shock capturing abilities of the algorithm.

Future Plans - Further evaluation of the 2D inviscid algorithm continues by comparisons with experimental data from 8<sup>1</sup> HTT tests. The vector code is being extended to 3D. Also, viscous terms are being added to the 2D code.

Figure 42(a).

# PATRAN SUPPORTED 2D FE ANALYSIS INVISCID FLOW OVER FORWARD FACING STEP

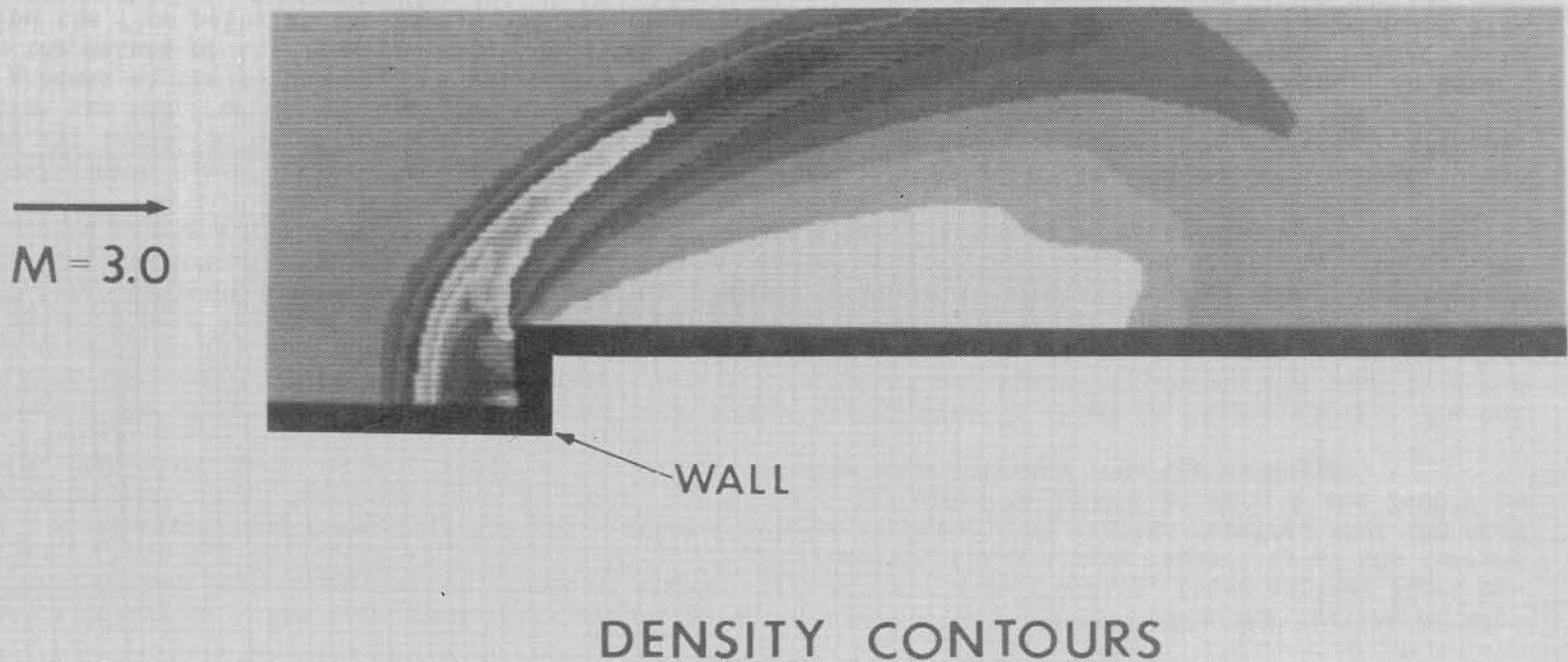


Figure 42(b).

## MIXER PROTOTYPE FOR THE 8' HIGH TEMPERATURE TUNNEL

Richard L. Puster  
Aerothermal Loads Branch  
Extension 3115

RTOP 505-33-53

Research Objective - The test capability of the 8' HTT is being enlarged to enable the testing of air-breathing engines and integrated airframes on a full scale basis. Almost always, these engines start operation at Mach 3.5 to 4.0 at altitudes of 40 to 50 kft; however, the altitude capability of the current 8' HTT simulates flight from 80 to 130 kft. Using the large diameter test section requires that the mass flow be significantly increased and the total temperature decreased to 1600°R at Mach 4 and 2400°R at Mach 5. The mixer performs this function by mixing cold flow with hot flow from the combustor.

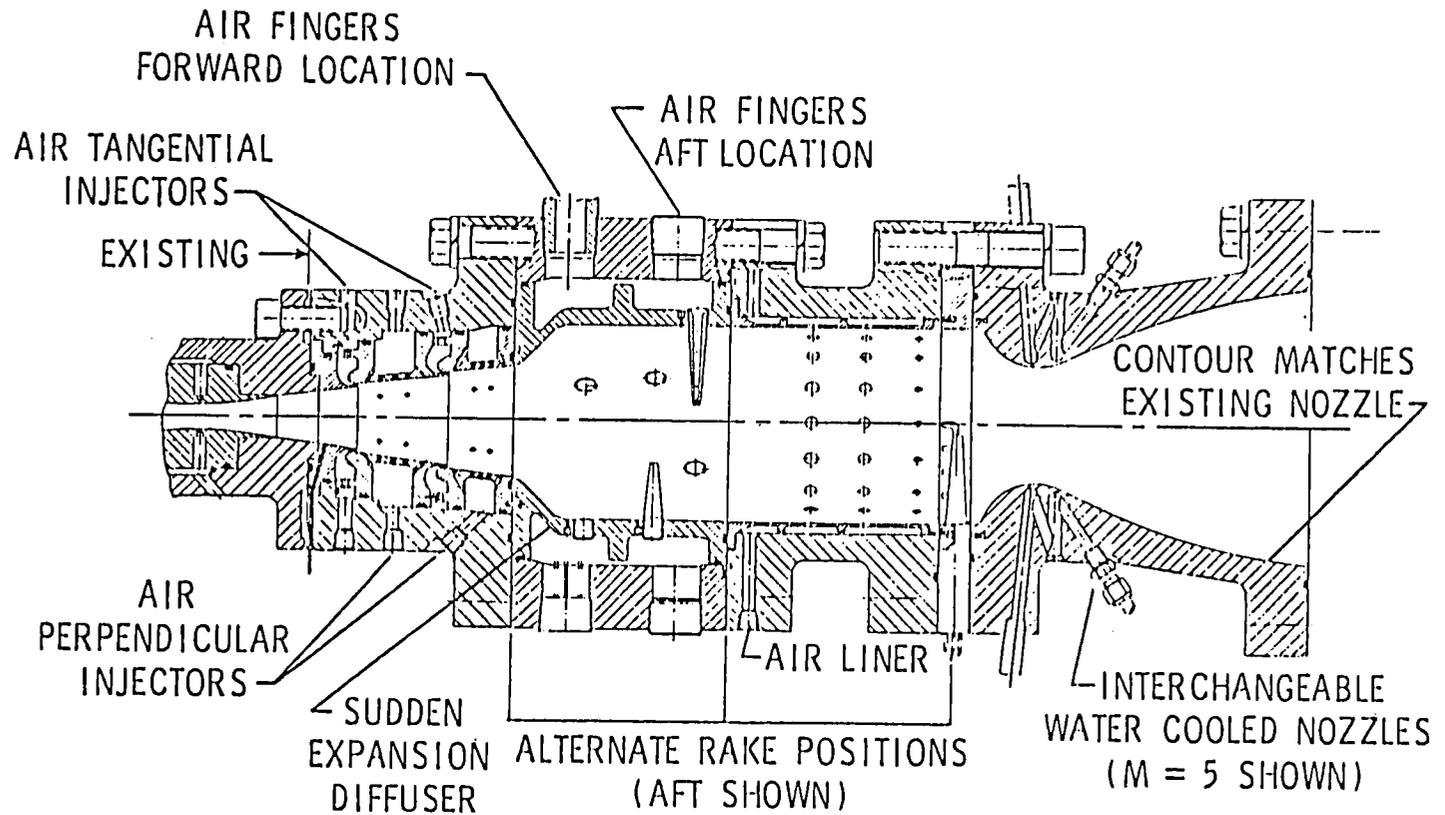
Approach - The mixer apparatus and Alternate Mach Number arrangement is shown in figure 43(b). The hot flow from the combustor is expanded to about Mach 3; tangential and normal injection are used in a two stage process to (1) cool the walls; (2) start the mixing process; (3) diffuse the flow. When the flow reaches the main injector it is decelerated to subsonic flow by a normal shock wave and mixed with massive injection of air normal to the flow with a sudden increase in area. The flow then flows through a cooled chamber that stills the flow which is then fed to a second nozzle as shown in the figure. The downstream nozzle expands the flow to Mach 4 or 5. A scale model has been built and tested.

Accomplishment Description - To determine the effectiveness of the mixer, a temperature subsonic probe array was installed between the end of the mixer and nozzle. In addition, total temperature in the test section was measured across the flow using three different types of probes. To evaluate the effect of the process on the container, the walls were instrumented with temperature and pressure taps. To ascertain the nature of the flow process, the mixer walls were coated with a fluorescent mixture that would follow the flow paths at the wall. The oil flow pattern is shown figure 43(c) and the temperature distributions shown in figure 43(d). The mixer works extremely well with a Mach 4 throat. The hot and cold gases are thoroughly mixed. The oil flow photographs give graphic evidence of how the mixing proceeds. The flow from the normal injectors at the step induce a secondary flow back upward and around the injectors. This secondary flow has another tumbling vortex type flow pattern in between the secondary flow. In addition, there is a staggered array of holes slightly upstream of the holes at the step with a similar mixing process. As evidenced by the oil flow, the entire separation and mixing zone reattaches to the wall of the stilling chamber near the beginning of the cooled mixer liner. Thus, mixing of two widely dissimilar fluids is completed in a relatively short distance, remarkable considering the high velocity of the fluids (normal, sonic; mean - 400 ft/s).

Future Plans - The mixer will next be evaluated with the Mach 5 nozzle in place and with dynamic instrumentation. Controlled oxygen addition will be evaluated at all Mach numbers (4, 5, and 7) with both an open and a closed loop system. the closed or precision control will use an active detector and controller to keep the oxygen content at 21 percent.

Figure 43(a).

TESTS VERIFY DESIGN FOR 8-FT HTT  
MIXER PROVIDES UNIFORM TEMPERATURE



SCALE IS 1/12.8 OF FULL SCALE

PROTOTYPE OF MIXER WITH ALTERNATE  
MACH NUMBER NOZZLE

Figure 43(b).

NASA  
L-84-13,431

# FLOURESCENT OIL FLOW PATTERN

RECIRCULATION REGION

AIR INJECTOR

6 AT STEP

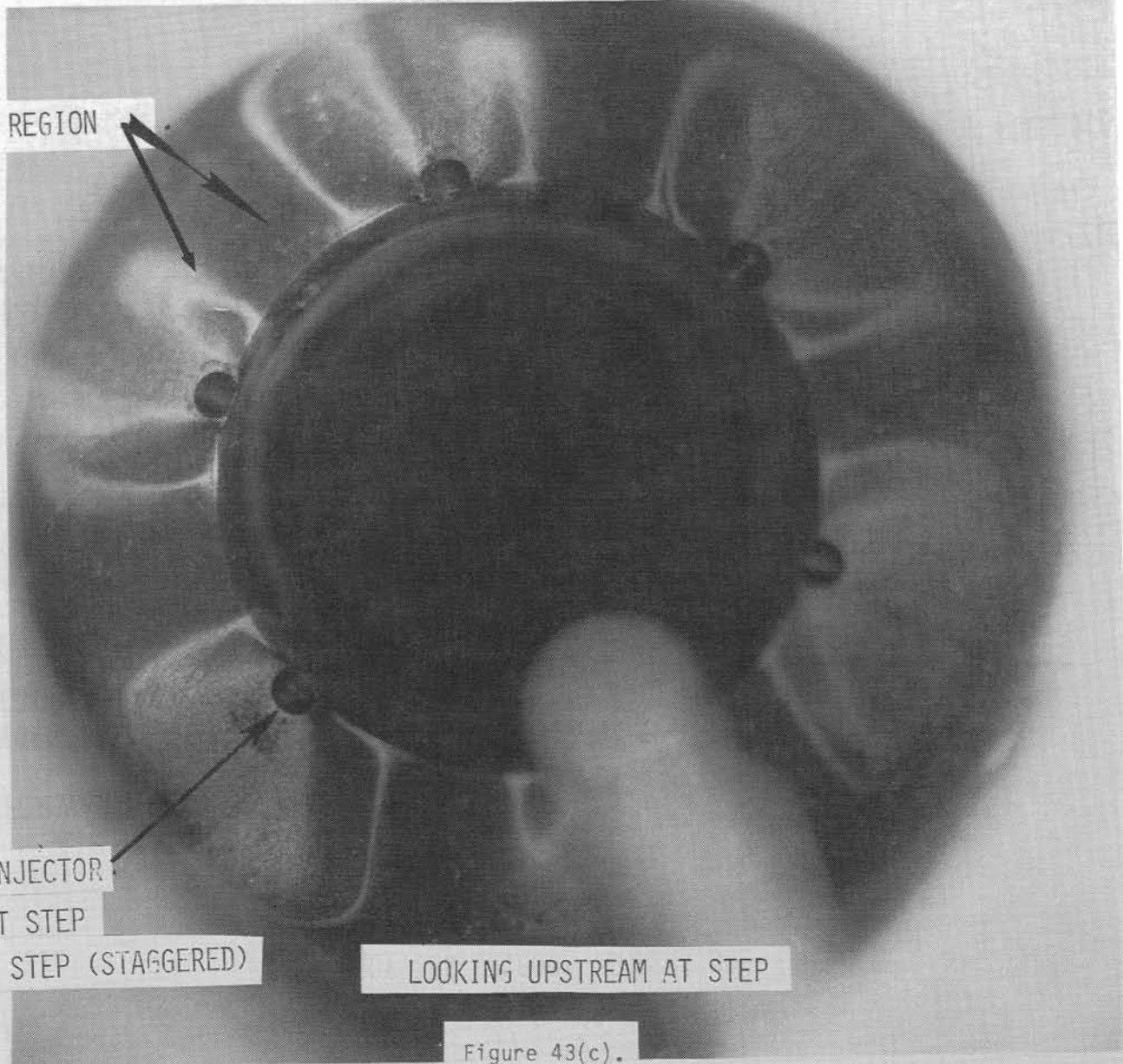
6 AHEAD OF STEP (STAGGERED)

LOOKING UPSTREAM AT STEP

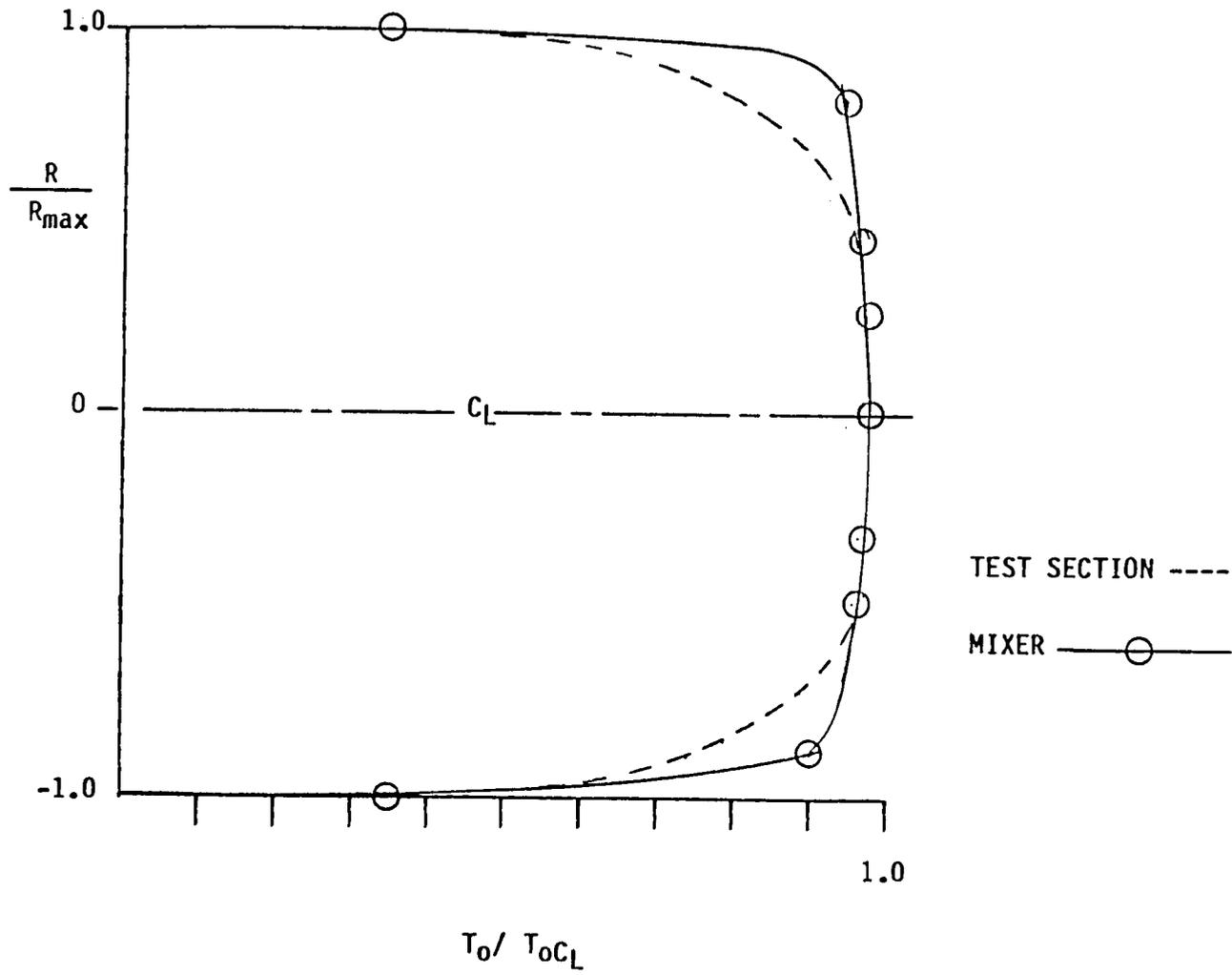
Figure 43(c).

100

100



# TOTAL TEMPERATURE PROFILES



101

Figure 43(d).

# THERMAL STRUCTURES

Concept Development  
and Verification

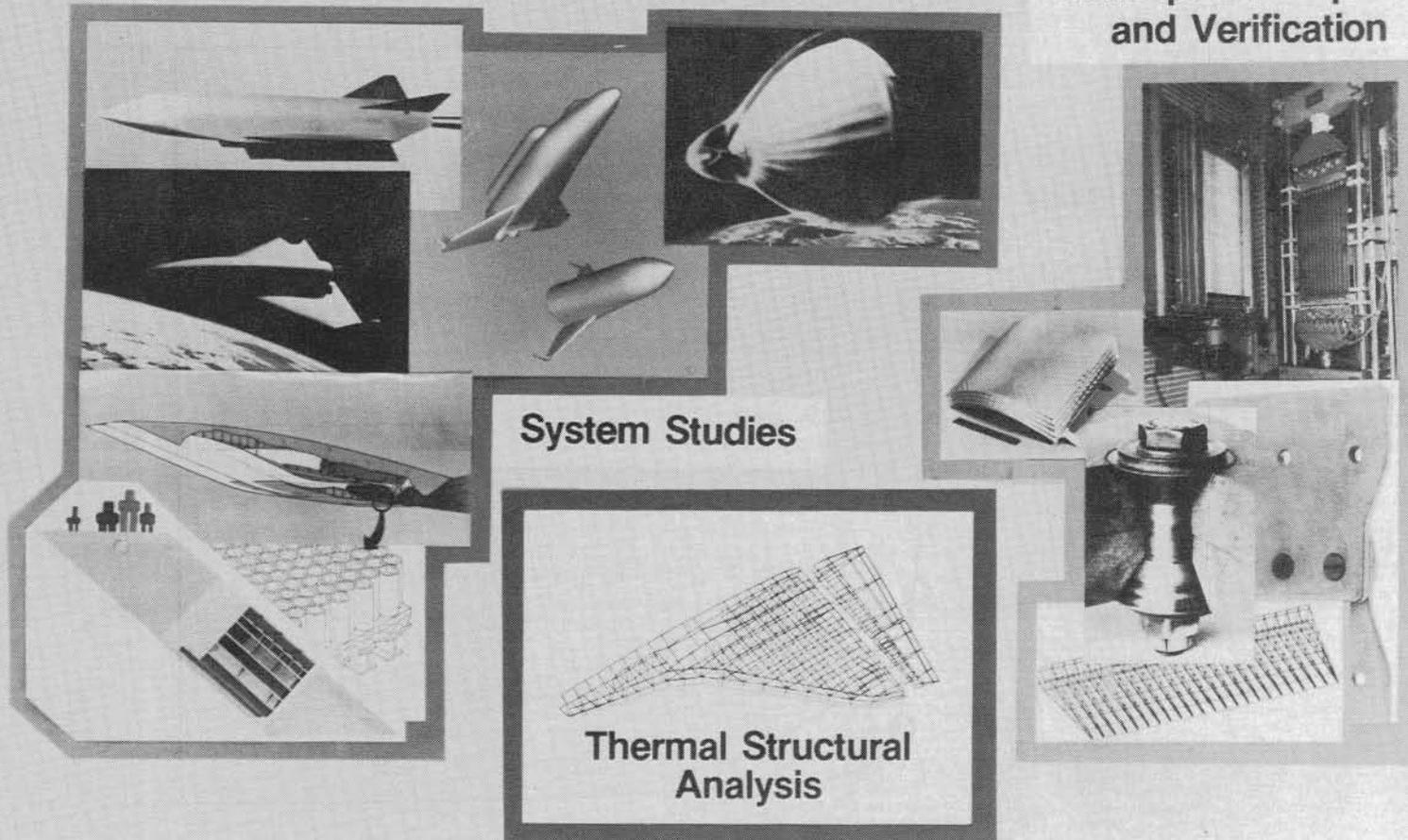


Figure 44.

THERMAL STRUCTURES  
5-YEAR PLAN

DISCIPLINE	FY 84	FY 85	FY 86	FY 87	FY 88	EXPECTED RESULTS
STRUCTURAL SYSTEM STUDIES	AEROSPACE VEHICLES					TIMELY IDENTIFICATION OF STRUCTURAL CONCEPTS AND TECHNOLOGY NEEDS FOR FUTURE VEHICLES
	HIGH SPEED CRUISE VEHICLES					
	AEROASSISTED ORBITAL TRANSFER VEHICLE					
CONCEPT DEVELOPMENT	PROPULSION STRUCTURES					ADVANCE CRITICAL TECHNOLOGIES FOR STRUCTURAL CONCEPTS
	AIRFRAME AND TANK STRUCTURES					
	THERMAL PROTECTION SYSTEMS					
ANALYTICAL METHODS AND APPLICATIONS	FINITE ELEMENT ANALYTICAL METHODS					PROVIDE IMPROVED ANALYSIS CAPABILITY & EFFICIENT STRUCTURE
	STRUCTURAL OPTIMIZATION ANALYSES					
THERMAL STRUCTURES EXPERIMENTS	DEFINITION OF NEEDS					PROVIDE IMPROVED TEST CAPABILITY FOR THERMAL STRUCTURES
	IMPROVEMENT TO APPARATUS					
	LABORATORY PLANNING					

Figure 45.



## STRUCTURAL SYSTEM STUDY OF AEROASSISTED ORBITAL TRANSFER VEHICLE

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Thermal Structures Branch  
Extension 4295

RTOP 506-53-43

Research Objective - Aeroassisted Orbital Transfer Vehicles (AOTV's) are being studied as a means of moving payloads between low Earth orbit and geosynchronous orbit. On return to low-Earth orbit the vehicle decelerates by passing through the upper atmosphere. The objective of this study is to identify an efficient AOTV structural system.

Approach - The structural system of a vehicle is strongly dependent on the configuration. Therefore, several configurations were investigated and the most promising was selected for more detailed study. For the selected configuration, an efficient structural arrangement and efficient wall constructions were identified and sized by thermal and structural analyses.

Accomplishment Description - The three main AOTV concepts considered in the study are shown in figure 46(c). The ballute consists of a cylindrical vehicle, which fits within the cargo bay of the Shuttle, and an inflatable fabric structure with flexible TPS to provide a low ballistic coefficient and thereby limit peak aerodynamic heating. Because the ballute has no lift, it must accommodate atmospheric density variations by deflating the fabric structure to modulate drag as indicated in the figure by dashed lines. The partially deflated ballute may be subject to flutter and localized overheating in the grooves between the resulting lobes.

The shaped lifting brake consists of a large stiffened shell covered with RSI tiles. Because the shell is 40 feet in diameter, it cannot fit in the Shuttle cargo bay and, therefore, the vehicle must be carried to orbit in subsections and assembled and checked out in space. The lift to drag ratio of 0.25 is sufficient to accommodate atmospheric density variations.

The slant-nose cylinder is a cylindrical vehicle with a blunt, slanted nose and will fit within the Shuttle cargo bay. This vehicle provides an enclosed payload bay to protect the payload from aerodynamic heating during the pass through the atmosphere. Two interchangeable payload bays are provided to accommodate various size payloads. The ballistic coefficient of this vehicle is higher than the other two vehicles and the heating on the nose is therefore higher. Current estimates indicate that the three vehicles weigh roughly the same.

Figure 46(a).

## STRUCTURAL SYSTEM STUDY OF AEROASSISTED ORBITAL TRANSFER VEHICLE

(Continued)

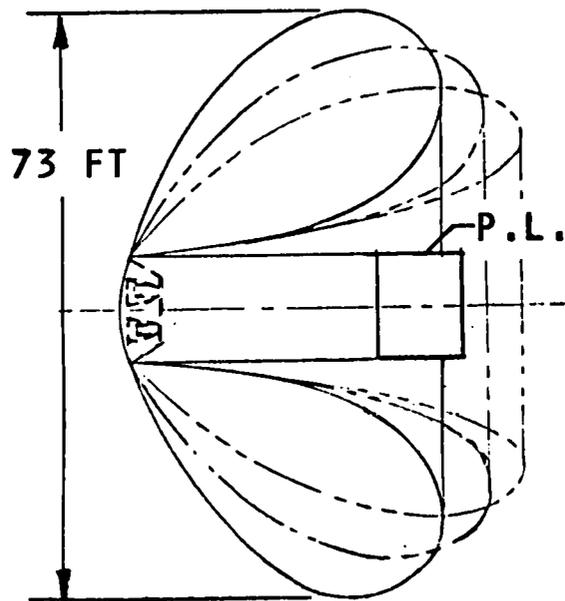
The simplest, most attractive configuration is the slant-nose cylinder. The most critical region of this vehicle is the nose which is exposed to high heating rates. Three passive, reusable nose wall constructions (fig. 46(c)) were identified. The first two maintain the carbon-carbon surface below the present reuse temperature of the coating (3000°F). The first concept provides a beryllium heat sink which allows the carbon-carbon heat shield to radiate from both sides, and the second concept uses a carbon-carbon heat shield thick enough to act as a heat sink. The third concept uses a higher temperature coating now under development which can accommodate the radiation equilibrium temperature. The first two concepts weigh the same, but the third is significantly lighter, which offers the potential to reduce the weight of the slant-nose cylinder vehicle by about 500 pounds.

The thermal responses of all three nose cap concepts are shown in figure 46(d).

Future Plans - The study will continue with a focus on reducing weight by using more advanced technologies, and detailed study of a flight experiment vehicle using the slant-nose cylinder configuration.

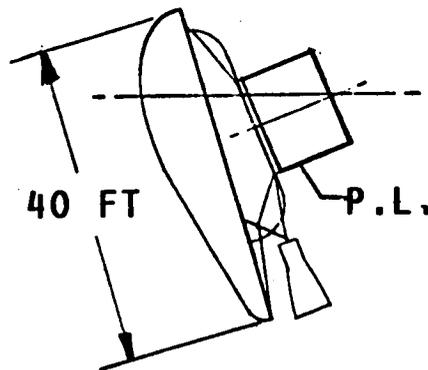
**AOTV CONCEPTS**  
 (RETURN PAYLOAD = 12000 LB, 14 FT. DIA. X 10 FT. LONG)

107



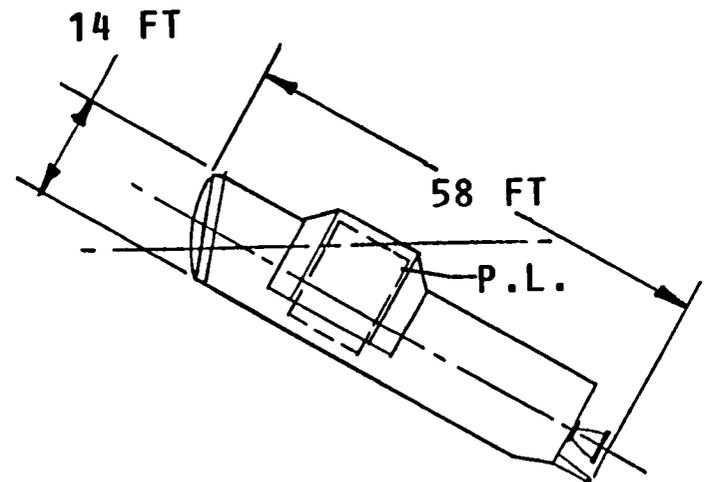
**BALLUTE**

L/D = 0  
 EMPTY WEIGHT=11200 LBS.  
 TPS = RSI & FLEXIBLE INSUL.  
 FEATURE=FLEXIBLE STRUCTURE



**SHAPED LIFTING BRAKE**

L/D = 0.25  
 12500 LBS.  
 RSI OR ACC  
 SPACE ASSEMBLY



**SLANT-NOSE CYLINDER**

L/D = 0.45  
 11500 LBS  
 ACC & RSI  
 GROUND ASSEMBLY  
 & REPLACEABLE  
 PAYLOAD BAY

Figure 46(c).

# WALL CONSTRUCTION CONCEPTS FOR AOTV NOSE

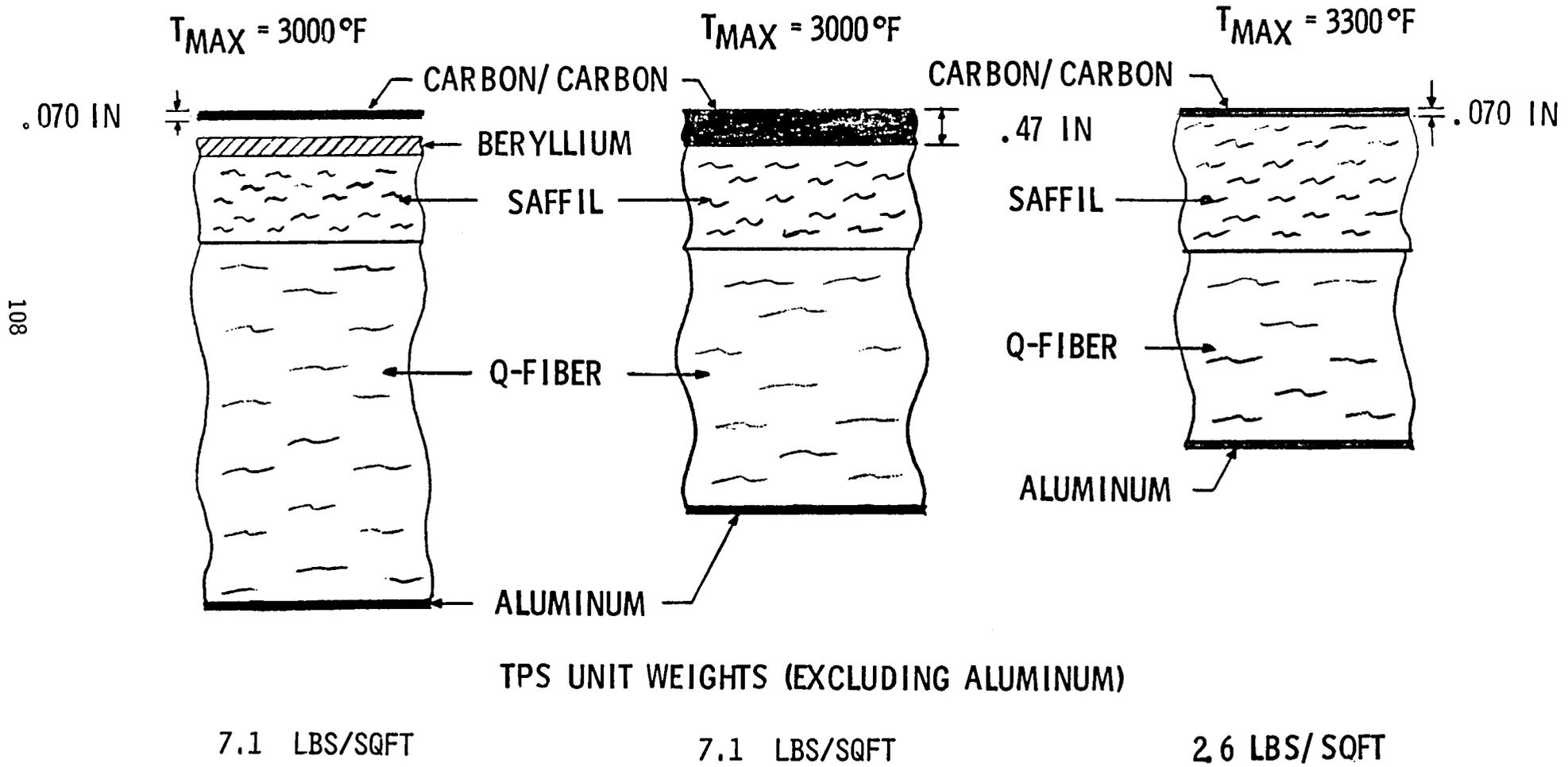


Figure 46(d).

108

# AOTV ATMOSPHERIC PASS CARBON-CARBON TEMPERATURES

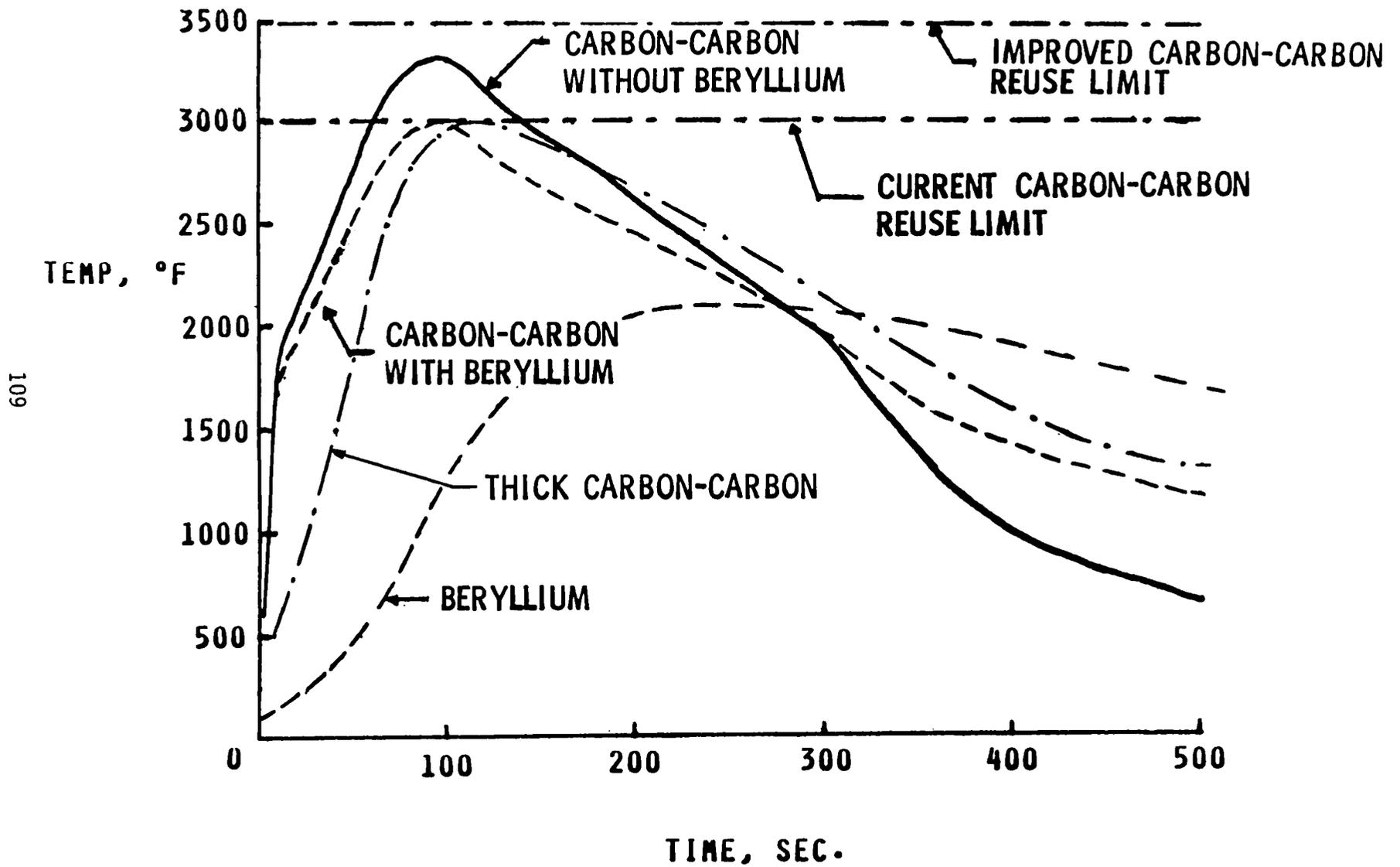


Figure 46(e).

## AEROTHERMAL AND ENVIRONMENT TESTS VERIFY TITANIUM AND SUPERALLOY TPS CONCEPTS

John L. Shideler  
Aerothermal Loads Branch  
Extension 2425

RTOP 506-53-33

Research Objective - Although the Reusable Surface Insulation (RSI) currently used on the Space Shuttle is an excellent insulation, it is very fragile. More durable metallic TPS concepts have been developed, and verification of these concepts by testing is required.

Approach - TPS panels of Titanium Multiwall (Ti M/W), which consists of alternate layers of flat and dimpled foil joined dimple to dimple, and Superalloy Honeycomb (SA/HC), which consists of an outer Inconel 617 honeycomb sandwich, fibrous insulation, and an inner titanium honeycomb sandwich, have been exposed to the types of tests listed in figure 47(b).

Accomplishment Description - In thermal/vacuum tests at JSC and KSC, panels have been subjected to combined surface temperature and pressure histories representative of Space Shuttle entry trajectories; results indicate acceptable and predictable thermal performance. Results from acoustic, vibration, and lightning strike tests indicate that the concepts can withstand severe dynamic and lightning environments while sustaining only minimal damage. The upper photograph in the figure shows a SA/HC 20-panel array in the Mach 7 aerothermal stream of the LaRC 8' HTT. The thermal deflections of the panels, which had a maximum surface temperature of about 1850°F, resulted in panel "pillowing" which caused slightly higher temperatures to occur on the upstream side than on the downstream side of the individual panels. Two coating materials with different emittances were applied to the array. The greater brightness (and higher temperature) of the right-hand side of the array is caused by the lower emittance of the panels on that side. Test results indicate that heating due to flow in the gaps between panels occurs when the gaps are parallel with the flow. Earlier aerothermal tests on a Ti M/W model have shown that gap heating does not occur when the gaps are 30° to the flow. Gap blockers will be installed in future test arrays to reduce gap flow.

The lower photograph shows 1st generation Ti M/W panels during environmental exposure tests designed to determine the water absorption/retention characteristics under actual rainfall conditions. Results of the KSC tests have shown that water absorption is not a problem for this concept.

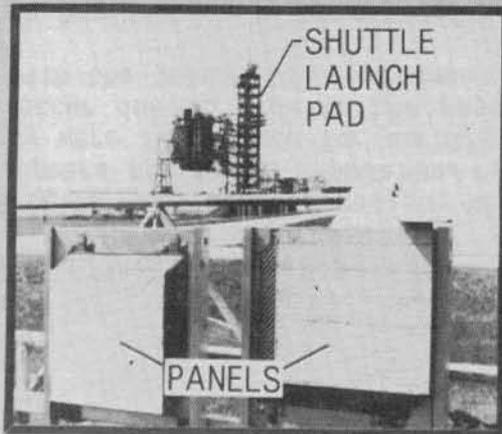
Future Plans - An array of curved SA/HC panels with gap blockers will be tested in the 8' HTT. Additional environmental tests including thermal cycles are planned to evaluate the long-term effects of repeated exposure to atmospheric/launch-pad contaminants and simulated mission thermal cycles.

Figure 47(a).

# AEROTHERMAL AND ENVIRONMENTAL TESTS VERIFY TITANIUM AND SUPERALLOY TPS CONCEPTS



SA/HC ARRAY IN 8ft HTT



Ti M/W PANEL EXPOSURE

## VERIFICATION TESTS

- THERMAL/VACUUM (JSC, KSG)
- ACOUSTIC/VIBRATION (JSC, LaRC)
- LIGHTNING STRIKE (LaRC)
- AEROTHERMAL (LaRC)
- ENVIRONMENTAL (KSG)

Figure 47(b).

## AEROTHERMAL TESTS OF METALLIC TPS

John L. Shideler and Granville L. Webb  
Thermal Structures Branch  
Extension 2425

RTOP 506-53-33

Research Objective - The objective is to determine aerothermal performance, particularly heating in the gaps between metallic TPS panels.

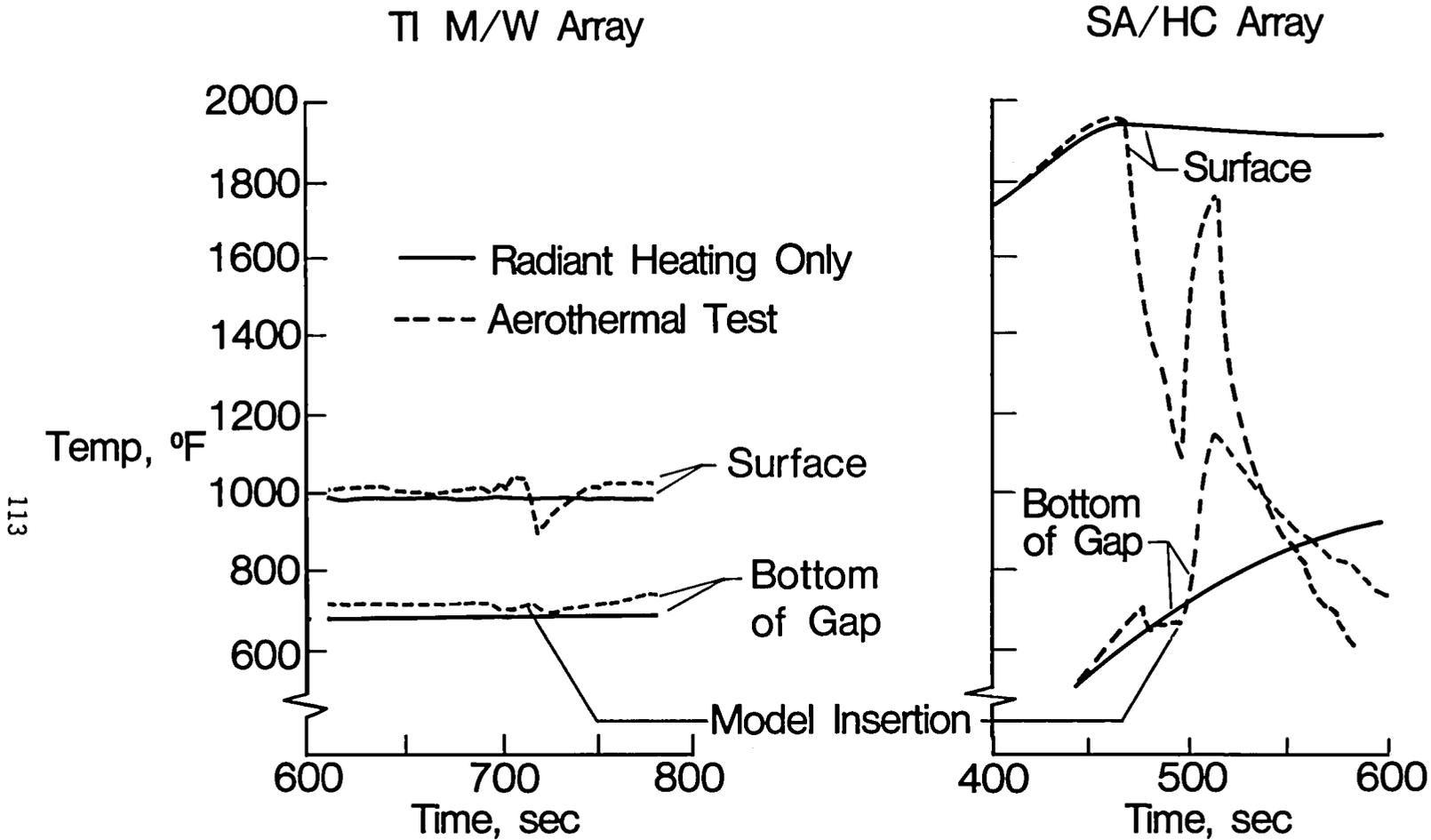
Approach - Results from earlier radiant heating tests indicate that the "heat short" effect at the edge of a panel appears not to be a problem. A titanium multiwall 20-panel array and a superalloy honeycomb 20-panel array were fabricated for aerothermal tests to evaluate the severity of any additional heating which might occur due to flow in the gaps between panels. The arrays were oriented in a "worst case" orientation with the intersections between panels running parallel to the direction of flow.

Accomplishment Description - Representative results from aerothermal tests in the LaRC 8' HTT are shown in figure 48(b) for each array. Surface temperatures and temperatures at the bottom of a gap are shown for a test with radiant heating only and for an aerothermal test. During the radiant heating tests, the surface was heated to the design temperature and held constant for about 200 seconds. While the surface temperature remained constant, the temperature at the bottom of the gap between titanium panels also remained constant, and the temperature at the bottom of the gap between superalloy panels gradually increased toward a constant steady-state value. During the aerothermal tests, after the surface was raised to the design temperature by the radiant heaters, the surface temperature decreased between the time the heaters were turned off and the time the array was inserted into the aerothermal stream, and then increased again after insertion. The temperature at the bottom of the gap between titanium panels was not affected by the aerothermal exposure, but the gap temperature between superalloy panels exceeded the temperature measured during the radiant-heating-only test by nearly 400°F. These results indicate that significant gap heating occurs between superalloy panels when the array is oriented with panel intersections parallel to the flow. The gap between superalloy panels is wider and deeper than the gap between titanium panels, and this increased gap size may be the cause of the aerothermal heating between superalloy panels.

Future Plans - Although aerothermal tests on flat metallic TPS are complete, aerothermal tests of curved superalloy panels are scheduled for FY 85. The panels will have "flow stoppers" located at the panel corners in the gaps. These flow stoppers consist of metal tabs which protrude across each panel intersection. Environmental exposure tests for flat panels will be completed in FY 85.

Figure 48(a).

# EFFECT OF AEROTHERMAL EXPOSURE ON GAP TEMPERATURE



113

Figure 48(b).



## ADVANCED CARBON-CARBON HEAT SHIELD RESEARCH

Granville L. Webb  
Thermal Structures Branch  
Extension 2425

RTOP 506-53-33

Research Objective - One of the most promising durable TPS concepts for application to the highly heated areas of future space transportation systems is advanced carbon-carbon (ACC). This material is a derivative of the reinforced carbon-carbon (RCC) material which is being used successfully on the Shuttle nose-cap and wing leading-edges. The RCC material has been modified to improve strength and oxidation resistance and renamed ACC. The ACC TPS concept consists of large overlapping ACC panels (approximately 3 ft. x 3 ft.) mounted on post supports with packaged fibrous insulation between the ACC panels and the main vehicle structure. The objective of this research is to develop high temperature durable TPS systems.

Approach - The ACC test article is shown in figure 49(b). The test article is composed of four panel segments, representing the intersection of four ACC multipost concept panels. The overall size of the test article is 1 ft. x 2 ft. The test article has been tested in radiant heating environments. It will be tested in an arc-tunnel environment in FY 84 & 85. Testing in these two environments should provide a good measure of the thermal efficiency of the heat shield and a comparison of the performance between the two environments should indicate whether hot gas flow through the panel joints is significant. Any weight change in the ACC panels due to high temperature exposure can also be determined.

Accomplishment Description - Figure 49(c) compares test data from a radiant heating test with data from an arc-tunnel test. The data suggest the existence of hot gas ingress through the overlapping joint down into the insulation joint. The tunnel condition resulted in a surface temperature as shown by thermocouple (TC) 1 that is nearly 100°F less than the temperature obtained during the radiant heating test. However, the temperature measured by TC 2 during the arc-tunnel test was not less than that obtained during the radiant heating test as would be expected. This suggests hot gas ingress through the joint, but the temperature at the bottom of the joint (TC 3) does not support the above conclusion.

Future Plans - Additional testing and analysis are continuing in an attempt to explain this discrepancy.

Figure 49(a).

NASA  
1-83-11.677

ACC MULTIPOST ARTICLE

TESTED AT 2300°F FOR:  
75 MIN., THERMAL VACUUM  
AND  
10 MIN., ARC TUNNEL

Figure 49(b).

# EFFECT OF AEROTHERMAL EXPOSURE ON GAP TEMPERATURE ACC MULTIPOST

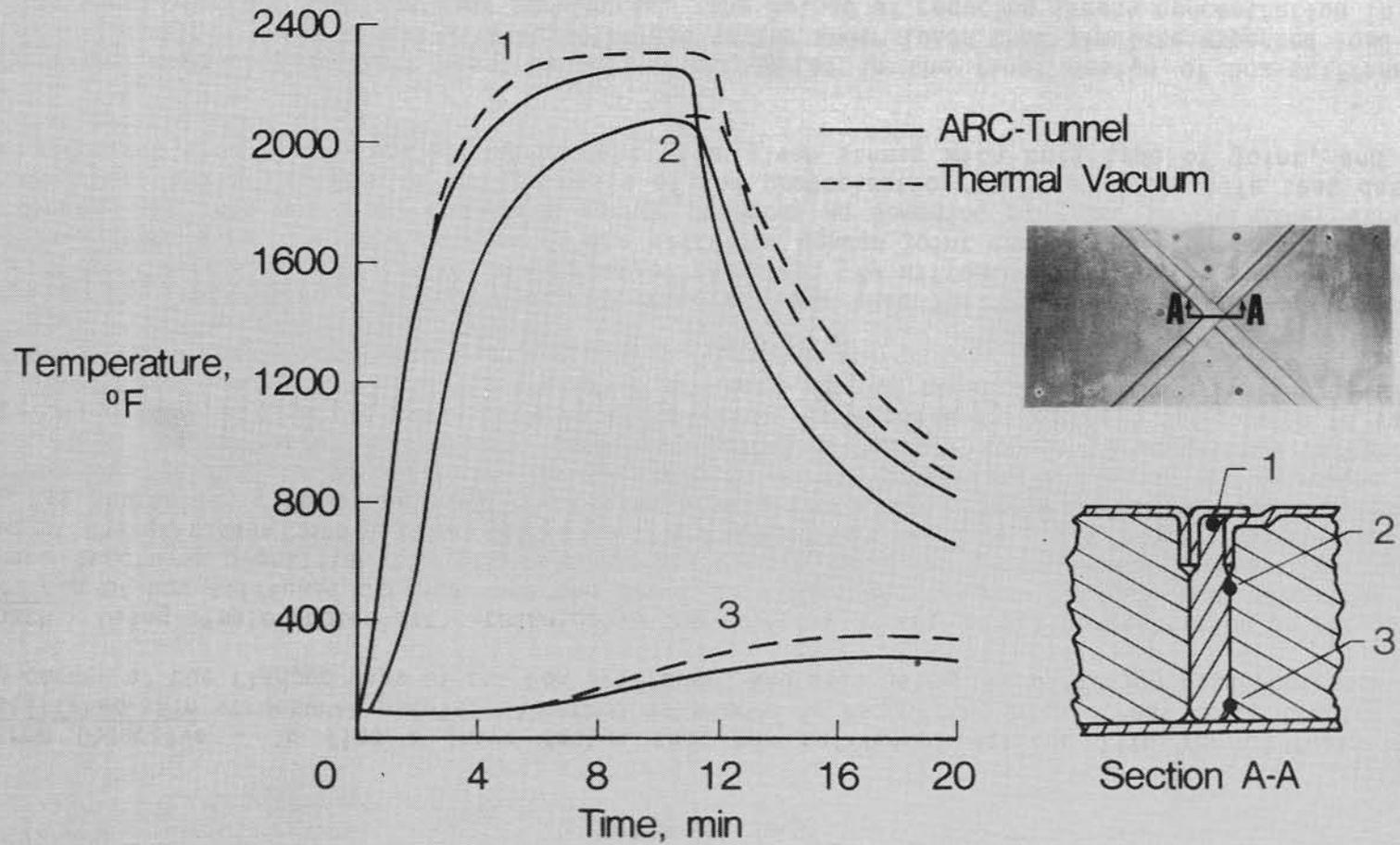


Figure 49(c).

## METHOD FOR REDUCING STRESS CONCENTRATION IN DIFFUSION BONDED JOINTS

Randall C. Davis  
Thermal Structures Branch  
Extension 2291

Paul L. Moses  
Kentron International  
865-1010

RTOP 505-33-53

Research Objective - To find a joint design that has sufficient fatigue life for diffusion bonded box-stiffened-skin structural panels. A method is needed to reduce the high stress concentration at the sharp corner of the flanged caps of the box stiffeners and skin using the diffusion bond process.

Approach - Using simple photoelastic techniques, the severity of the stress concentration at the junction of the cap of box stiffened stringer and the panel skin was identified. Fatigue test data from actual hardware specimens quantified the stress concentration in terms of cycle life at stress for the sharp corner of the diffusion bonded joint. As shown in figure 50(b), various joint concept were investigated using the photoelastic technique, and the better joints identified. These joints were made in titanium specimens and the fatigue tested to verify the results of the photoelastic study. As indicated in the figure, a diffusion bonded square-edge doubler produces a fatigue stress concentration factor  $K_T$  of about 7.0, a flanged-edge cap (the geometry of interest) produces a  $K_T$  of about 3.5, which is too high for aircraft application. A one-step chem-mill produces a  $K_T$  of about 1.5; however, a flanged-edge cap is necessary for panel stability.

Accomplishment Description - The photoelastic results show that for the one-step chem-milled skin, a region of zero stress is produced at the corner of the step. By diffusion bonding the stringer flange to the skin at this step, the sharp corner of the diffusion bonded joint can be positioned at this point of zero stress. At this position, the sharp corner produces no damaging stresses in the panel structure. Specimens were built in titanium on the basis of the photoelastic results. Cycle life test data from these specimens show a substantial improvement at a given stress with this type of joint, and the  $K_T$  has an acceptable value of 2.5.

Future Plans - Results of this research were incorporated in the final design of box-stiffened-skin compression panels. These panels will be built and tested under loads that simulate expected load cycles for high speed aircraft fuselage tank structures. The method of reducing stress concentration in diffusion bonded joints in box-stiffened panels can be used for other types of diffusion bonded construction.

Figure 50(a).

# CONCEPT IDENTIFIED FOR REDUCING STRESS CONCENTRATIONS IN DIFFUSION BONDED JOINTS

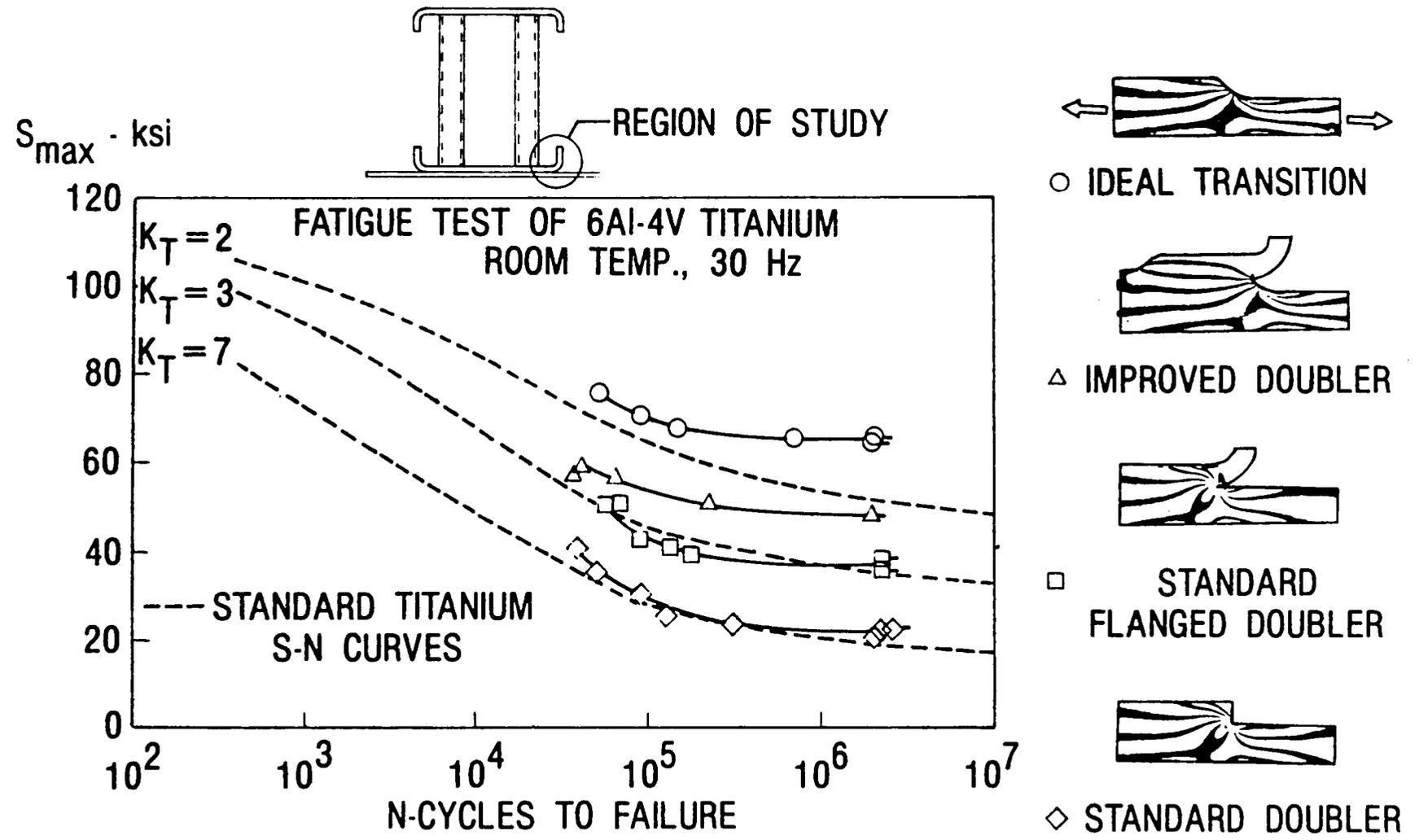


Figure 50(b).

PRELIMINARY TEST TO CRYOGENIC AND ELEVATED TEMPERATURES VERIFIES POTENTIAL  
OF NEW FOAM FOR REUSABLE CRYOGENIC INSULATION

Randall C. Davis  
Thermal Structures Branch  
Extension 2291

RTOP 505-33-53

Research Objective - Develop design concepts for low-mass reusable tanks for containment of cryogenic fluids suitable for use as fuselage/tank structure for advanced space transportation systems, space platforms and orbital transfer vehicles.

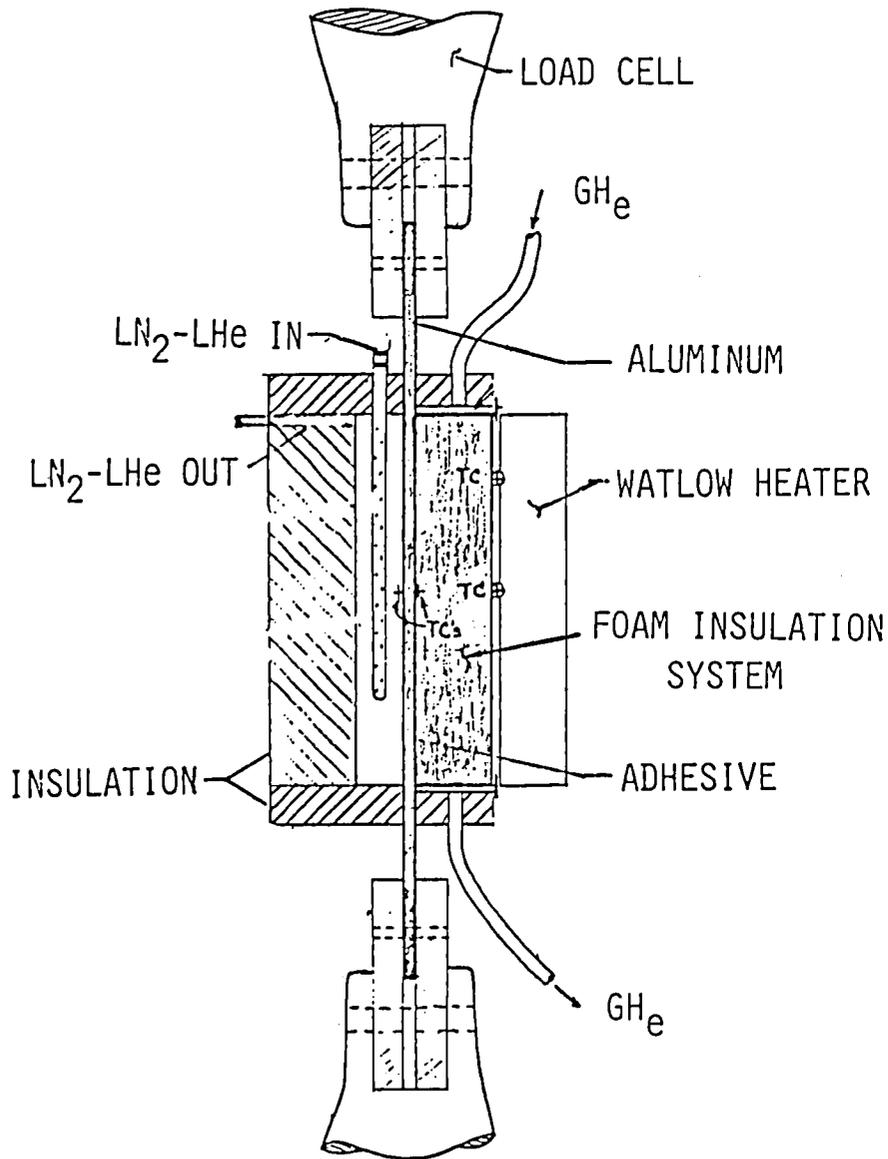
Approach - Identify candidate closed-cell foam insulation materials that would advance the state of the art in reusable cryogenic insulation. Design a simple inexpensive cryogenic testing facility and develop a procedure to test the ability of candidate materials to withstand the thermal gradients expected for each mission cycle life for cryogenic tanks on future space transportation systems. A Rohacell polymethacrylimide closed cell foam was identified as the best candidate for initial testing as an insulating material on a 2219 aluminum tank panel.

Accomplishment Description - A test setup was designed as shown in figure 51(b) that uses liquid nitrogen and liquid helium, thereby eliminating the expense and hazards usually associated with liquid hydrogen testing. Liquid nitrogen is used to precool the test chamber and liquid helium is then used to bring one side of the test panel down to the test temperature. Use of non-flammable nitrogen and helium permits the use of electric resistance heaters to heat the opposite side of the test specimen to duplicate mission cycle life temperatures. Test panels are exposed to thermal cycles with simultaneous application of cryogenic temperature and elevated temperature. One mission cycle is shown in the figure. The aluminum side is exposed to  $-423^{\circ}\text{F}$  while the foam side is heated to  $400^{\circ}\text{F}$  as shown. The first test panel survived 14 such cycles without damage due to the test temperature gradient before being dismantled for study.

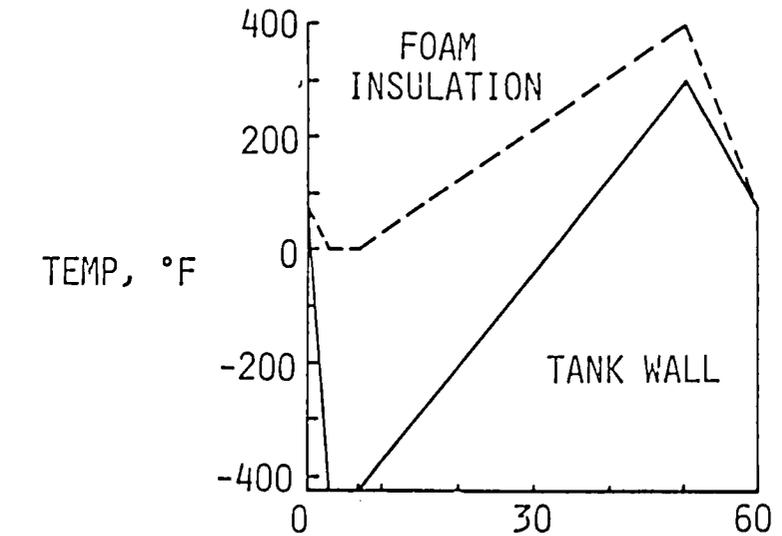
Future Plans - Combine tensile and thermal loading in the tests to demonstrate a 1000 cycle life capability for the new foam insulation system. A larger facility is planned for larger specimens to attain thermal conduction data with low edge effects. Alternate lower density foams are to be tested for improved mass efficiency.

Figure 51(a).

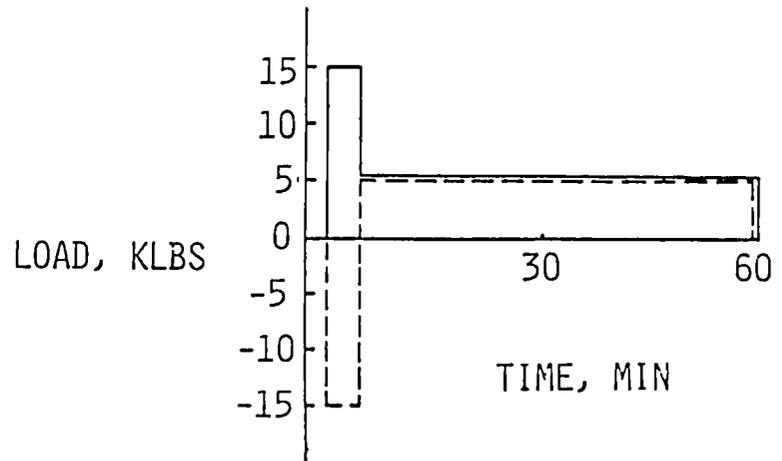
# HIGH TEMPERATURE CRYOGENIC FOAM CYCLIC TEST



TEST APPARATUS



● 14 THERMAL CYCLES



TEST PARAMETERS

Figure 51(b).

## CARBON-CARBON HOT STRUCTURE DESIGN

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Thermal Structures Branch  
Extension 4295

RTOP 506-53-33

Research Objective - Investigate the potential weight savings which can be achieved by replacing the insulated aluminum control surfaces of the Shuttle with carbon-carbon hot structure.

Approach - Develop a conceptual design of the Shuttle body flap using carbon-carbon hot structure. Identify and analyze all failure modes to obtain a credible design and weight estimate.

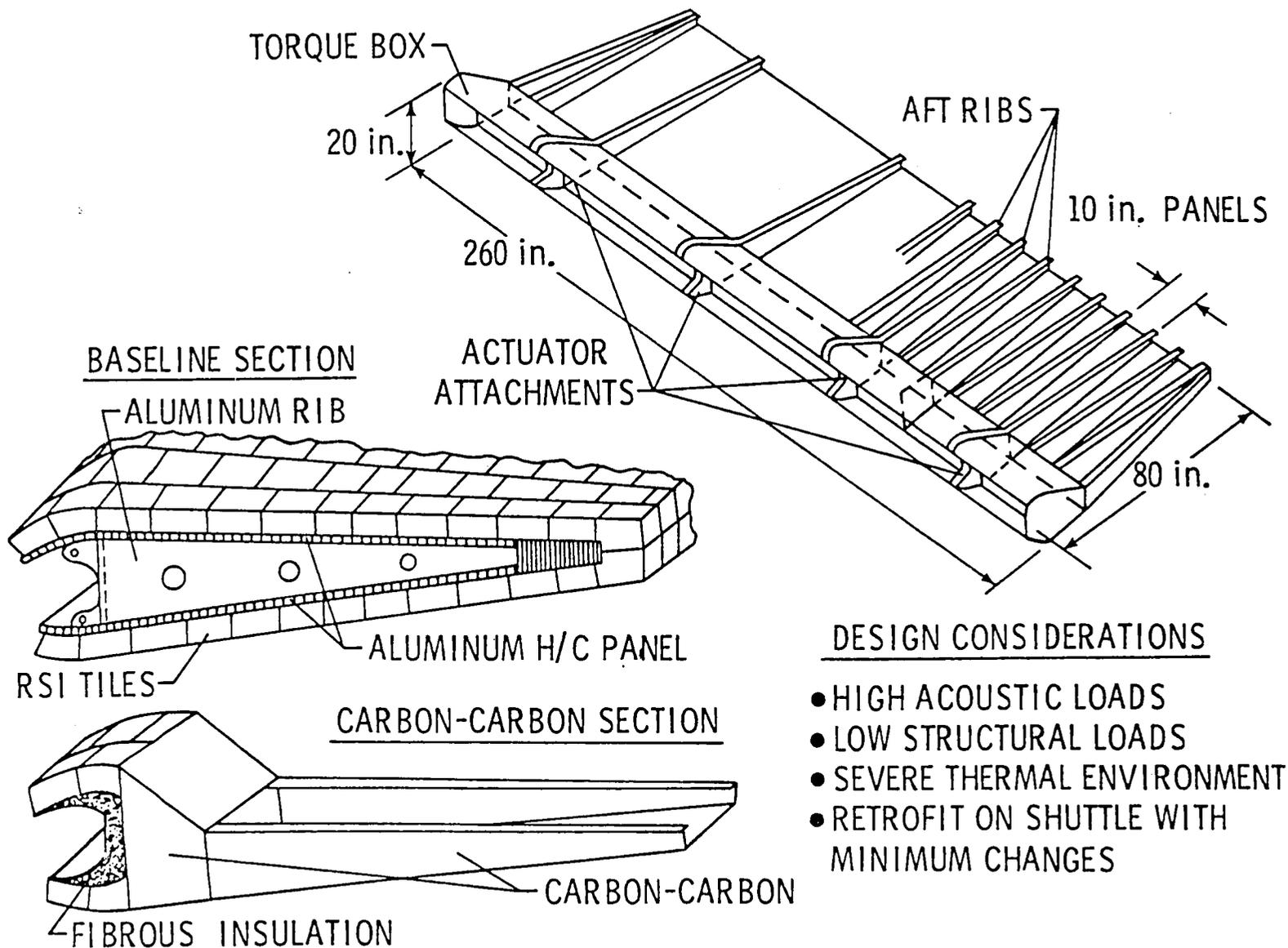
Accomplishment Description - A carbon-carbon hot structure design of a replacement body flap for the Shuttle Orbiter has been identified (figure 52(b)) and analyzed. The baseline body flap design, currently used on the orbiter, consists of upper and lower honeycomb-core panels which are supported by aluminum ribs every 20 inches and connected to a full depth honeycomb-core sandwich trailing edge. The aluminum structure is protected from entry heating on both the upper and lower surfaces by thick reusable surface insulation (RSI) tiles, which comprise the majority of the body flap weight.

The carbon-carbon hot structure design, which eliminates almost all of the RSI tiles, consists of a torque box and tapered ribs which support the continuous lower skin. Because the leading edge of the body flap is sealed, no significant amount of air flows over the upper surface and therefore, the upper skin was removed to save weight. Removal of the upper surface also allows more heat to be radiated from the lower surface, thereby reducing its peak temperature from 2700°F of the baseline body flap to below 2400°F. The key considerations for the body flap design include: high acoustic loads from the main engines during liftoff, low aerodynamic pressure loads, high entry heating, and ease of retrofitting the replacement body flap. Both static and dynamic analyses show the carbon-carbon design to be stiffer than the baseline body flap and, therefore, a more effective control surface. Only a small weight increase was required to accommodate the severe acoustic environment. Thermal stresses were found to be well below the material allowables, and the calculated weight indicates that this concept has the potential to save over half the weight of the existing body flap.

Future Plans - Additional thermal analysis of body flap and complete documentation of results of study are planned.

Figure 52(a).

# CONCEPTUAL DESIGN OF CARBON-CARBON BODY FLAP



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Figure 52(b).

## DERIVATION AND TEST OF ELEVATED TEMPERATURE THERMAL-STRESS-FREE FASTENER CONCEPT

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Thermal Structures Branch  
Extension 4201

RTOP 506-53-33

Research Objective: Develop a thermal-stress-free fastening technique that can be used to provide structurally tight joints at all temperatures even when the fastener and joined materials have different coefficients of thermal expansion.

Approach: Derive curves that are thermal-stress-free boundaries between various combinations of two materials with different coefficients of thermal expansion. Using the derived boundary shapes, generate fastener configurations that preserve the thermal-stress-free characteristics of the boundaries. Manufacture and test typical joints employing the thermal-stress-free fastener concepts.

Accomplishment Description: Thermal-stress-free boundary shapes were derived for various combinations of fastener materials and materials being joined together. The equation for the boundaries is given in figure 53(b). Fastener and joint shapes derived using the equation are also shown in the figure for a wide variety of fastener materials and materials being joined together. The value of  $P$ , defined by the coefficients of thermal expansion for the two materials, determined whether the fastener shape is convex, concave, or conical. The fastener configurations shown for  $P$  greater than zero appear practical. For values of  $P$  less than zero, the resulting fastener/joint configuration is rather inefficient. However, the joint is thermal-stress-free and may be the only configuration possible for certain combinations of materials. For two isotropic materials which may have large differences in coefficient of thermal expansion, the value of  $P$  is 1 and the resulting fastener has a conical shape. For some non-isotropic material combinations, the value of  $P$  is near 1 and the fastener shape approaches a cone. For those material combinations, approximating the thermal-stress-free shape by a conical shape should result in joints with low-thermal-stresses. Conical shaped fasteners that approximate the thermal-stress-free shapes were derived for an ODS metal fastener in carbon-carbon material. The resulting conical fastener/joint configuration was manufactured and tested in shear to failure. The results showed that the joint responded to mechanical loads similar to joints which used conventional fasteners. Strength estimates based on bearing area reasonably predicted the ultimate strength of the joint.

Future Plans: Additional tests are planned for mid 1985 to develop parametric data for use in designing low-thermal-stress joints in carbon-carbon.

Figure 53(a).

# TYPICAL-STRESS-FREE FASTENER CONFIGURATIONS

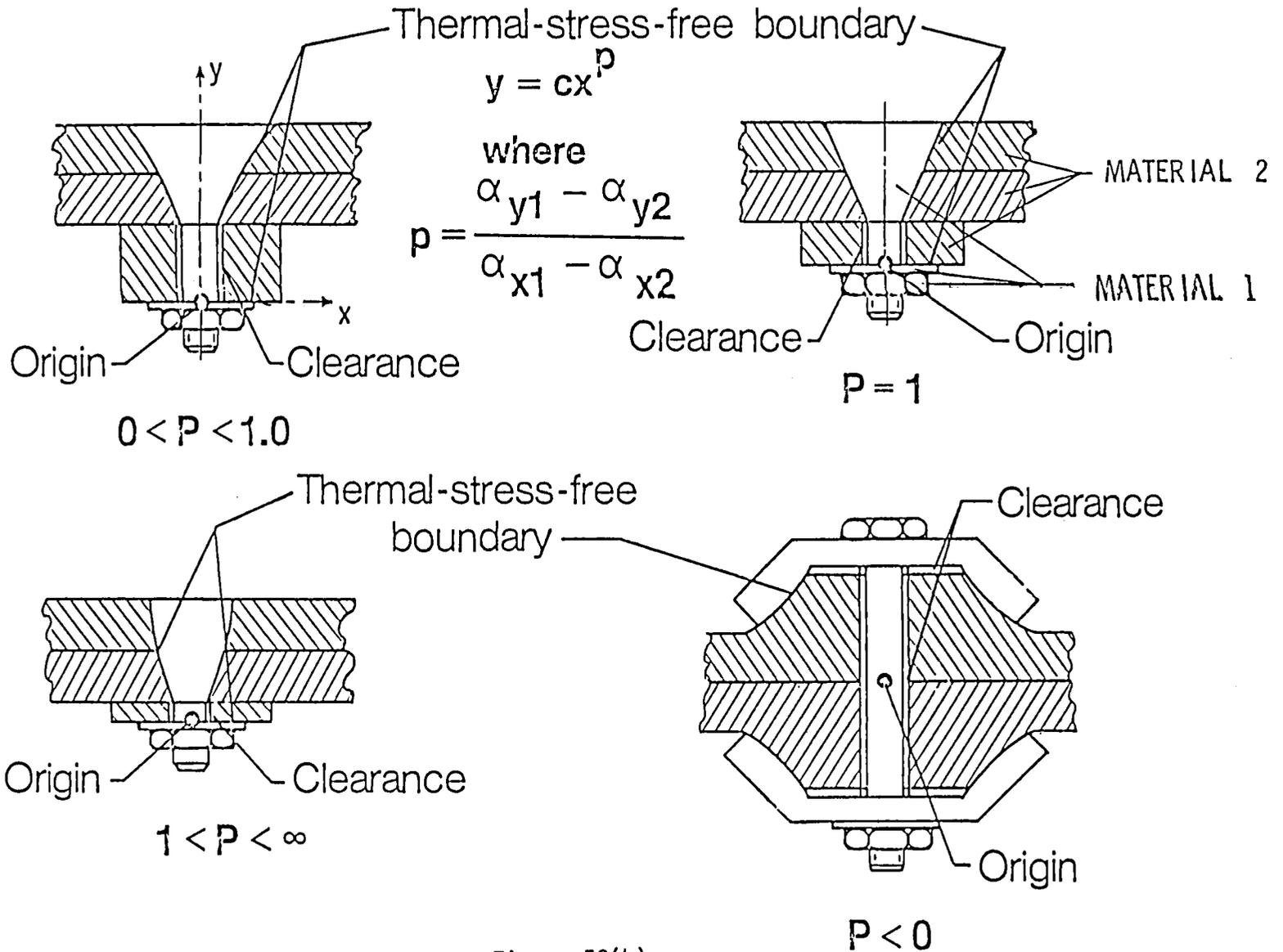


Figure 53(b).

## HIERARCHICAL INTEGRATED THERMAL/STRUCTURAL ANALYSIS

Allan R. Wieting  
Aerothermal Loads Branch  
Extension 3423

RTOP 506-53-33

Research Objective - Often a lack of compatibility exists between thermal and structural analyses because the finite element thermal model may require a different discretization than the finite element structural model. A method that can produce accurate results from a single geometric model can enhance both the speed and accuracy of thermo-mechanical analyses.

Approach - An integrated thermal/structural finite element methodology has been developed based on a common geometric model. A hierarchical finite element approach is used where the mesh is fixed, and solutions are covered by increasing the order of element interpolation functions. The approach (called the P method) converges faster than the usual approach of mesh refinement (the H method). The hierarchical finite element approach for integrated thermal/structural analysis seeks improvement in the effectiveness of the analyses by: (1) using hierarchical temperature interpolation functions to converge the thermal solution, (2) using the converged temperature distribution to compute improved structural loads, and (3) using hierarchical displacement interpolation functions to converge the structural solution. The results are that more accurate thermal stress solutions are obtained with less engineering time because only one geometric model is constructed.

Accomplishment Description - A simple application of the approach is demonstrated in figure 54(b). A more sophisticated application to a curved thermal protection system test panel is described in AIAA Paper No. 84-0939. These and other applications have demonstrated benefits of the method including the importance of having flexibility in refining thermal and structural analyses independently.

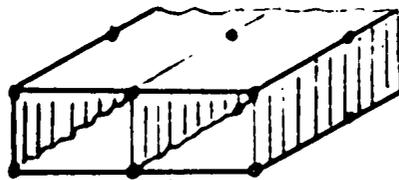
Future Plans - Extend the availability of the integrated thermal elements by their inclusion in the SPAR thermal analyzer. Evaluate the integrated thermal elements by solving practical heat transfer problems including multiple modes of heat transfer and thermal models of general three dimensional structures.

Figure 54(a).

# HIERARCHICAL INTEGRATED T-S ANALYSIS

## APPROACH

1. COMMON DISCRETIZATION TO SUIT GEOMETRY
2. THERMAL SOLUTION
  - SEQUENCE OF ANALYSES (IF NEEDED)
  - ACCURACY IMPROVED VIA THERMAL HIERARCHICAL ELEMENTS ( $P_T = 1, 2, \dots$ )
3. CONSISTENT THERMAL FORCES
4. STRUCTURAL SOLUTION
  - SEQUENCE OF ANALYSES (IF NEEDED)
  - ACCURACY IMPROVED VIA STRUCTURAL HIERARCHICAL ELEMENTS ( $P_S = 1, 2, \dots$ )



WING SECTION

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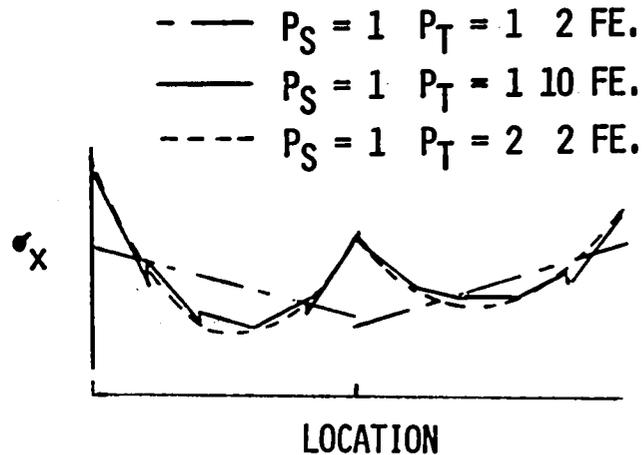
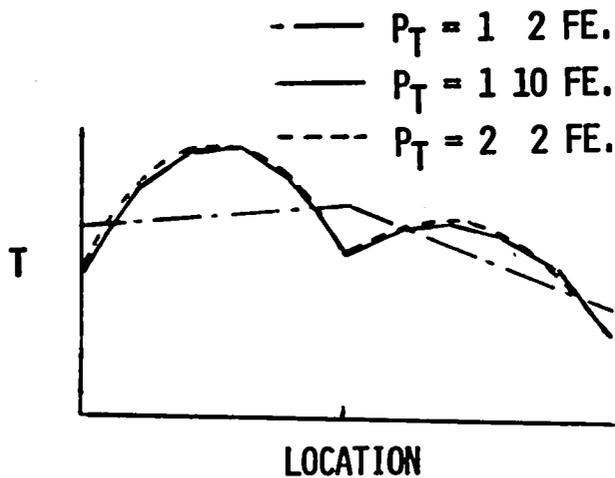


Figure 54(b).

## CONFIGURATION AEROELASTICITY

### FY 1985 PLANS

- 0 COMPLETE MODIFICATIONS TO TDT
- 0 COMPLETE DCP FLIGHT TEST PROGRAM
- 0 SUPPORT USAF IN FLUTTER CLEARANCE STUDY OF JAS-39 AIRPLANE
- 0 DESIGN, FABRICATE AND WIND-TUNNEL TEST IN TDT A SERIES OF BASELINE AND ADVANCED UH-60 MODEL ROTOR BLADES
- 0 BEGIN STUDY OF REGRESSION ANALYSIS TECHNIQUES APPLICABLE TO CONFORMABLE ROTOR DATA
- 0 COMPLETE DESIGN AND FABRICATION OF ARES II RIGID BODY SIMULATOR
- 0 INITIATE COOPERATIVE EFFORT WITH BELL FOR DESIGN AND TEST IN TDT OF A BEARINGLESS TAILORED ROTOR
- 0 FINALIZE THE DYNAMIC DESIGN OF A VARIABLE SPEED ROTOR
- 0 DOCUMENT HHC FLIGHT TEST PROGRAM
- 0 DOCUMENT JVX ANALYSIS/EXPERIMENT CORRELATION
- 0 COMPLETE AH-1G COUPLED ROTOR-AIRFRAME VIBRATIONS STUDY
- 0 COMPLETE AH-64 AND UH-60 FINITE ELEMENT MODELS AND GROUND VIBRATION TESTS
- 0 COMPLETE SHAKE TEST OF MODEL 360 COMPOSITE AIRFRAME

Figure 55.

PITCHING AND PLUNGING SUSPENSION SYSTEM  
FOR 2-D TRANSONIC FLUTTER TESTING

Maynard Sandford  
Configuration Aeroelasticity Branch  
Extension 2661

RTOP 505-33-43

Research Objective - The objective of this effort is to build an apparatus to enable systematic study of transonic flutter of lifting and nonlifting two-dimensional wings in the Langley 6-by 28-inch Transonic Blowdown Tunnel.

Approach - Simple analogs of physical systems are frequently used in engineering to simulate complex processes with simpler ones. Two-dimensional airfoil models have been used for static tests since the beginning of aeronautics. However, mechanisms to simulate wing flutter with two-dimensional airfoil sections have been conspicuously absent. This is due to the difficulty of implementing the linear and rotational springs required to simulate the plunging and pitching motions while maintaining realistically lightweight moving masses. The approach to be used in this effort uses a novel compound spring system to keep spring deflections within reasonable limits while generating the large forces needed to counteract the transonic airloads. This will allow the effect of steady angle of attack upon the flutter boundary to be studied. The spring system will also allow the pitch and plunge frequencies to be varied over a reasonably wide range. In addition, the pivot location for the pitching motion will be adjustable so that motions similar to those of swept-back wingtips may be simulated.

Status/Plans - Fabrication of the system will be completed about mid year. The system will be checked out initially in the laboratory and then in the Langley 6-by 19-inch Transonic Tunnel where the installation will be similar to that in the 6-by 28-inch tunnel where the actual flutter tests will be conducted later. The smaller tunnel will be used for checkout because it is not as heavily scheduled as the larger tunnel. The fabrication of two lightweight models to supplement two heavier models on loan from the U.S. Air Force will be completed this year.

Figure 56.



MODIFICATIONS TO UPGRADE THE LANGLEY TRANSONIC DYNAMICS TUNNEL  
(DENSITY INCREASE)

Bryce M. Kepley  
Configuration Aeroelasticity Branch  
Extension 2661

Background - The TDT is designed for and dedicated to studies and tests in the field of aeroelasticity and has special features which make it a national resource for flutter and buffet tests. Some of these features are its large 16 ft. x 16 ft. subsonic-transonic flow slotted test section, the ability to use dense Freon-12 gas or air as a test medium, a computer-controlled data acquisition system specifically designed to handle large quantities of dynamic data in near real time, special model mounting systems, a gust generator, a "flutter-stopper," safety screens upstream of the tunnel fan, and good model visibility while testing. The facility is used to verify the flutter and aeroelastic characteristics of most U.S. high-speed aircraft designs; for rotorcraft and active controls research; for flutter, buffet, and ground-wind loads tests of the Space Shuttle and other launch vehicles; and for confirmation of unsteady transonic flow theory. The increased density capability is needed chiefly for development testing involving the flutter clearance and validation of the flutter characteristics of high-speed aircraft and space vehicles such as the Shuttle. Models of these aircraft must be dynamically and aeroelastically scaled if the tests are to be valid. In addition to simulating the external shape, the stiffness, and stiffness distribution, these models must also simulate the mass density ratio which is the ratio of the distributed mass of the vehicle to the mass of the flight medium surrounding it. As airplanes become lighter (more structurally efficient), as with the use of composite major structures, or as they incorporate the use of active controls (which means the models have to employ relatively heavy active control hydraulic systems internally), it becomes increasingly difficult to fabricate models which are light enough to match full-scale mass-density ratios with current TDT density capability.

Approach - This FY 83 C of F project will provide for increasing the maximum test density by 50 percent in the Mach number range from 0.6 to 1.2 as shown by figure 57(b). The increased density capability will be provided by rewinding the existing fan motor to increase the power rating from 20,000 hp to 30,000 hp. Additional tunnel cooling capacity will be provided to accommodate the increased tunnel power limit. Other major modifications include changes to the electrical power distribution system and installation of a new speed control system.

Status/Plans - The design for the modifications, which was developed under contract by DSMA Engineering Corporation, provides for dividing the main aspects of work into a series of independent work packages to be performed by separate contractors. Contracts have been awarded for all seven of these work packages; these contracts encompass the major portions of the work to be done, namely, increasing the fan horsepower, installing new cooling system, and modifying electrical distribution system (figure 57(c)). The modifications are expected to be completed in early 1985, followed by a three month checkout period. The TDT is expected to be fully operational with the increased density capability in June 1985.

Figure 57(a).

## TDT DENSITY INCREASE



SCHEDULE					
	1981	1982	1983	1984	1985
DESIGN	██████████				
PROCUREMENT		██████████			
CONSTRUCTION			████████████████████		
TUNNEL SHUTDOWN				██████████	
CHECKOUT					██████████

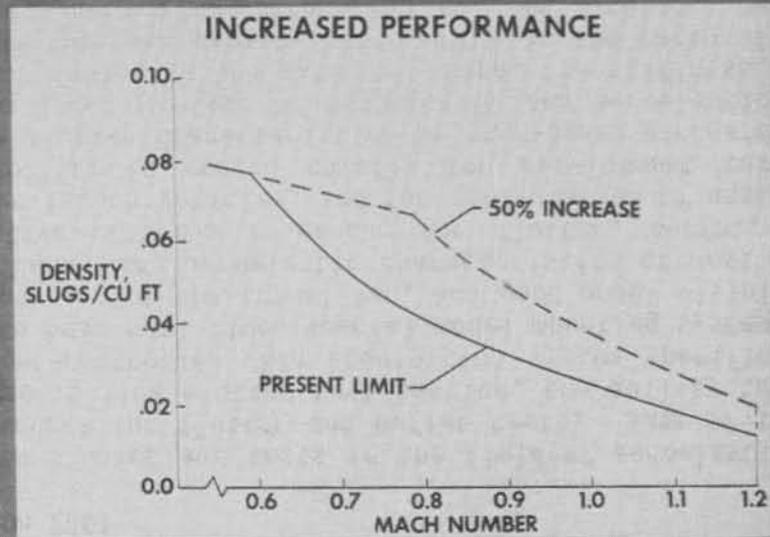
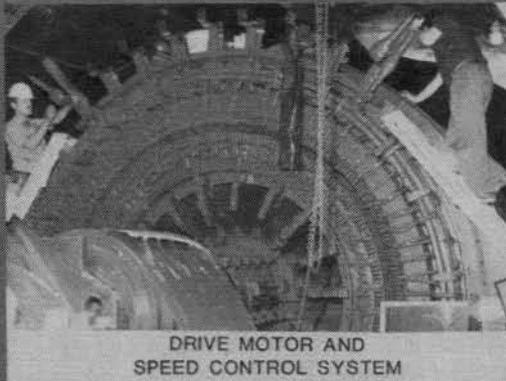
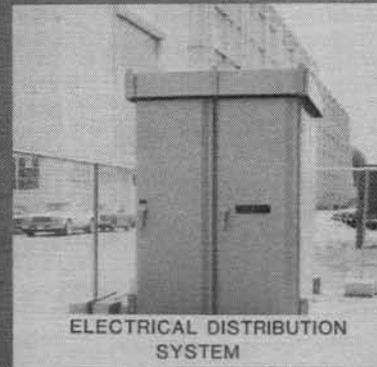


Figure 57(b).

**MODIFICATIONS TO UPGRADE  
LANGLEY TRANSONIC DYNAMICS TUNNEL ON SCHEDULE**



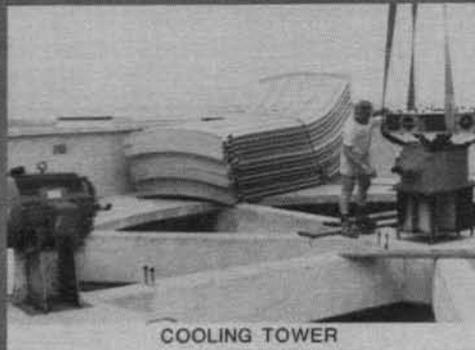
DRIVE MOTOR AND  
SPEED CONTROL SYSTEM



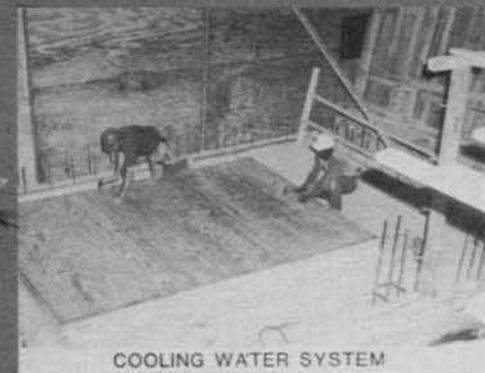
ELECTRICAL DISTRIBUTION  
SYSTEM



TUNNEL COOLING COIL



COOLING TOWER



COOLING WATER SYSTEM

Figure 57(c).



## UPGRADING THE DATA ACQUISITION SYSTEM FOR THE LANGLEY TRANSONIC DYNAMICS TUNNEL

Bryce M. Kepley  
Configuration Aeroelasticity Branch  
Extension 2661

Objective - The objective is to increase the productivity of the Langley Transonic Dynamics Tunnel (TDT) by replacing the existing computer-controlled data acquisition, display and control system with a new system that takes advantage of technological advances that have been made since the original system was designed over ten years ago. The new system will provide increased reliability, more flexibility, more data channels, faster data rates, and enhanced real time analysis as compared to the present system.

Approach - The conceptual design of the new system has been developed by a Langley in-house team. A simplified block diagram of the system's configuration is shown in figure 58(b). The three computer systems will provide the flexibility needed to perform multiple tasks. The two analog "front end" systems provide for support of multiple tests. The new system supports tests in the General Rotor Aeroelastic Laboratory (hover facility) adjacent to the TDT building. This facility is not connected to the present system.

Status/Plans - The functional requirements for the system have been defined and documented. The top level software design is nearing completion (figure 58(c)). The detailed software design specifications will be completed this year. Most of the hardware has been ordered. Delivery of all major equipment items will occur this year. The setup of the hardware in the TDT building will be completed by the end of the year, but the system is not expected to be fully operational until 1986.

Figure 58(a).

# BLOCK DIAGRAM OF NEW DAS FOR TDT

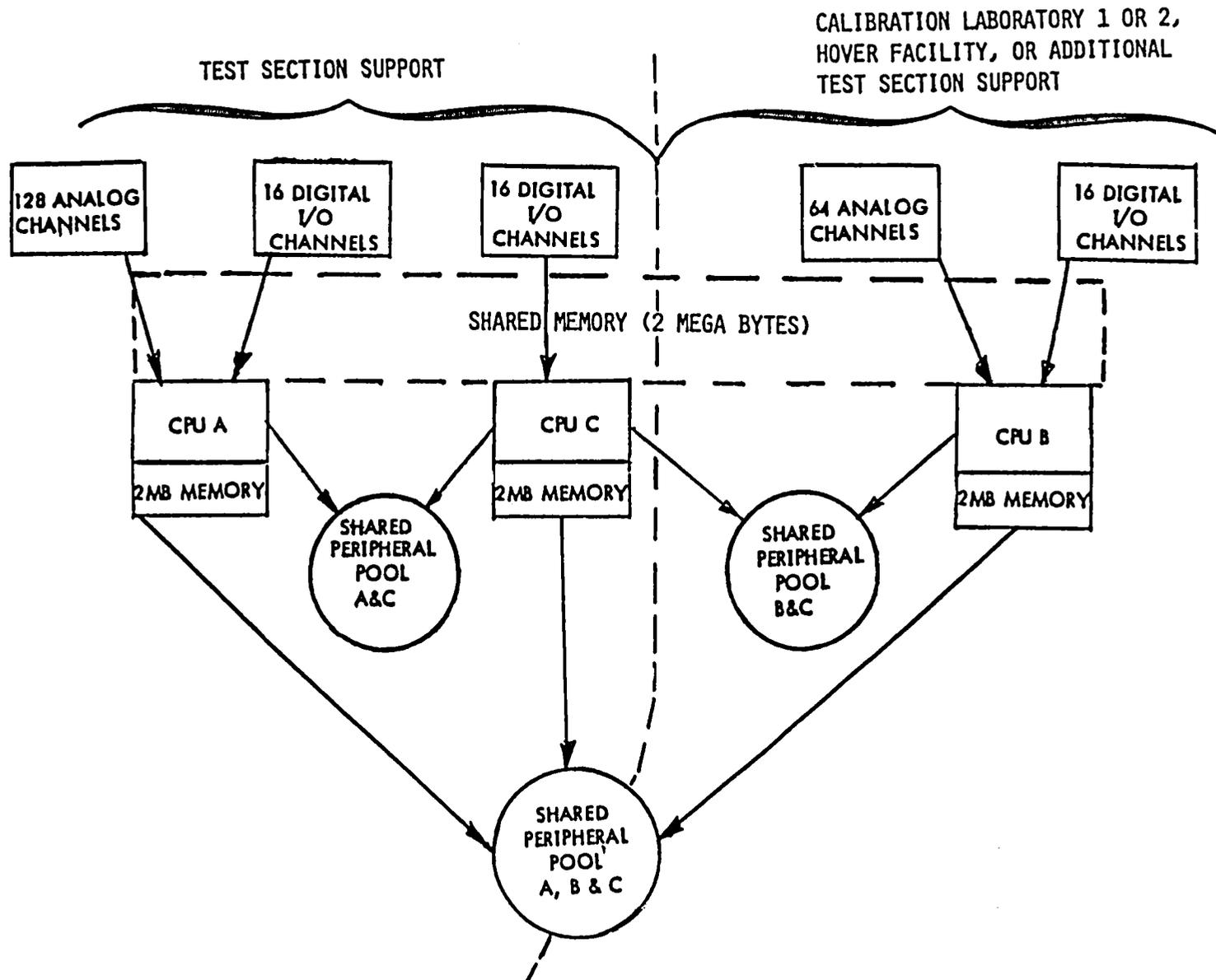
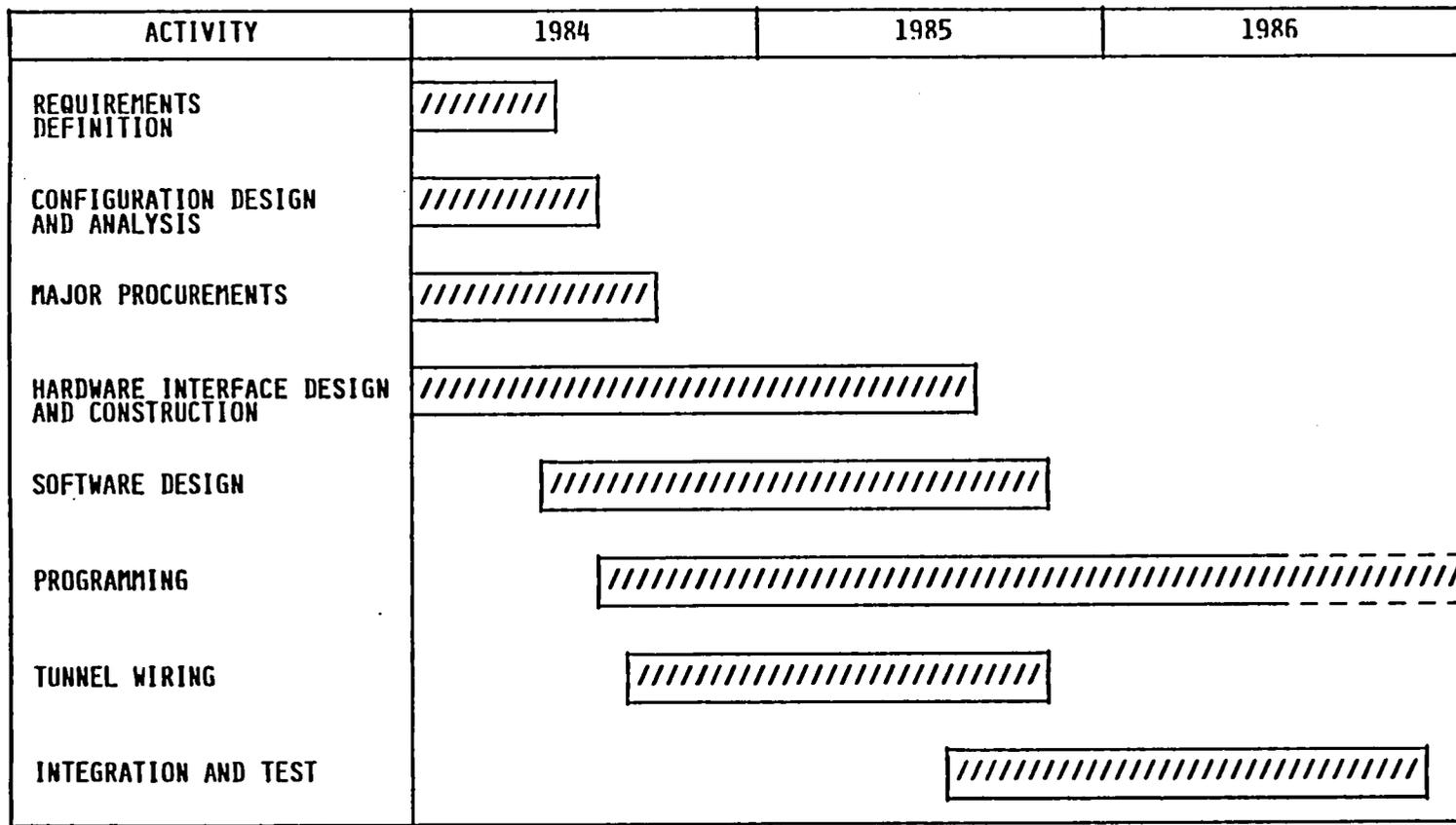


Figure 58(b).

IDT DATA ACQUISITION AND ANALYSIS SYSTEM PROJECT SCHEDULE



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Figure 58(c).

## ROTORCRAFT DYNAMICS AND AEROELASTICITY

Wayne R. Mantay  
Configuration Aeroelasticity Branch  
Extension 2661

RTOP 505-42-23

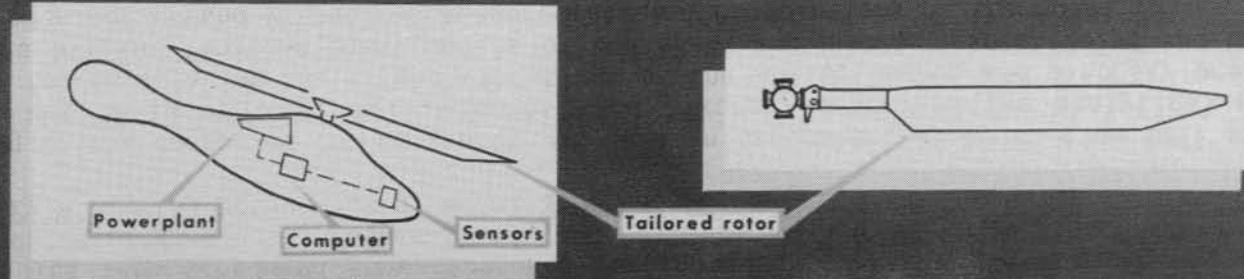
Research Objectives - The triad of research objectives in this area are (1) to conduct research in aeroelasticity, aerodynamics and dynamics of rotors; (2) to support design of advanced performance helicopters in the areas of loads, vibration and aeroelastic stability; and (3) to develop the experimental and analytical techniques necessary to extend TDT capabilities to future research opportunities, such as bearingless rotors.

Approach - This research area is a joint effort of the Loads and Aeroelasticity Division and the U.S. Army Structures Laboratory which is co-located at the NASA Langley Research Center. The work is a combination of experimental studies, tests in the TDT and the General Rotor Aeroelastic Laboratory (hover facility), and analytical studies. The Aeroelastic Rotor Experimental System (ARES) is a key test bed in the experimental studies. The in-house civil service research is supported and supplemented by industry contracts and university grants.

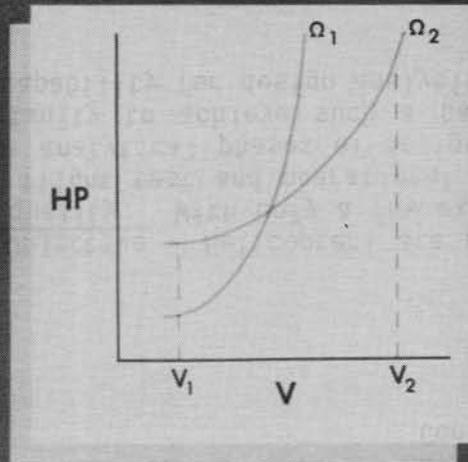
Status/Plans - Research during this year will address a variety of topics such as aeroelastic stability of hingeless rotors, rotor gust response, rotor optimization evaluations, advanced rotor track and balance characteristics, and a variable speed rotor concept. Figure 59(b) illustrates one of the projects, namely, the variable speed rotor. In this effort advanced analytical methods are being used to make a dynamic design of a rotor for improved hover and forward flight operation, utilizing substantial rpm changes in flight. The hingeless system design will be compatible with unique aerodynamic constraints and will utilize advanced analyses methods for aeroelastic tailoring.

Figure 59(a).

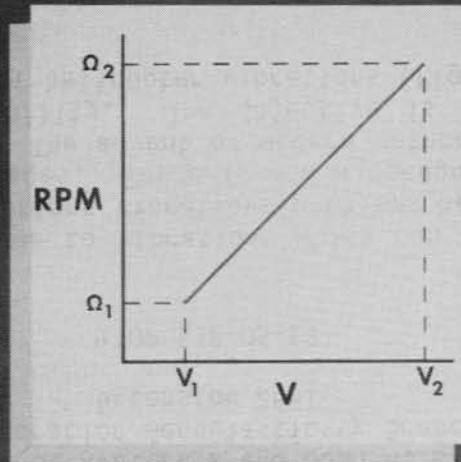
# AERODYNAMICALLY AND DYNAMICALLY ADVANCED MULTI-SPEED (ADAM) ROTOR



Power Savings Potential



Operating Schedule for  
Minimum Power



Modal Tailoring

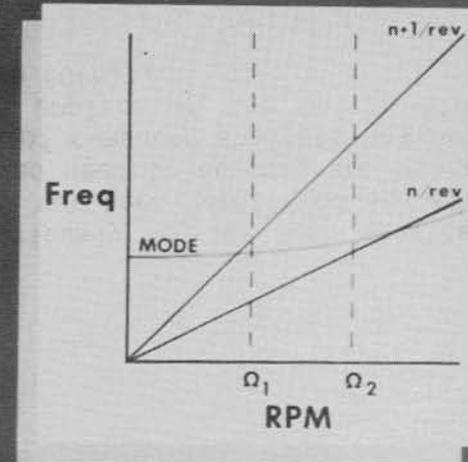


Figure 59(b).

# A NATIONAL CAPABILITY TO ANALYZE VIBRATION AS PART OF HELICOPTER STRUCTURAL DESIGN

Raymond G. Kvaternik and John H. Cline  
Configuration Aeroelasticity Branch  
Extension 2661

RTOP 532-06-13

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. 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 emplace in the United States a superior capability for design analysis of helicopter vibrations (figure 60(b)).

Approach/Status/Plans - As a culmination of considerable planning and coordination work by NASA and the Helicopter Industry, contracts have been issued under which industry teams will carry out analysis, measurement, correlation, and mutual critique procedures designed to develop confidence in finite-element analysis methods to predict vibrations of helicopter airframes. The approach follows the recently completed project at Boeing Vertol where the analysis subject was the CH-47D airframe. In addition, the program will use the AH-1G finite-element model developed by Bell a number of years ago and take advantage of results from scheduled Army sponsored developmental vibration tests. The airframes to be treated are: Metal Airframes: (1) Hughes AH-64, and (2) Sikorsky UH-60; Composites Airframes: (3) Bell ACAP, (4) Sikorsky ACAP, (5) Boeing Vertol Model 360, and (6) Hughes 500D Composite Version.

The first meeting of the companies for critiques was held in September, at which time Bell presented the data describing the AG-1G vehicle; Bell, Hughes, and Sikorsky described the analytical methods they intend to employ for calculating the flight vibrations of the AH-1G; Hughes and Sikorsky presented their plans for forming NASTRAN finite-element models of the AH-64 and UH-60, respectively; and Boeing Vertol presented their plan for ground vibrations measurements and correlations of the Model 360.

The following tasks will be completed this year: (1) analysis/experiment correlation for coupled rotor-airframe flight vibrations analysis of AH-1G, (2) AH-64 and UH-60 finite element models, (3) AH-64 and UH-60 ground vibration tests, and (4) shake test of model 360 composite airframe.

Figure 60(a).

DESIGN ANALYSIS METHODS FOR VIBRATIONS (DAMVIBS) PROGRAM

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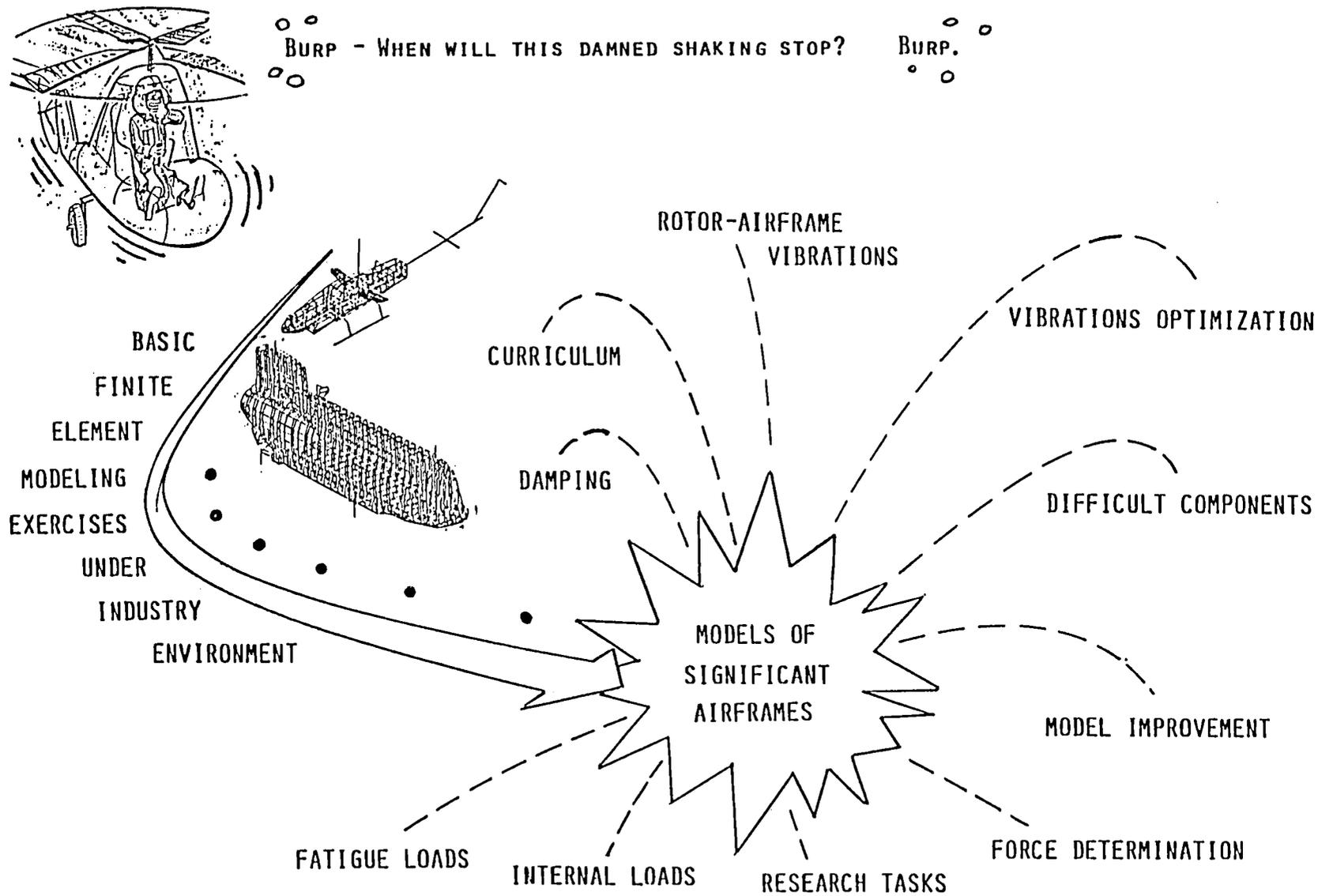


Figure 60(b).



## UNSTEADY AERODYNAMICS

### FY 85 PLANS

- 0 APPLICATION AND AUGMENTATION OF XTRAN CODES
  - COMPARISONS WITH UNSTEADY PRESSURE DATA BASES
  - MULTIPLE SURFACE INTERFERENCE CAPABILITY - CANARD/WING
  - COOPERATIVE AGREEMENTS WITH AIR FORCE AND INDUSTRY
  - ACCURATE, EFFICIENT VISCOUS BOUNDARY LAYER MODEL
  - COMPARISON WITH MEASURED TRANSONIC FLUTTER BOUNDARIES
  - RE EFFECTS FROM 0.3M TCT DATA
  
- 0 INITIATE DEVELOPMENT OF TIME-ACCURATE EULER CODE
  - DYNAMIC GRID MOTION EFFECTS
  
- 0 CONTINUE STUDY OF TRANSONIC INSTABILITY WITH DAST ARW-2 TEST IN TDT
  
- 0 CONTINUE IMPLEMENTATION OF SUPERSONIC INTEGRAL EQUATION METHOD
  
- 0 CONTINUE DEVELOPMENT OF ROTOR UNSTEADY AERO THEORY

Figure 61.

## DEVELOPMENT AND APPLICATION OF XTRAN3S

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Unsteady Aerodynamics Branch  
Extension 4236

RTOP 505-33-43

Research Objective - The objective of this research is to assess the accuracy of the XTRAN3S computer code and to extend its geometrical modeling capability to enable calculations for realistic aircraft configurations.

Approach - The XTRAN3S code solves the unsteady three-dimensional transonic small disturbance equation. The coupled aerodynamic and structural equations may be numerically solved simultaneously in time such that complete aeroelastic transients are generated to determine flutter characteristics. The original code only allowed calculations for isolated wings and had a severe stability restriction with respect to wing sweep and taper. A modified grid transformation has been incorporated which significantly alleviates the latter problem. Extensions of the geometrical modeling ability of the code are underway with the capability to treat multiple surfaces (e.g. canard-wing) already incorporated. It is planned to continue to add modeling capability such that the effect of the fuselage and engine mounted components (pylons, nacelles, stores) may be treated. The strip boundary layer capability of the code will be upgraded with recent developments which enhance efficiency and accuracy. Second order terms will be added to the code which will enable strong shock cases to be accurately calculated and which resolve nonuniqueness problems.

Status/Plans - Comparisons of XTRAN3S calculations with experimental unsteady pressures obtained in Langley's oscillating pressure model program are continuing. Calculations will be made for the clipped delta wing model, the ACEE wing model, the rectangular supercritical wing model and for the DAST ARW-2 model. The latter calculations are of particular interest due to the transonic instability encountered in the tunnel test. In addition, calculations for an AGARD tailplane model tested by the RAE are underway in conjunction with a NASA/RAE cooperative agreement. Cooperative efforts with industry and the Air Force are also ongoing to allow the user community to gain experience with the capability of the code. Finally, aeroelastic stability of canard-wing configurations will be studied.

Figure 62.

## UNSTEADY FULL POTENTIAL CALCULATIONS USING A FLUX DIFFERENCING METHOD

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Unsteady Aerodynamics Branch  
Extension 4236

Mohamed M. Hafez  
Computer Dynamics Incorporated  
Extension 2627

RTOP 505-33-43

Research Objective - The objective of this effort is to develop a more accurate method for predicting transonic unsteady aerodynamic loads.

Approach - The objective will be accomplished by using finite difference methods to solve the conservative unsteady full potential equation. A monotone differencing method, based on differencing the flux function, is used to discretize the flow equations. This method does not allow nonphysical expansion shock waves to be computed as part of the numerical solution. Calculated shock waves have a width of no more than two grid points.

Status/Plans - The flux differencing method has been incorporated into a computer program for solving the full potential equation on a body-fitted grid. Results for steady flows have been obtained using an AF1 iteration technique. The structure of the AF1 method makes it relatively easy to vectorize. For transonic flows, shock waves calculated using the flux differencing method are sharper than those calculated using a density biasing method. Additional terms, necessary to calculate unsteady flows, have been added to the analysis. Check out of the program for calculating unsteady flow is underway.

Figure 63.

## SUPERSONIC INTEGRAL EQUATION CODE

Robert N. Desmarais  
Unsteady Aerodynamics Branch  
Extension 4236

RTOP 505-33-43

Research Objective - The objective is to develop a robust code for computing supersonic unsteady aerodynamic forces. This code will encompass the same flight regime as the Mach box method but will not have the innacuracy and numerical difficulties of the Mach box method.

Approach - The intent is to complete the development of the supersonic Green's function algorithm developed by Freidman and Tseng of Boston University. This algorithm appears to extract all relevant singularities correctly. It should result in a program that is economical to use and that is accurate within the limitations of linear supersonic aerodynamic theory.

Status/Plans - The code will be used to compute flutter speeds and frequencies for some configurations for which experimental results are available. It will also be used to compute aerodynamic forces for comparison with flat plate theory, the low supersonic kernel function method, and the Mach box method. If funding permits, it will also be compared with XTRAN3.

Figure 64.

## INVESTIGATION OF DAST ARW-2 TRANSONIC INSTABILITY BOUNDARY

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Extension 4236

Maynard Sandford  
Configuration Aeroelasticity Branch  
Extension 2661

RTOP 505-33-43

Research Objective - The objective of this research is to investigate an unusual transonic instability encountered in a wind tunnel test of the DAST ARW-2 wing and demonstrate the use of active controls to suppress the instability.

Approach - NASA's Drones for Aerodynamic and Structural Testing (DAST) program involves flight testing of several Aeroelastic Research Wings (ARW) on a drone aircraft. The ARW-2 right wing is instrumented for unsteady pressure measurements and is flexible enough to require a flutter suppression system. The right wing was tested in September, 1983 in the Langley Transonic Dynamics Tunnel (TDT) in order to obtain unsteady pressure measurements on an aeroelastic wing.

During the test an unusual instability boundary was encountered at dynamic pressures well below those predicted for flutter using linear aerodynamic theory. The instability was observed at 0.9 Mach number for all dynamic pressures within the tunnel operating limits. It was sensitive to wing angle-of-attack and was characterized by first-wing bending mode motion. The instability occurs outside the DAST vehicle's flight envelope but it is of great interest due to the extremely low values of dynamic pressure for which it occurs.

Status/Plans - A second tunnel test is planned in the TDT to further investigate the instability. The flutter suppression system included in the ARW-2 program will be modified to attempt to suppress the instability. Further pressure data will be taken to help identify the cause of the transonic instability. In addition, the increased density capability of the TDT will be used to investigate the behavior of the instability at higher dynamic pressures.

Measured unsteady pressure and dynamic wing deflection data from the first test are being analyzed and will be used to aid in the design of the flutter suppression system to suppress the transonic instability in the upcoming test. This test will provide further information of the mechanisms involved in such instabilities and will allow determination of the utility of active controls to alleviate this class of problem.

Figure 65.

## AEROSERVOELASTICITY

### FY 85 PLANS

- 0 ANALYSIS METHODS
  - APPLICATION OF FLEXSTAB
  - FURTHER DEVELOPMENT OF OPTIMAL SENSITIVITY ANALYSIS
  
- 0 CONTROL LAW SYNTHESIS METHODS
  - ADD STEADY-STATE RESPONSES AND SINGULAR VALUE CONSTRAINTS TO CONTROL LAW OPTIMIZATION
  - DEVELOP METHOD FOR SIMULTANEOUS CONTROL/STRUCTURE DESIGN (STATIC)
  
- 0 APPLICATIONS AND VALIDATIONS
  - COMPARE ANALYSIS WITH WIND-TUNNEL DATA FOR ARW-2
  - APPLY CONTROL LAW OPTIMIZATION METHODS TO INTEGRATED ACTIVE CONTROLS FOR A SUPERSONIC-CRUISE FIGHTER CONFIGURATION
  
- 0 DAST

Figure 66.

## PARAMETER SENSITIVITY METHODS IN CONTROL LAW SYNTHESIS

Michael G. Gilbert  
Kentron International Inc.

RTOP 505-33-43

Research Objective - Control law synthesis methods for multi-input, multi-output systems described in the time domain by linear, state-space equations suffer from the lack of established relationships between controlled system performance and control law design parameters. Because such relationships are lacking, control law design in the time domain usually requires a trial and error approach to satisfying controlled system performance specifications. The objective of this research is to relate time domain control law design parameters to common system performance measures using parameter sensitivity analysis, and then using this information to systematically design control laws in the time domain to satisfy design criteria.

Approach - The present approach to developing relationships between time domain control law design parameters and controlled system performance makes use of parameter sensitivity methods. A generic state-space model formulation of a controlled system is used to obtain the sensitivity of controlled system performance criteria, such as system time response, frequency response, eigenstructure, etc., to controller parameters. Since these performance criteria, and hence sensitivities, may be functions of time or frequency, cumulative sensitivity measures are needed. These results are then useful for redesigning existing control laws to meet specific performance objectives. Using the concept of the sensitivity of an optimum solution of an optimization problem to parameters fixed during the optimization, results are obtained for the important Linear Quadratic Gaussian (LQG) optimal control problem. Here the sensitivity (or the derivative) of the closed-loop state-space model coefficient matrices are obtained for parameters which are part of the specification of the LQG problem, that is, the elements of the cost function weighting matrices and noise covariance matrices. These derivatives are then used directly in the previously obtained performance sensitivity results to relate controlled performance with LQG problem specification parameters.

Status/Plans - The parameter sensitivity methods described above have been developed, but not formally documented or exercised on large-scale, realistic design problems. Thus, near term work will be concentrated in these areas. Follow-on development includes extending the performance sensitivity methods to parameters which are formally part of the dynamic system which is to be controlled, so that simultaneous design iteration of the system and the controller can be undertaken. This extension will serve as a means of integrated system/active control design.

Figure 67

## DAST PROGRAM - FY-85 PLAN

H. N. Murrow  
Aeroservoelasticity Branch  
Extension 3451

RTOP 505-33-43

Research Objective - The second research wing (ARW-2) in the Drones for Aerodynamic and Structural Testing Program was designed such that its structural integrity is dependent on successful operation of an active feedback control system. The active control system provides functions of gust and maneuver load alleviation flutter suppression, and vehicle stability augmentation. The overall objective of the research is to acquire flight test data and correlate results with prediction to validate analysis and synthesis methods for design of multiple-purpose, integrated active control systems. The instrumentation system will also allow measurement of steady and unsteady surface pressure distributions.

Approach - Since the flights tests are of high risk due to the research involving structural integrity, the test vehicle is remotely-piloted. Experimenters transmit commands to excite the wing at various flight conditions and from near real-time assessments advise the pilot on proceeding to the next test point. The wing and active control systems were designed under contract; the wing fabrication, instrumentation, active control system installation, and overall checkout is in-house; fabrication of the on-board active control electronics was under contract; and flight testing and data analysis and correlations will be in-house.

Plans - Planned accomplishments for FY-85 are shown in figure 68(b). The final analyses on the influence of flexibility on stability and control will be completed in FY-85. The analytical procedure will be finalized and results will be based on measured structural characteristics of the wing. These results will be inserted into the real-time simulation and piloted simulation studies conducted to plan an efficient series of test sequences to accomplish the flight objectives.

The design is complete of an uninstrumented wing with high strength and stiffness fabricated to the design cruise shape. This wing will be utilized to validate vehicle flight systems prior to flight of the research wing. This "rigid" wing is to be fabricated in LaRC shops during FY-85.

In the first entry in the TDT, the instrumentation on the right semispan of the flight ARW-2 wing with supercritical airfoil was exercised to measure oscillating wing surface pressures due to control surface oscillation. At the higher Mach Numbers ( $M = 0.9$ ) an instability was observed similar to so-called shock-induced instabilities observed previously in several other programs. A second TDT entry is planned to further explore this phenomenon. It is planned that a control law be developed to suppress the instability, with its demonstration included in the tunnel test series.

Figure 68(a).

DAST PROGRAM - FY-85 PLAN

- 0 COMPLETE FLEXSTAB ANALYSES
  - ESTABLISH FLEXIBILITY EFFECTS ON STABILITY CHARACTERISTICS
  - MEASURED STRUCTURAL CHARACTERISTICS
  
- 0 CONDUCT REAL-TIME PILOTED SIMULATION STUDIES
  - ESTABLISH TEST SEQUENCES
  
- 0 FABRICATE RIGID WING
  
- 0 IMPLEMENT ANOTHER ENTRY IN TDT WITH RIGHT SEMISPAN
  - STUDY "SHOCK-INDUCED" INSTABILITY
  - EVALUATE A CONTROL LAW FOR SUPPRESSION
  - ACQUIRE ADDITIONAL DATA TO EXPLORE PHENOMENON

## DAST FLIGHT SIMULATION ACTIVITIES

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Aeroservoelasticity Branch  
Extension 3527

RTOP 505-33-43

Research Objective - The objective of these activities is to develop and validate a dynamic analysis capability for the DAST flight vehicle. A sampling of the objectives of the simulation activity with regard to the primary AFCS (Aircraft Flight Control System) would be as follows: (1) to evaluate dynamics associated with launch of the DAST aircraft at various altitudes ranging from 15 to 25 thousand feet at Mach numbers of about 0.42, (2) evaluate the time required to proceed to a test point after launch considering the altitude lost, (3) evaluate the capability to modify speed quickly using the throttle and or speed brakes at different flight conditions, (4) evaluate the capability to do push over/pull up maneuvers and/or wind up turns to achieve a slowly varying normal acceleration from near 0-g to 2.5-g at the MLA (Maneuver Load Alleviation) test point ( $M = .42$  at 10K feet altitude), (5) evaluate the capability of the control law shaping of the pilots stick input to keep the aircraft from attaining structurally unsafe normal load factors at all points within the flight envelope, (6) evaluate the flying qualities of the aircraft at all points within the flight envelope, and (7) to evaluate the capability of accomplishing landing approaches with engine off.

Approach - The initial approach was to incorporate the current DAST ARW-2 stability and control aerodynamic data base into a six-degree-of-freedom (figure 69(b)) real time simulation program operating in conjunction with a general purpose simulation cockpit (GPSF). The simulation has the capability of using standard flight instruments, an electronically generated landing scene (IDS) or the visual landing display system (VLDS).

Status/Plans - The simulation program has been checked out and is operational. In addition the capability to perform horizontal landing under a variety of turbulent conditions has been evaluated. During the course of the simulation activity it is expected that it will be desirable to try variations of the various control laws, i.e., aircraft flight control system, active control systems (relaxed static stability, maneuver/gust load alleviation), and backup control system. Simulation activities to evaluate the other research objectives are proceeding on a regular basis. The current data base includes the effects of wing flexibility as a function of flight dynamic pressure. The wing as fabricated was stiffer in torsion than originally expected so the data base will be updated to account for this change.

Figure 69(a).

## DAST FLIGHT SIMULATION ACTIVITIES

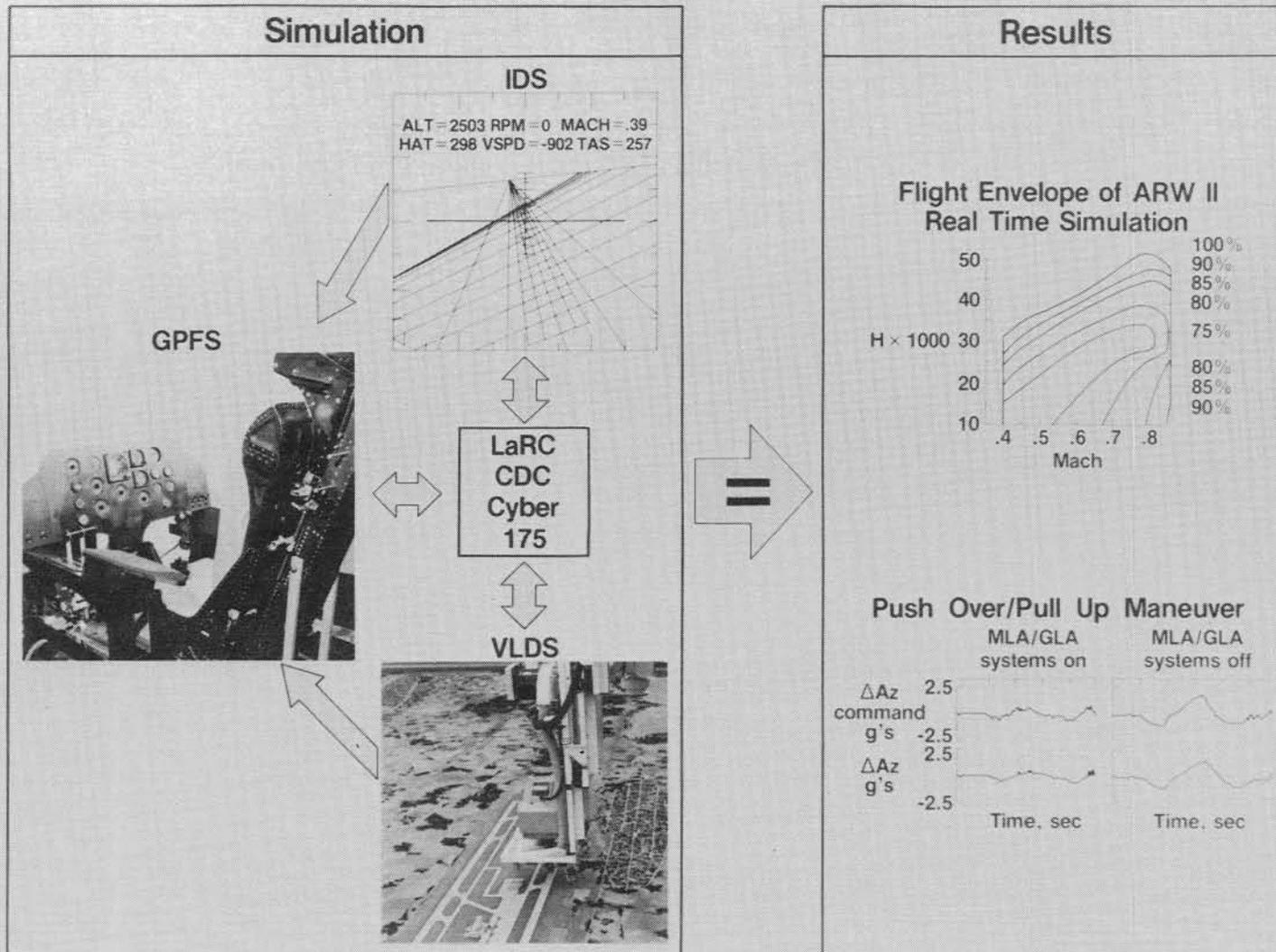


Figure 69(b).



## AEROTHERMAL LOADS

### FY 85 PLANS

- 0 COMPLETE TESTS OF SLOTTED RENE' 41 HOT TANKAGE STRUCTURE, CURVED SA TPS, AND ACC TPS
- 0 COMPLETE AEROTHERMAL LOADS TEST ON SIMULATED TPS ARRAY AND CHARACTERIZATION OF 8' HTT TURBULENT BOUNDARY LAYER
- 0 COMPLETE DEVELOPMENT AND INITIATE VALIDATION OF PROTOTYPE 2D F.E. NAVIER STOKES CODE ON 8' HTT EXPERIMENTAL RESULTS
- 0 DOCUMENT SPHERICAL PROTUBERANCE, SPLIT ELEVON, AND CONE/MASS ADDITION AEROTHERMAL LOADS TEST RESULTS
- 0 EVALUATE PROTOTYPE ALT MACH CONCEPTS IN 7" HTT
- 0 COMPLETE TESTS OF CHINE GAP HEATING MODEL IN 8' HTT

Figure 70.

## QUILTED TILE ARRAY SIMULATING THERMALLY BOWED METALLIC TPS

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Extension 3423

RTOP 506-51-23

Research Objective - Metallic TPS panels on high speed vehicles are subject to thermal distortions when they experience large through-the-thickness temperature gradients. The panel, anchored at the corners, bow up into the flow field altering a smooth vehicle moldline to a quilted surface configuration. Although the bowed height of the panels is expected to be less than the local boundary layer thickness, the complex interaction of the high-speed flow field and the bowed surface will affect the local and global aerothermal loads to the vehicle. An experimental aerothermal study is required to compliment current analytical studies of this problem.

Approach - A quilted array of ceramic tiles were designed and fabricated to fit the 8'HTT panel holder as shown in figure 71(b), which provides two-dimensional laminar or turbulent boundary layer flow. The tiles were designed 10 inches square with protuberance heights of 0.1, .2, and .4 inches. The same tiles were designed to be tested in both aligned and staggered array configurations. The array includes two types of metallic instrumented tiles to be inserted into the ceramic tile array for surface pressure and thin-wall heat-transfer measurements.

Status/Plans - Currently, the model is being installed in the panel holder in the model preparation area. All of the critical model component fitting checks have been successfully completed. Only the packing and wiring of instruments remain in this effort. Aerothermal test are scheduled for FY 85.

Figure 71(a).

QUILTED TILE ARRAY SIMULATING THERMALLY BOWED METALLIC TPS



Figure 71(b).

## CHINE TILE-GAP HEATING MODEL

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RTOP 506-51-23

Research Objective - The application of reusable surface insulation tiles on the Shuttle has introduced local flow disturbances associated with the gaps between tiles. The effects of these disturbances and the resulting penetration flow into the tile gaps has been studied extensively on flat surfaces. Important parameters including gap width and length and flow angularity have been identified, and the effects on localized and total heat loads have been evaluated. Similar studies are needed for curved surfaces where natural pressure gradients occur that would cause greater flow ingestion into the tile gaps and augment the aerothermal loads. For the Shuttle, many of the tile gaps are filled to circumvent this problem, but this practice costs in weight and labor. The actual aerothermal loads associated with tile gaps on the chine with high surface pressure gradients need to be defined to serve as a data base for future thermal protection system designs.

Approach - The Curved Surface Test Apparatus, CSTA, as shown in figure 72(b), has been developed as a test bed for the Langley 8-Foot High Temperature Tunnel. The CSTA is representative of the forward portion of a lifting body and the complex, three-dimensional flow field around this body has been defined experimentally and analytically. An extensive array of simulated tiles has been designed and fabricated to cover a large portion of the CSTA. Three thin-wall, metallic heat-transfer tiles, one for the small radius chine and two for the large radius chine, will be located adjacent to solid tiles instrumented with pressure orifices to determine the aerothermal effects in the tile gaps.

Status/Plans - The Chine Tile-Gap Heating Model fabrication was completed in FY 84 and aerothermal test in the 8'HTT are scheduled for early FY 86. During the interim, the application of additional newly developed instrumentation that will help characterize flow in the gaps is being investigated.

Figure 72(a).

CHINE TILE-GAP HEATING MODEL

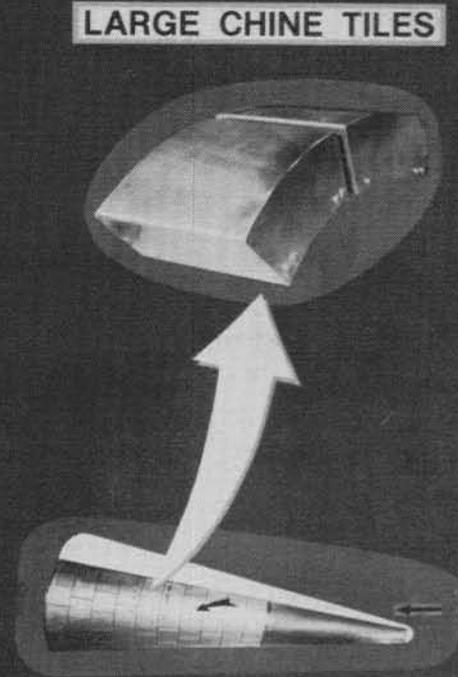
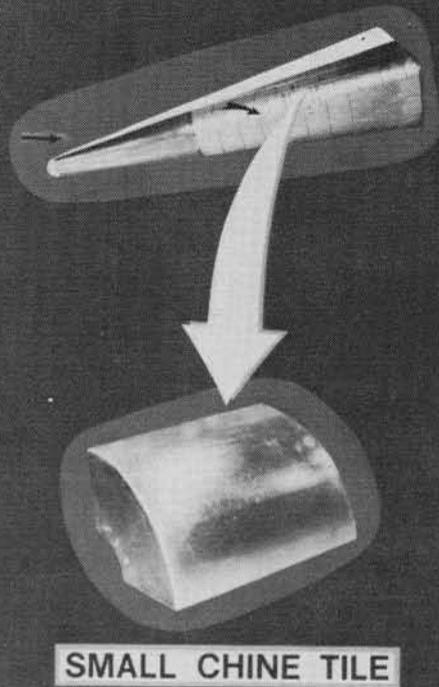


Figure 72(b).



OXYGEN ENRICHMENT AND ALTERNATE MACH NUMBER MODIFICATION  
TO THE 8' HIGH TEMPERATURE TUNNEL

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Extension 3423

Research Objective - The 8' HTT is to be modified to provide a unique national research facility for hypersonic air-breathing propulsion systems. The modified facility, which can accommodate free standing engines, will complement existing lower speed air force facilities by providing true flight simulation for Mach numbers from 4 to 7 over a wide range of altitudes. These expansions in capability are shown in figure 73(b).

Approach/Accomplishments - An oxygen enrichment system will be added which will maintain the correct oxygen concentration for propulsion testing in the methane-air combustion-heated test stream. Supplemental nozzles will be provided for testing at Mach numbers of 4, 4.5, and 5. These nozzles will be coupled with a mixer which will reduce the temperature of the test stream and increase the mass flow to provide the correct flow conditions for the lower mach numbers. In addition, various existing tunnel components will be refurbished and updated to increase facility productivity. One of these modifications will be the replacement of the present water cooled contracting section and the film cooled throat of the Mach 8 nozzle with air transpiration cooled components which should improve the performance and increase the life of the nozzle. Transpiration cooling will also be used for regions of the lower Mach number nozzles requiring cooling.

Status/Plans - The complete schedule for completing the modifications is shown in figure 73(c). Design and verification of the modifications will be completed by April 1985. Construction and checkout of the modifications should be completed and the facility ready for propulsion testing by the fall of 1988.

Figure 73(a).

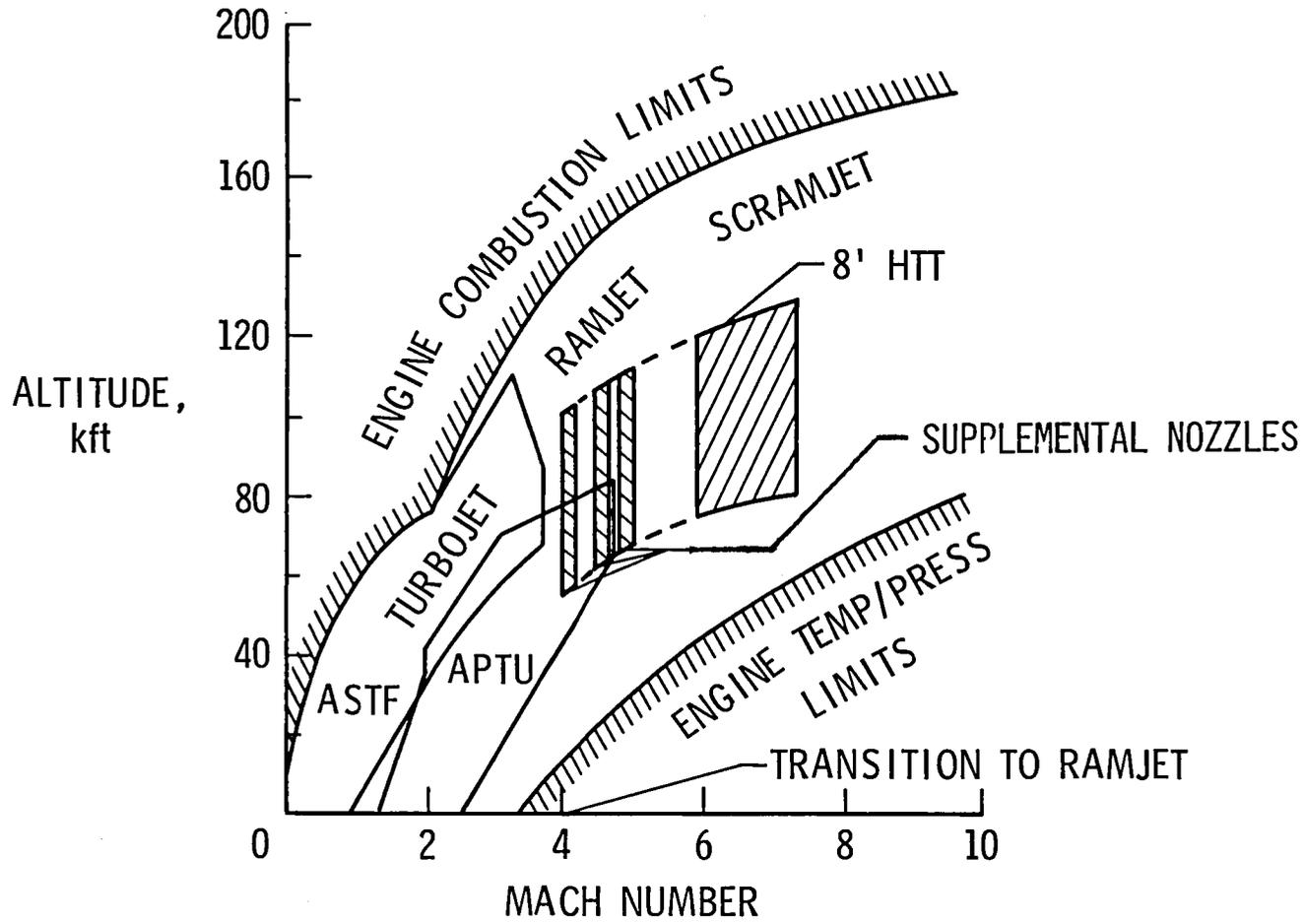
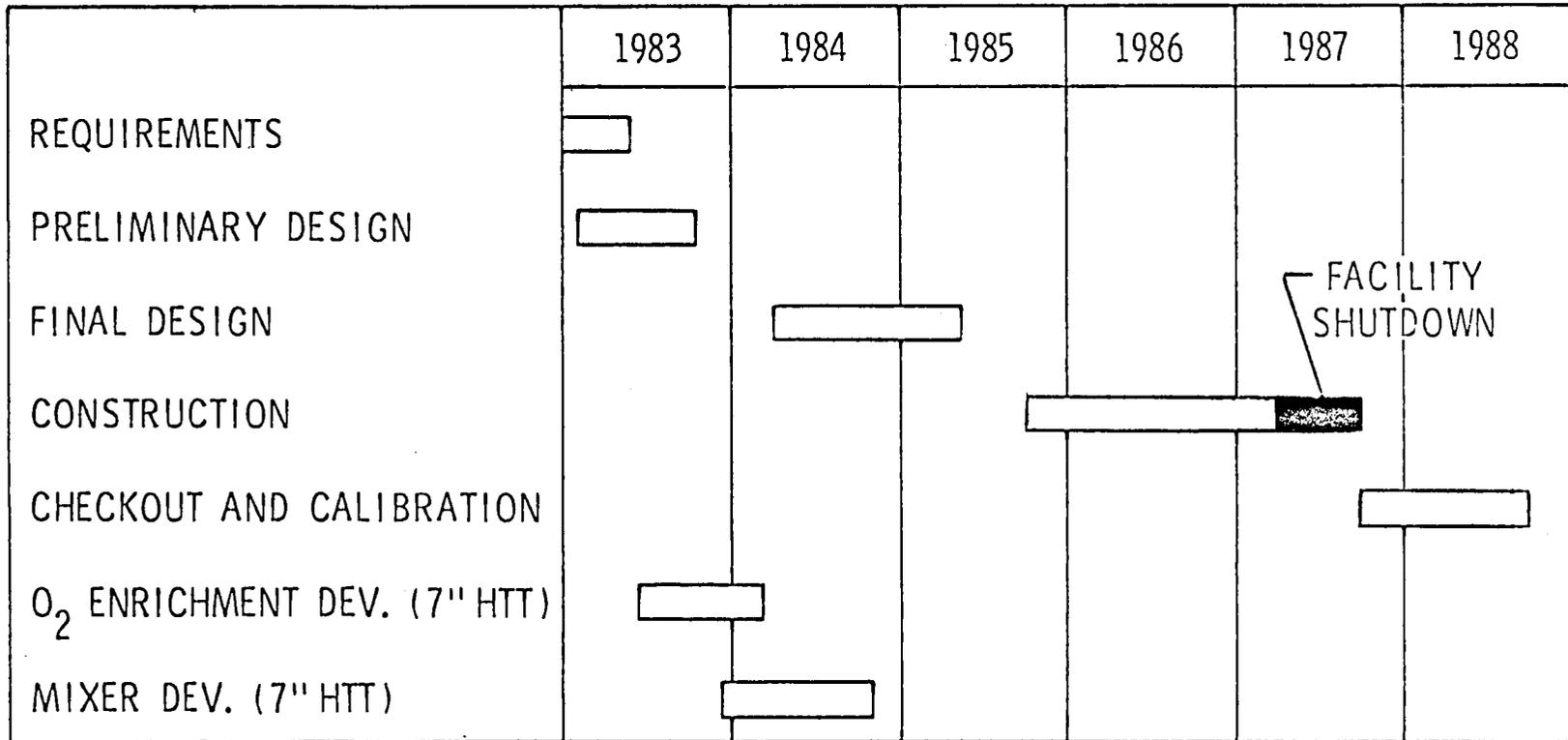


Figure 73(b).

## SCHEDULE FOR 8-FOOT HTT MODIFICATION



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Figure 73(c).

## THERMAL STRUCTURES

### FY 85 PLANS

- 0 CONTINUE STRUCTURAL SYSTEM STUDIES
  - MILITARY AEROSPACE VEHICLES (MAV)
  - ENTRY RESEARCH VEHICLE (ERV)
  - AEROASSISTED ORBITAL TRANSFER VEHICLE (AOTV)
  
- 0 PROPULSION STRUCTURAL CONCEPTS VERIFICATION
  - RECEIVE SCRAMJET STRUT AND INITIATE TEST PROGRAM
  - CONTINUE RAMJET INDIRECT COOLING SYSTEM STUDY AT UTRC
  
- 0 AIRFRAME STRUCTURAL CONCEPT VERIFICATIONS
  - DESIGN AND FAB G/PI PANELS
  - DESIGN AND FAB B/AL PANELS WITH WELDED AL FRAMES
  - DESIGN AND FAB C/C STIFFENED PANELS
  
- 0 TPS CONCEPT VERIFICATION
  - TEST CURVED SUPERALLOY PANELS
  - COMPLETE TEST AND ANALYSIS OF C/C PANEL JOINT
  - INITIATE LEADING EDGE HEAT PIPE STUDY
  
- 0 ANALYTICAL METHODS AND APPLICATIONS
  - ENHANCE CAPABILITY OF SPAR THERMAL ANALYZER
  - DOCUMENT CONCEPTUAL DESIGN ANALYSIS PROGRAMS
  - CONTINUE USE OF PANDA AND PASCO FOR STRUCTURAL OPTIMIZATIONS

Figure 74.

## STRUCTURAL SYSTEM STUDY OF MILITARY AEROSPACE VEHICLES

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Extension 4296

RTOP 506-53-33

Research Objective - To identify the most effective structural arrangement and the most efficient wall construction, which satisfy the vehicle and structural design requirements for military aerospace vehicles. Since airbreathing propulsion is a consideration for such vehicles, the structural arrangement and wall construction of the engines are included.

Approach - Identify alternative structural arrangements and wall constructions for the airframe and engine of military aerospace vehicles in coordination with High Speed Aerodynamics and Space Systems Divisions. Since the optimum structural arrangement and wall construction depend on vehicle configuration, various configurations are studied to obtain a least gross take off weight for a given set of vehicle and structural design criteria. Also, since the propulsion system weight is a function of the airbreathing propulsion cycle used, various engine cycles are compared to achieve the least gross weight. Results enable identification of configuration, engine cycle, structural arrangements of airframe and engine and wall constructions of the airframe and engine that yield least gross weight for a given set of criteria. Least gross weight is determined by analysis of the structure and TPS based on loads and failure modes and an optimum ascent trajectory/vehicle sizing analysis to minimize fuel required considering acceleration and dynamic pressure constraints.

Status/Plans - Alternative configurations, propulsion cycles, structural arrangements and wall construction have been identified. The structural/TPS weight calculation and trajectory/vehicle sizing analytical methods have been generated. The use of these methods for the various alternatives is planned for this year. The results will be used to identify technology needs for the class of aerospace vehicles studied.

Figure 75.

## CURVED METALLIC TPS

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Extension 2425

RTOP 506-53-33

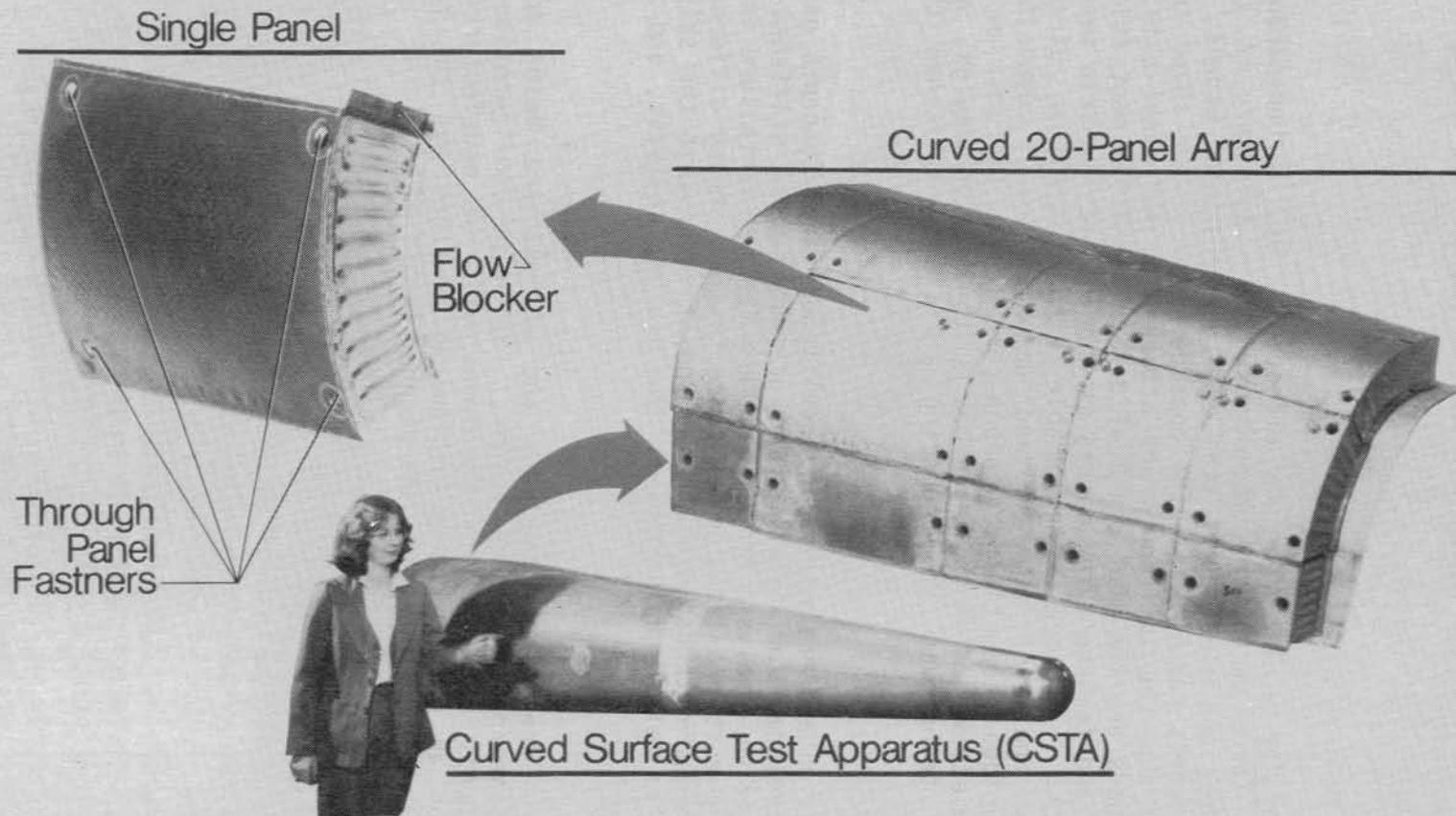
Research Objective - While much of the surface of a typical Space Transportation System (STS) is flat or nearly flat, some areas are necessarily curved. Wind tunnel test data for flat metallic Titanium Multiwall (Ti M/W) and Superalloy Honeycomb (SA/HC) prepackaged Thermal Protection Systems (TPS) have indicated that heating in the gaps between panels can occur and that surface pressure gradients may increase the severity of gap heating. Also, analysis indicates thermal stresses are much higher for curved TPS than for flat TPS. The objective is to fabricate curved SA/HC panels to confirm that fabrication of panels is feasible, to test panels under pressure gradient conditions, and to assess the severity of thermal stresses in curved panels.

Approach - The feasibility of fabricating curved Ti M/W panels has been previously demonstrated. Figure 76(b) shows an array of curved SA/HC panels which has been fabricated for testing on the Curved Surface Test Apparatus (CSTA). The radii of the panels vary from 9 inches to 12 inches. A section of the CSTA will be cut out so that the curved array can be inserted flush with the CSTA skin. Wind tunnel tests in the LaRC 8' HTT will provide temperature and pressure data to allow evaluation of gap heating under pressure gradient conditions. A single curved panel has been fabricated and after being instrumented with strain gages and thermocouples, will be tested under radiant heating to evaluate thermal stresses.

Status/Plans - The array of curved SA/HC panels has been installed in the CSTA, and wind tunnel tests in the 8' HTT are scheduled for March 1985. Since aerothermal tests of flat SA/HC panels indicated that heating can occur between panels when flow is parallel to their edges, flow stoppers have been located at the corners between panels to reduce heating in the gaps.

Figure 76(a).

# CURVED SUPERALLOY HONEYCOMB TPS PANELS



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Figure 76(b).

DESIGN, ANALYSIS, AND TESTS OF STIFFENED  
CARBON-CARBON STRUCTURAL PANELS

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Extension 4201

RTOP 506-53-33

Research Objective - In order to effectively use carbon-carbon composite materials for large structural components, it is necessary to learn how to design, analyze, and manufacture stiffened carbon-carbon panels. The design of stiffened panels requires careful consideration of the shape and number of stiffeners and how the stiffeners are attached to the panel skin. Shear and compression loads are the most critical design loads for most stiffened panel applications, and failure often occurs as a result of the skin separating from the stiffeners. Although considerable effort has been directed toward designing and analyzing metallic and graphite epoxy composite stiffened panels, the unique properties of carbon-carbon material must be considered. Thus, the objective of this research is to develop design, analysis and manufacturing techniques for producing efficient, stiffened carbon-carbon panels.

Approach - Design and analyze various stiffened panel concepts using analysis techniques developed for metallic and graphite epoxy stiffened compression panels. Verify the analysis by building and testing to failure in compression several of the most promising panel concepts that employ different stiffener configurations and methods of attaching the stiffeners to the skins. Use the test results to make improvements in the design, analysis, and construction techniques.

Status/Plans - Preliminary analysis and designs are being developed for several stiffener/panel configurations. Specimen configurations should be defined by April 1985. Test specimens will be manufactured by Vought Corporation on an existing task assignment contract and the test will be conducted at Langley. The tests are planned for FY 86.

Figure 77.

## THERMAL/STRUCTURAL ANALYSIS AND OPTIMIZATION

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Thermal Structures Branch  
Extension 2291

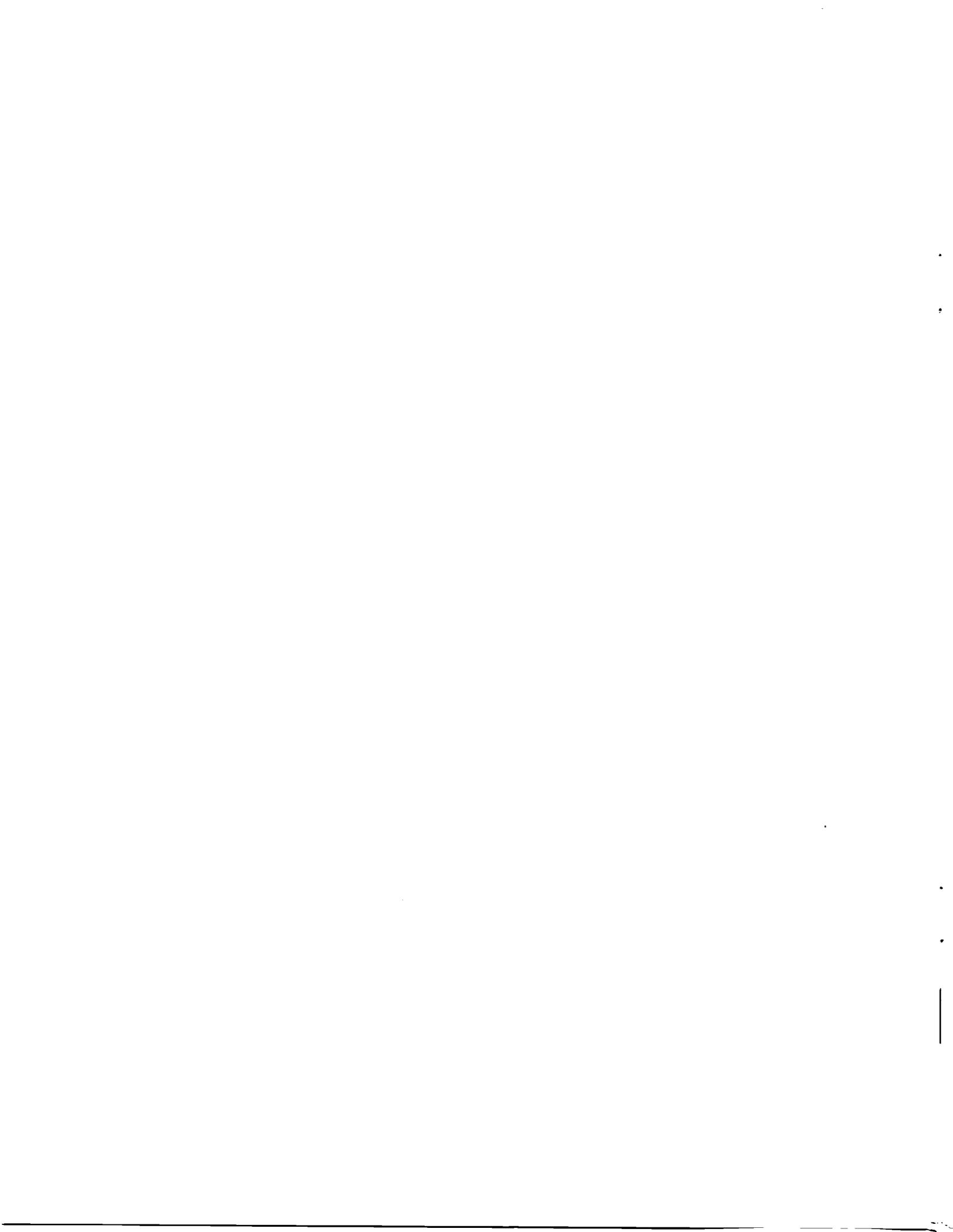
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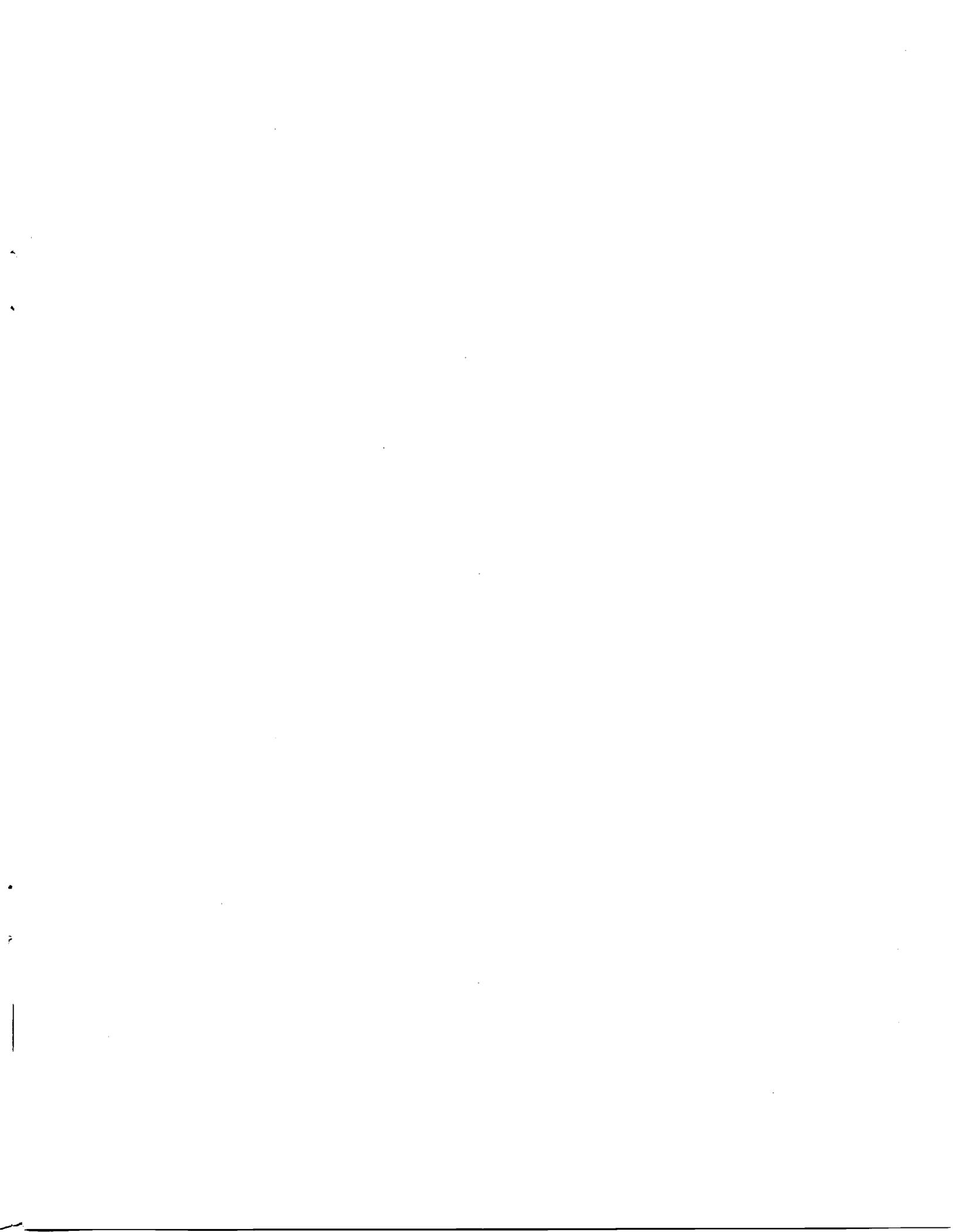
Research Objective - The design process for a hypersonic vehicle requires detailed thermal and structural analyses of a system composed of thermal protection which has high temperature resistance but no capability to carry primary vehicle loads plus primary structure which carries the loads and can absorb heat but is limited to use at temperatures well below surface radiation equilibrium temperatures. Furthermore, the thermal response of the system is transient and nonlinear resulting in considerable analysis effort for each of the numerous flight paths that must be studied. The objective of this research is to (1) develop optimization procedures that will enable the analyst to select the best variation of a given concept for moderate-sized models and (2) improve the efficiency of the analytical tools so that (1) (above) or the single analysis of a large analytical model is affordable in terms of cost and calendar time.

Approach - Initiate a task-oriented contractual effort with Engineering Information Systems, Inc., for a period of approximately two years to do, at least, the following: (1) Develop reasonably general logical procedures, based on EAL runstreams, to automatically resize a system (assuming variables to be continuous, if necessary) to satisfy constraints on results such as stress and temperature, while minimizing a merit function, such as system mass. (2) Incorporate a general purpose mathematical optimization program (such as CONMIN or ADS) as an EAL processor. (3) Modify the various programs for efficient operation on the LaRC VPS-32 computer for increased solution speed and allowable model size. (4) Solve a small number of demonstration problems to check out resulting methods and procedures. (5) Maintain and update procedures and thermal analyzer over period of contract.

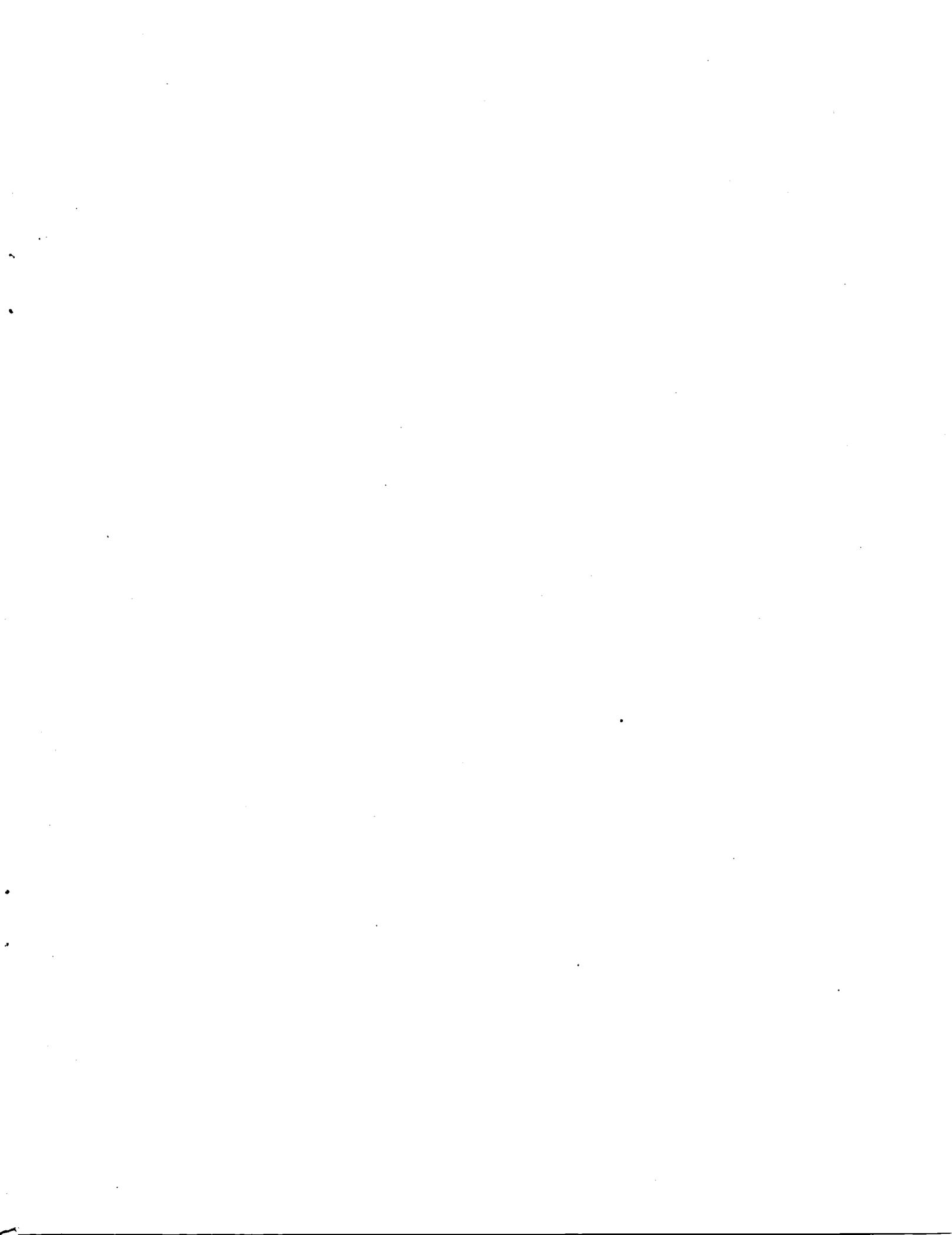
Status/Plans - Initiate request for proposal for the contract early in CY 1985 with contract award by mid-CY 1985.

Figure 78.





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