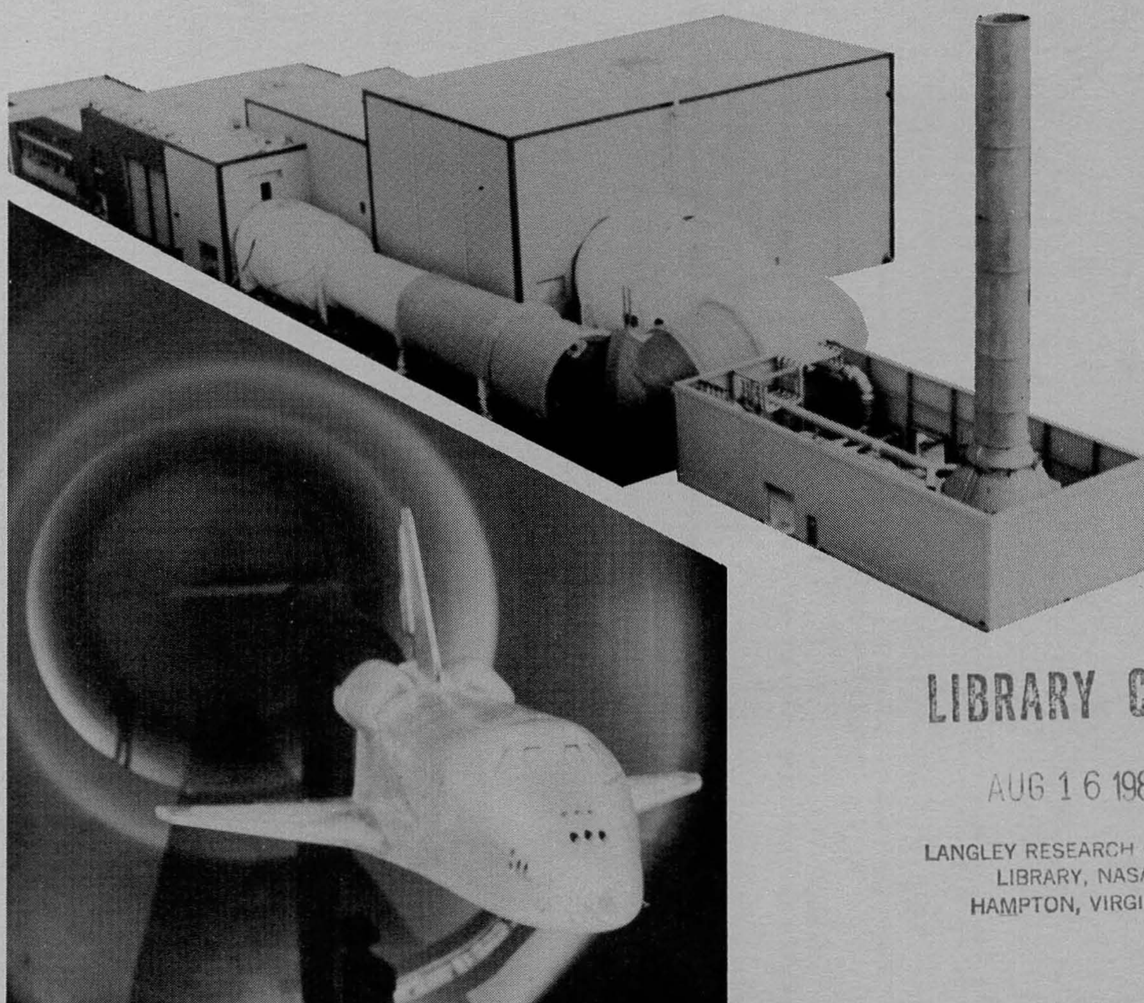
NASA Technical Memorandum 84519

Langley Test Highlights - 1981



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National Aeronautics and
Space Administration

1982

Foreword

The role of the Langley Research Center is to engage in the basic and applied research necessary for the advancement of aeronautics and space flight, to generate new and advanced concepts for the accomplishment of related national goals, and to provide research advice, technological support, and assistance to other NASA installations, other government agencies, and industry. This report highlights some of the significant tests which were performed during calendar year 1981 in Langley test facilities, a number of which are unique in the world. The report illustrates both the broad range of the research and technology activities at the Langley Research Center and the contributions of this work toward maintaining United States leadership in aeronautics and space research. Other highlights of Langley research and technology for 1981 are described in "Research and Technology—the 1981 Annual Report of the Langley Research Center." Further information about both reports is available from the Office of the Chief Scientist, Mail Stop 103, Langley Research Center, Hampton, Virginia 23665 (804-827-3316).

Donald P. Hearth
Director

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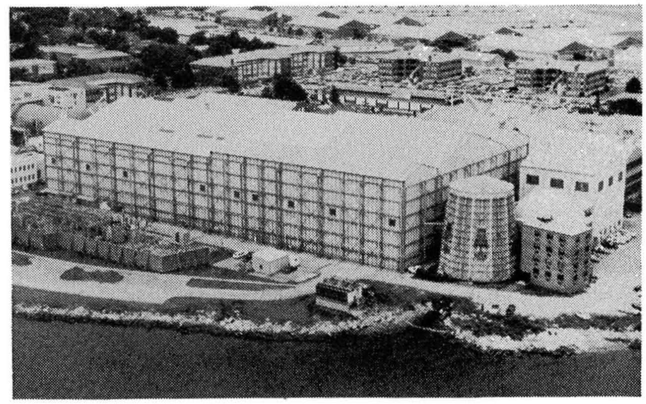
30- by 60-Foot Wind Tunnel



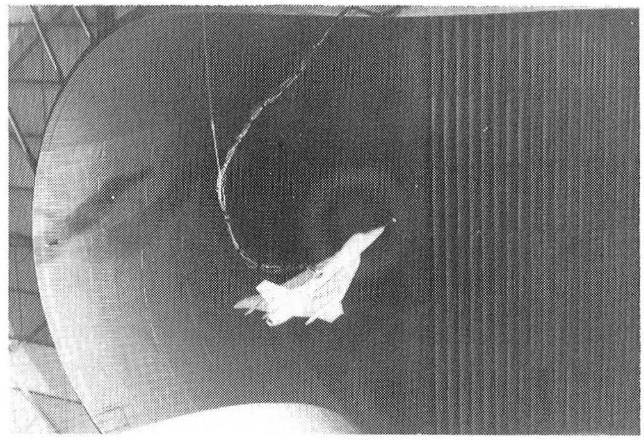
The 30- by 60-Foot Wind Tunnel is a continuous flow, double return, open throat type tunnel. The test section is 30 feet high and 60 feet wide and can accommodate airplanes or models having spans to about 40 feet. The tunnel is powered by two 4-blade, 35.5-foot-diameter fans, each driven by a 4000 hp electric motor. The maximum speed of the tunnel is about 100 mph. When this tunnel was first placed into operation in 1931 its maximum speed was equal to the top speed of many of the airplanes then flying. Since then the maximum speed of airplanes has not only surpassed the top speed of the tunnel manyfold, but transonic and supersonic airplanes operate in realms into which the low-speed data cannot be extrapolated. The design of these airplanes, however, has required wing shapes and airfoil sections that result sometimes in poor low-speed characteristics. This tunnel is well suited to investigate means of alleviating these low-speed problems because full- or large-scale hardware can be used, and the model or airplane is readily accessible.

In addition to the testing capabilities of extensive flow measurement and visualization for large-scale models, the tunnel is equipped with shielded struts for 6-component scale balance testing, and also can be used for free-flight tests of subscale models. These tests are particularly suited to the study of high-angle-of-attack flight dynamics for advanced fighter configurations.

F-16E Free-Flight Test

As a proposed derivative of the current F-16, the F-16E incorporates an advanced arrow wing which affords a number of advantages including enhanced supersonic performance. Since 1978 when the design was in its infancy, a cooperative program with General Dynamics has been under way in the Langley 30- by 60-Foot Wind Tunnel to investigate the critical low-speed, high-angle-of-attack characteristics of the airplane. Using a model of an early design, low-speed static tests were conducted over large angle-of-attack and sideslip ranges. During the tests, several problem areas were identified and configuration modifications to alleviate these problems were developed. These modifications have been incorporated into the final design and include a number of modifications to the wing to improve longitudinal and lateral stability.

Subsequently, tests were conducted on a 0.18-scale model of the final F-16E design. In addition to static and dynamic captive-model testing, free-flight tests of this configuration are also being conducted. In these free-flight tests, the dynamically-scaled model is flown unconstrained in the test section of the tunnel. The output of the free-flight tests is in the form of pilot comments and records of the model motions, as documented on film and measured by sensors mounted in the model. The results show that the flying characteristics of the configuration under actual flight conditions correlate well with those indicated by the captive model tests in the 30- by 60-Foot Tunnel.

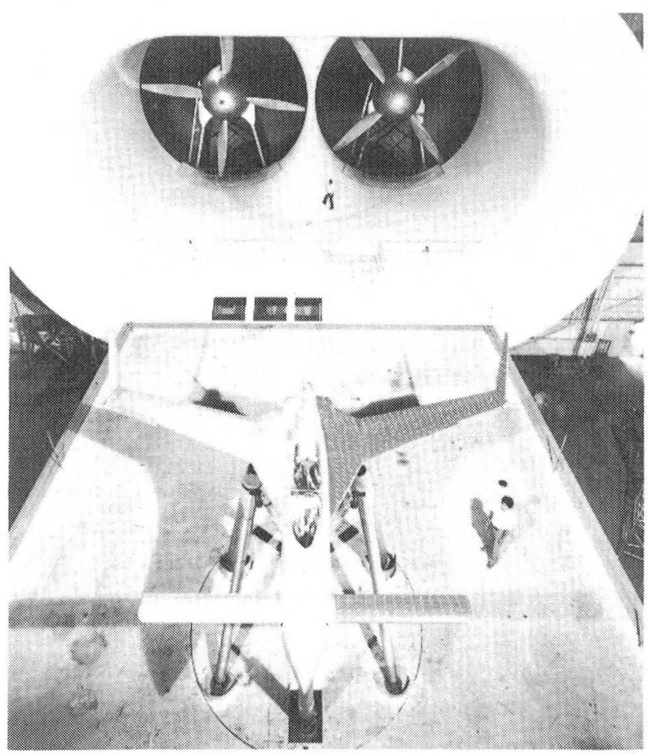


F-16E Tests

Advanced Canard-Configured Aircraft

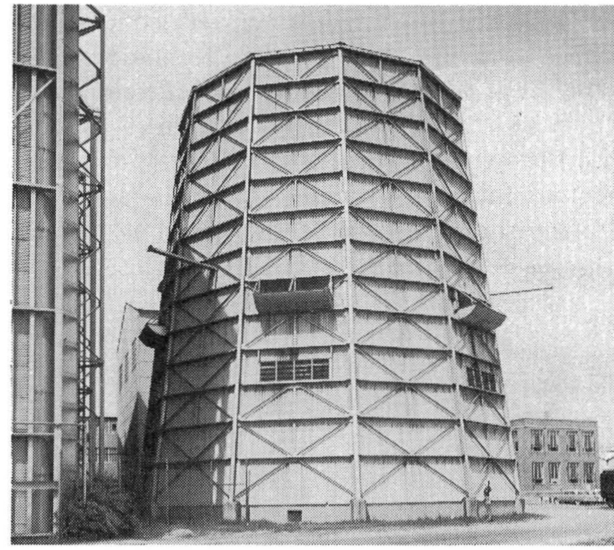
As part of a broad research program to provide a data base on advanced general aviation configurations, a wind-tunnel investigation was conducted in the Langley 30- by 60-ft tunnel to determine the aerodynamic characteristics of an advanced canard-configured general aviation airplane. The investigation included measurements of forces and moments of the complete configuration, isolated canard loads, propeller thrust and torque loads, and pressure distributions on the wing, winglet, and canard. Flow visualization was obtained by using surface tufts to determine regions of flow separation and by using a chemical sublimation technique to determine boundary layer transition locations. Additionally, other tests were conducted to determine simulated rain effects on boundary layer transition. Investigation of configurational effects included variations of canard locations, canard airfoil section, winglet cant angle, winglet size and incidence angle, and use of a leading-edge droop on the outboard section of the wing.

Results of this investigation indicated that this canard configuration was stall resistant due to the angle-of-attack limiting effect of the canard. Natural laminar flow was found to exist to 55 percent chord of the canard. However, loss of laminar flow by means of artificial tripping at 5 percent chord resulted in large nose-down pitching moments. Tests of a canard with a NACA 0012 airfoil indicated that the airfoil section properties can strongly affect the configuration's stall and post-stall characteristics. The addition of an outboard leading-edge droop to the wing prevented the spread of spanwise flow and delayed tip stall to a higher angle of attack.



Advanced Canard-Configured Aircraft

20-Foot Vertical Spin Tunnel

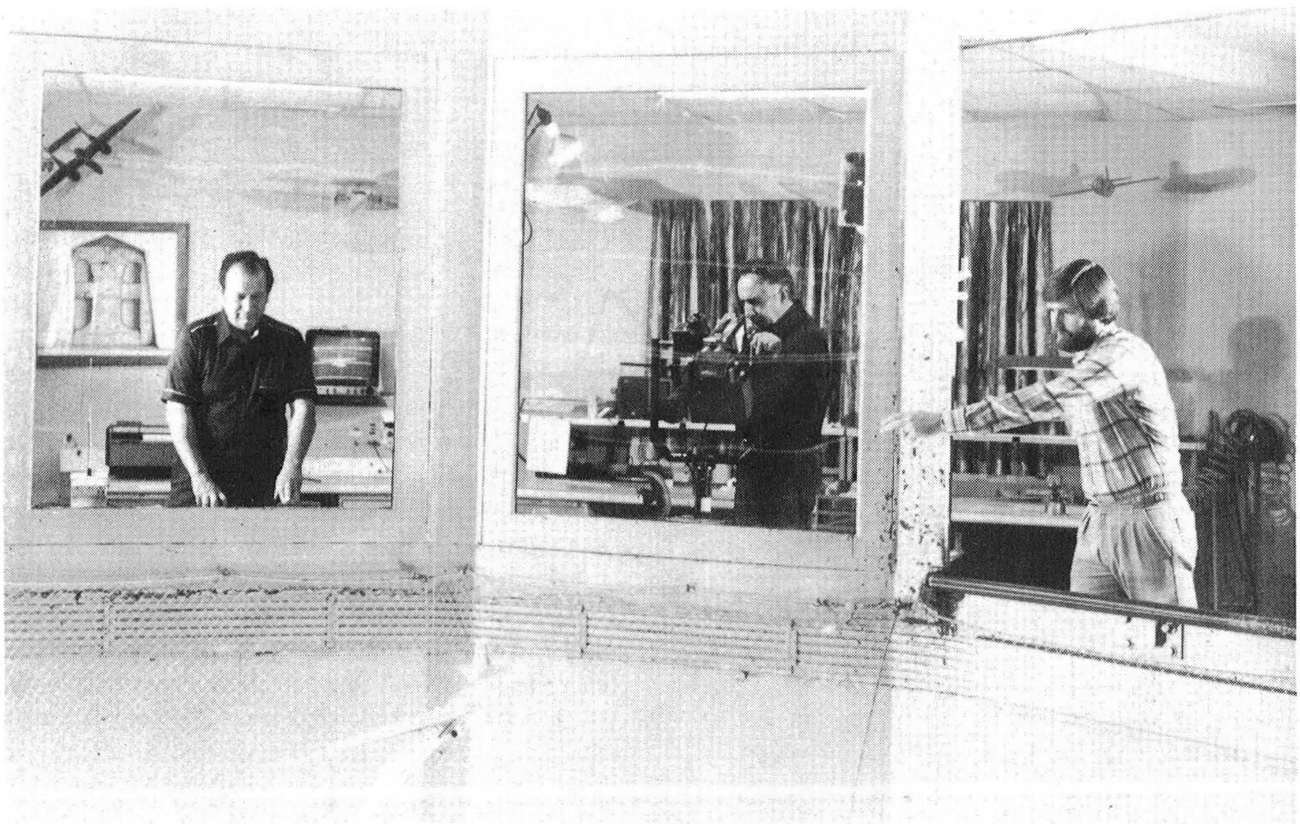


The Langley 20-Foot Vertical Spin Tunnel is the only operational spin tunnel in the United States and one of only two in the free world. The tunnel is used to investigate spin characteristics of dynamically-scaled aircraft models. It is a free-spin tunnel with a closed throat and an annular return passage. The vertical test section has 12 sides and is 20 feet across by 25 feet high. The test medium is air. Tunnel speed is variable from 0 to 90 ft/sec with accelerations to 15 ft/sec^2. This facility is powered by a 1300 hp main drive. Spin recovery characteristics are studied by remotely actuating the aerodynamic controls of models to predetermined positions. Force and moment testing is performed using a gooseneck rotary arm model support which permits angles of attack from 0° to ±90° and sideslip from 0° to ± 20°. Motion picture and video records are used to record the spinning and recovery characteristics in the spin tunnel tests. Force and moment data from the rotary balance tests are recorded in coefficient form and stored on magnetic tapes.

NASA AD-1 Oblique-Wing Aircraft Spin Tests

As part of the research program of the AD-1 oblique-wing research airplane, a spin tunnel test program was conducted on a 1/13-scale model of the AD-1. The tests were conducted to identify the spin modes, spin angle of attack, spin rate, the number of turns for recovery, and the effect of the oblique wing on the spin and recovery. The results indicated that when the wing was in the 0° wing skew position, two spin modes were possible. One mode was very steep and recoveries by full rudder reversal were within one turn or less. The second spin mode was flat with an angle of attack of about 78° and a spin rate of about 145 deg/sec. Recoveries from the flat spin were very slow (3 to 5 turns).

For the wing skewed positions, the results indicated that the spin could be obtained only in the direction of the forward-skewed wing (right spins since only the right wing rotated forward). The spins obtained at the 25° wing skew position were oscillatory. As the wing skew angle increased, the spins became less oscillatory and at 60° wing skew the spin was smooth with an angle of attack of about 65°. The recoveries were either very slow or virtually nonexistent unless the wing was moved to zero wing skew at the time recovery controls were applied. The optimum recovery technique for the AD-1 from the flatspin mode is deflection of the rudder to full against the spin, the ailerons to full with the spin, the elevators to neutral, and movement of the wing to zero wing skew angle. The airplane should recover about the time the skew angle gets to zero. Since about 30 seconds is required to move the wing to zero from 60°, ten or more turns may be required to recover from the 60° wing skew configuration. In general, the recovery characteristics of the AD-1 oblique-wing airplane appear to be very poor.



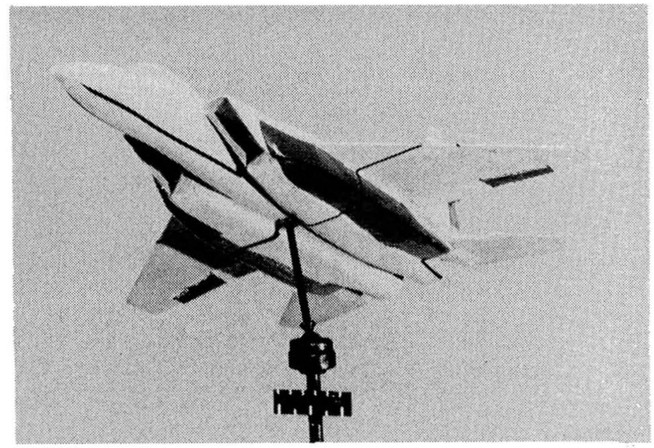
Oblique-Wing Aircraft Spin Tests

Effects of Conformal Fuel Tanks on F-15 Spin Recovery

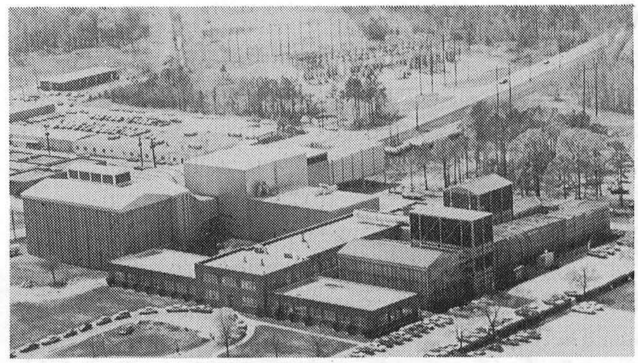
At the request of the United States Air Force, a spin-tunnel test program has been conducted to determine the effects on the spin and recovery characteristics of installing conformal fuel tanks (the darkened structure along the fuselage under both wings) on the F-15 design. The tests were conducted on a 1/30-scale model for free-spinning tests, and on a 1/12-scale model for the rotary-balance tests. The free-spinning test data identify spin modes, spin angle of attack, spin rate, and number of turns for recovery. The rotary-balance test data provide complete force and moment information over an angle-of-attack range from 8° to 90° and a range of spin rates, and provide information on predicted spin modes.

Both the free-spinning tests and the rotary-balance tests indicated that the addition of conformal fuel tanks (CFT's) adversely affected the spin and recovery characteristics of the F-15 airplane.

Without the tanks the free-spinning model predicted a flat spin mode with an angle of attack of about 81° and a spin rate of about 150°/sec, and required about four turns for recovery. When the tanks were added, the spin was a little flatter and faster and about six turns were required for recovery. The additional two turns required for recovery of the CFT configuration are critical since six turns is more than the maximum allowed for satisfactory recovery characteristics. The spin-tunnel model results are expected to be highly representative of the full-scale airplane configuration since Reynolds number effects on the spin-tunnel results were relatively small.



F-15 Conformal Fuel Tank Spin Tests



7- by 10-Foot High-Speed Tunnel

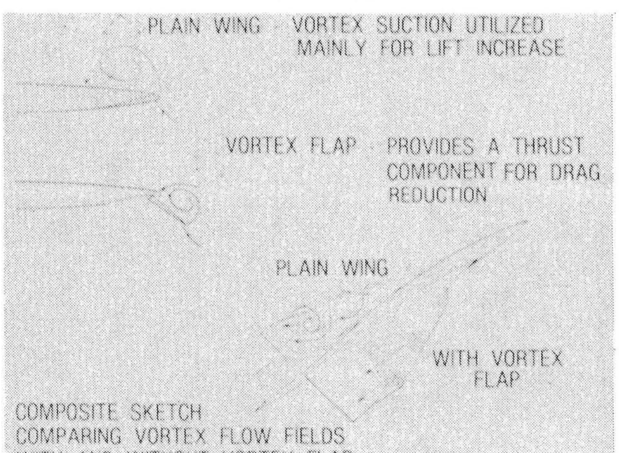
The Langley 7- by 10-Foot High-Speed Tunnel is a closed circuit, single return, continuous flow, atmospheric tunnel, with a test section 6.6 feet in height, 9.6 feet in width, and 10 feet in length. A 14,000-hp electric motor drives a series of fan blades to provide subsonic operating speeds from Mach 0.2 up to 0.9, and to produce a maximum Reynolds number of 4×10^6 per foot. In addition to static testing of complete and semispan models, the facility is equipped for both steady-state roll and oscillatory stability testing.

Currently the facility is playing an important role in a wide range of basic and applied aerodynamic research including advanced vortex lift concepts, fuel conservative aircraft design technology, highly maneuverable aircraft concepts, and the development of improved aerodynamic theories such as the difficult separated flow and jet interaction effects needed for computer-aided design and analysis.

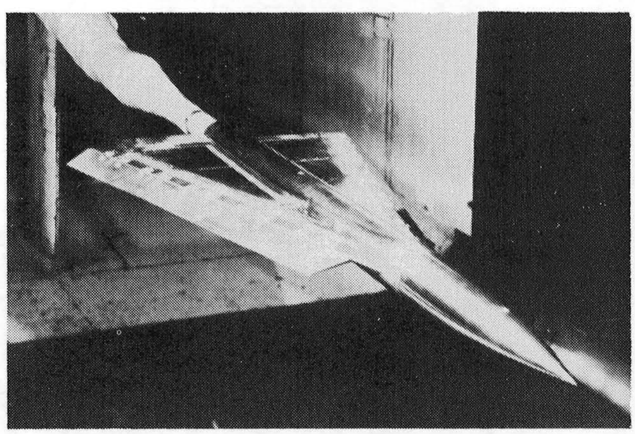
Vortex Flap for Delta-Winged Supercruise

Efforts are under way at Langley to develop the technology necessary for coupling an efficient supersonic cruise capability with ample transonic maneuver capability. The supersonic cruise requirement primarily drives the configuration toward a highly swept planform with a low slenderness ratio. Under transonic maneuver conditions, such configurations are conducive to stable separation-induced vortex lift increments at moderate to high angles of attack, but not without

THE CONCEPT



74° DELTA WING-BODY MODEL



Vortex Flap

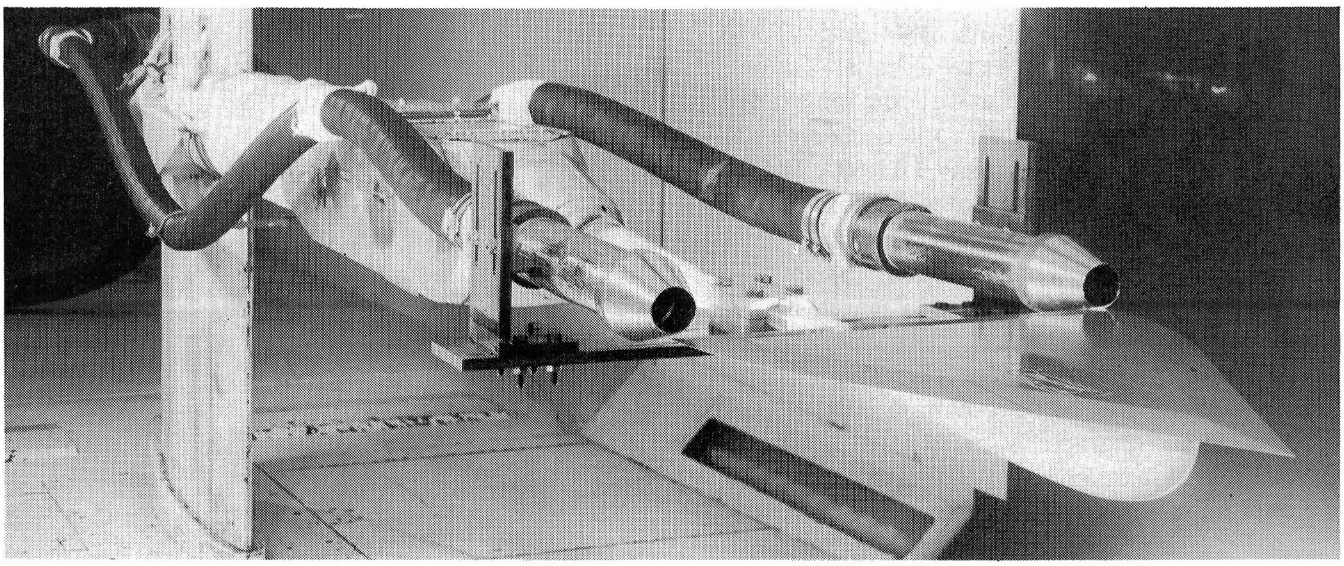
a substantial drag penalty due to the loss of leading-edge suction. The large drag penalty severely degrades the maneuver performance and hence must be reduced.

One approach being investigated for solving the slender-wing drag problem is the "vortex-flap" concept. The basis of the concept is to underdeflect the leading edge relative to the local upwash and allow a small controlled vortex to form along the forward-facing surface. If the deflected leading edge or flap is properly designed, the vortex will induce suction pressures which act primarily on the flap, providing an effective leading-edge thrust recovery and promoting flow reattachment at, or immediately aft of, the hinge line. The concept has been tested in the 7- by 10-Foot High-Speed Tunnel and has demonstrated effective thrust recoveries on the order of 75 percent when used in conjunction with trailing-edge flap. In addition to force measurements, detailed surface pressure distributions on the wing and flap were measured and the agreement between the measured and calculated values was quite good.

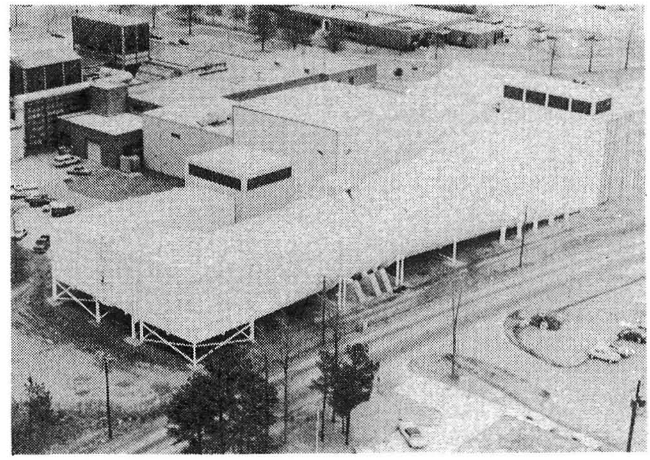
Pumped Vortex Concept

Increasing the vortex lift at moderate to high angle of attack on highly swept wings has been investigated for many years. Takeoff performance and maneuverability of advanced high-speed aircraft may require the favorable lift effect induced by a leading-edge vortex system in order to achieve performance goals. A novel concept has been proposed to utilize the propulsion system as a suction device to augment the leading-edge vortex. As a result, an investigation was conducted in the Langley 7- by 10-Foot High-Speed Tunnel to determine the aerodynamic performance of suction applied near the wingtips above the trailing edge of a 60° Gothic wing. Movable suction inlets were symmetrically mounted in the proximity of the trailing edge and the amount of suction varied to maximize wing lift.

Tests were conducted at Mach 0.15, 0.30, and 0.45, and the angle of attack was varied from −4° to 50°. The suction augmentation, which strengthens the vortex system and delays the vortex breakdown, results in an increase in lift coefficient over the entire range of angle of attack when compared with the same wing without the suction apparatus. The maximum lift coefficient was shown to be as much as 20 percent greater using the suction apparatus.



Pumped Vortex Concept

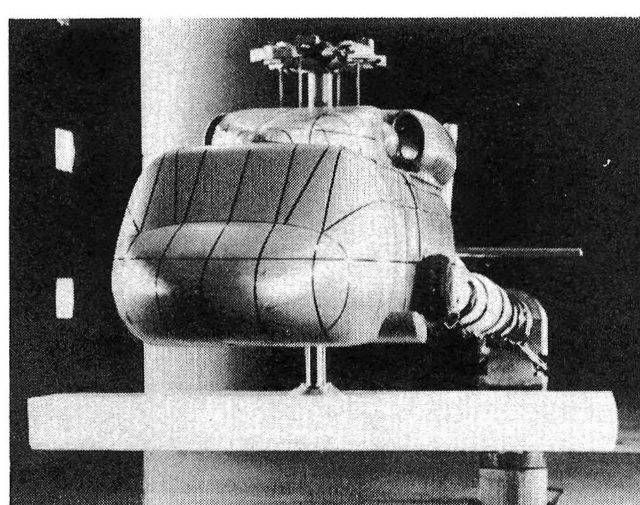


4- by 7-Meter Low-Speed Tunnel

The Langley 4- by 7-Meter Tunnel (formerly V/STOL Tunnel, or Vertical/Short Take Off and Landing Tunnel) is used for testing powered low-speed helicopters and various commercial and military aircraft. It is powered by dual drive motors which can provide precise tunnel speed control from 0 to 200 knots with the Reynolds number per meter range from 0 to 0.64×10^7. The test section is 4.4 m high, 6.6 m wide, approximately 15.2 m long, and can be operated as a closed tunnel with slotted walls, or as one or more open configurations by removing the side walls and ceiling for allowing extra testing capabilities, such as flow visualization and acoustic tests. Furthermore, a moving-belt ground board with boundary-layer suction and variable-speed capabilities for operation at test-section flow velocities can be installed for ground-effect tests.

Helicopter Stand-Off Target Acquisition System

A quarter-scale aerodynamic model of a Stand-Off Target Acquisition System (SOTAS) was tested in the Langley 4- by 7-Meter Tunnel. SOTAS is designed to detect and report long-range targets under adverse visibility conditions and is a derivative of the Army/Sikorsky UH-60A Blackhawk helicopter and the associated Motorola electronics package. The long, rectangular-shaped SOTAS antenna can rotate or be set at any specific angle, alternate back and forth in a search mode, or can be rotated at slow rates. The purpose of the investigation was to document performance and stability characteristics and to compare the results with theoretical characteristics. The tunnel tests were conducted in an unpowered mode; that is, the helicopter main and tail rotors were not used on the model. The forces and moments were measured



SOTAS Tests

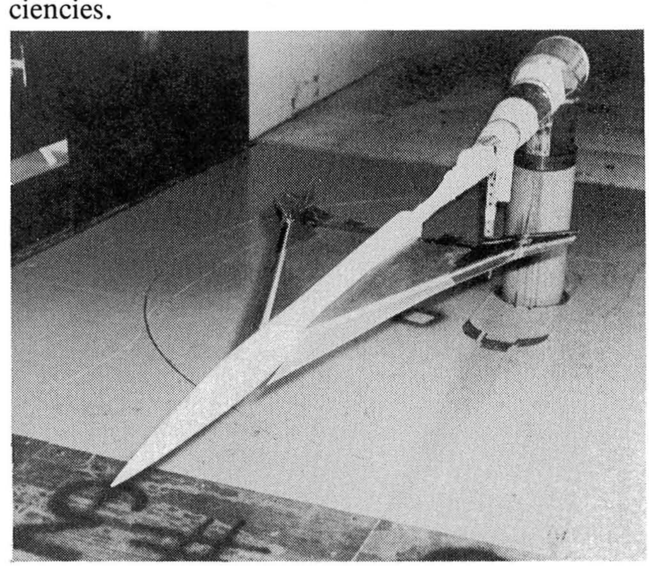
separately for the fuselage, the tail, the antenna, and the fuselage plus antenna.

The measured forces and moments agreed well with the predictions. The test showed that drag was highly dependent on the shape and orientation of the antenna. As expected, when the antenna was positioned broadside to the airstream, the aerodynamic drag of the square-edged antenna was very high. However, rounding the corner of the box-shaped antenna reduced the drag by 55-60 percent.

Supersonic Cruise Research (SCR)

Tests have been conducted in the 4- by 7-Meter Tunnel to explore means for improving the subsonic aerodynamic characteristics of future generation supersonic cruise aircraft. These aircraft concepts are intended to result in efficient flight over long ranges at supersonic speeds. Initially such concepts were considered as potential commercial aircraft; however, the current emphasis is shifting towards possible military use. Supersonic concepts of this type have been considered for over 25 years, but their poor low-speed aerodynamic characteristics have prevented their application. Several years ago, NASA and industry researchers, under Langley Research Center direction, mounted a cooperative effort to minimize or eliminate these low-speed aerodynamic deficiencies. Significant progress has been made and reported in NASA literature.

The present model utilized a wing with a inboard sweep angle of 74°, a midspan sweep angle of 70° and an outboard wing panel sweep of 60°. The test was made at a Mach number of 0.2 over an angle-of-attack range from −8° to 16° for a range of sideslip angles of −10° to 10°. The tests were specifically designed to explore leading-edge deflection concepts and to determine detailed pressure distributions over the wing surface to provide a comprehensive understanding of the fluid mechanics. Analysis of the data revealed that the 74/70 sweep intersection promoted leading edge flow separation, resulting in less efficient subsonic aerodynamic performance. Based on these results, a revised constant inboard 73° wing sweep was designed and will be tested to assure that this revised leading edge alleviates the aerodynamic deficiencies.



Supersonic Cruise Tests

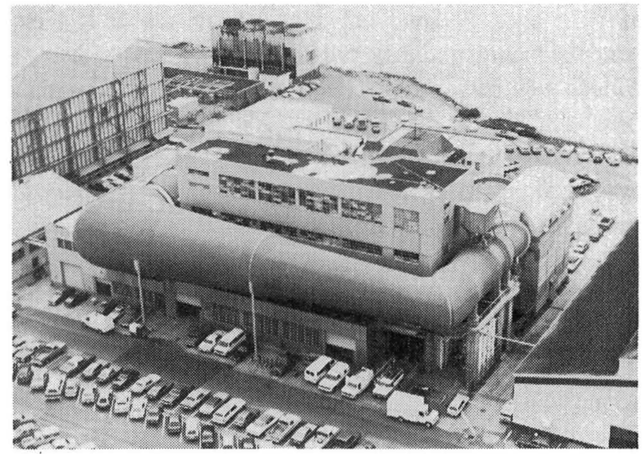
Energy Efficient Transport With a "T"-Tail

High-speed wind-tunnel tests on an energy efficient transport (EET) configuration with a T-tail geometry have shown a decrease in cruise trim drag and a reduction in pitchup when compared to a low tail configuration. To investigate the performance of the EET T-tail configuration in the low-speed flight regime, wind-tunnel tests were conducted in the NASA Langley 4- by 7-Meter Tunnel with a

12-ft span model. The particular configuration tested had an aspect-ratio-10, supercritical wing with a quarter chord sweep of 27°. The model was equipped with a full-span leading-edge slat and two segment trailing-edge flaps. The leading-edge slat and trailing-edge flap deflections were representative of both takeoff and landing conditions. The T-tail was designed to maintain the same horizontal and vertical tail volume as the original low tail configuration.

Force tests were conducted at a Mach number of 0.2 which corresponds to a Reynolds number of 1.63×10^6 based on the wing mean geometric chord. During the tests, mini-tuft flow-visualization techniques were used to observe the flow over the wing. The test results indicated that the T-tail geometry results in a decrease in trim drag as well as a reduction in pitchup compared to a low tail configuration in both takeoff and landing configurations. Consequently, the T-tail appears to be advantageous in both the high- and low-speed regimes.

Energy Efficient Transport Tests



8-Foot Transonic Pressure Tunnel

The Langley 8-Foot Transonic Pressure Tunnel is a closed circuit, single return, variable density, continuous flow type wind tunnel. The test section is 7.1 feet square and is slotted for about 5 percent porosity to minimize tunnel wall interference effects. The stagnation pressure can be varied from below 0.25 atmosphere at any Mach number to 2.0 atmospheres at 0.2 Mach number. At the higher Mach numbers, up to 1.3, the pressure is limited by the available power, 25,000 horsepower. The stagnation temperature is controlled by water-cooled fins upstream of the settling chamber. Tunnel air can be dried by a dryer using silica gel desiccant to prevent fogging due to expansion in the high-speed nozzle.

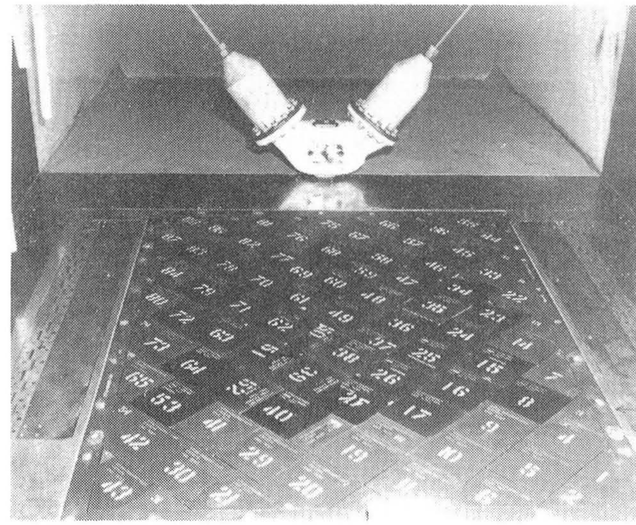
Because the tunnel has a very high contraction ratio, the turbulence level is quite low in the test flow even without screens. The tunnel is currently being re-configured for the Laminar Flow Control experiment. This will involve installation of screens and honeycomb in the settling chamber and a full liner in the test section. The slots will be closed and the schlieren capability lost, but the changes are reversible.

Combined Loads Orbiter Tests

Simulations of the time histories of Shuttle ascent loads on tiles bonded to "real" structures were conducted in the Langley 8-Foot Transonic Pressure Tunnel, a type of test never before attempted in a wind tunnel of this size. Simulation concepts and hardware were developed for three separate areas on the bottom of the Shuttle; only the two declared critical for the first flight were tested. The two areas are located ahead of, and behind, the forward external-tank/Shuttle orbiter connection yoke (or bipod) and are among the most critical on the Shuttle. The extremely high local heating rates can be amplified if a smooth surface is not maintained over this region. In addition, the heat of entry could be increased over the downstream areas to the point where the structural integrity of the aerospacecraft could be jeopardized.

The test panel behind the yoke (see photograph) was selected because of the high buffet loads induced by the unsteady wake of the yoke, the high vibroacoustic loads, and the possibility of a "step and gap" problem between the tiles due to these loads. The structure on this area is of the skin/stringer variety. The panel ahead of the yoke includes the doors of the forward landing gear compartment. These doors are constructed of a thick honeycomb sandwich material and are so rigid that structural vibrations were not a concern in this area. However, buffet loads due to extremely high unsteady pressure levels caused by the supersonic bow shock were a concern.

The successful completion of these tests not only contributed greatly to the degree of confidence for the first Shuttle flight, but also marked a major milestone in the application of a large transonic tunnel to conduct tests on flight articles such as the Shuttle tiles with the following features: (1) combination of structural aerodynamic and acoustic loads, (2) time simulation of loads, and (3) diagnostic loads and response data.

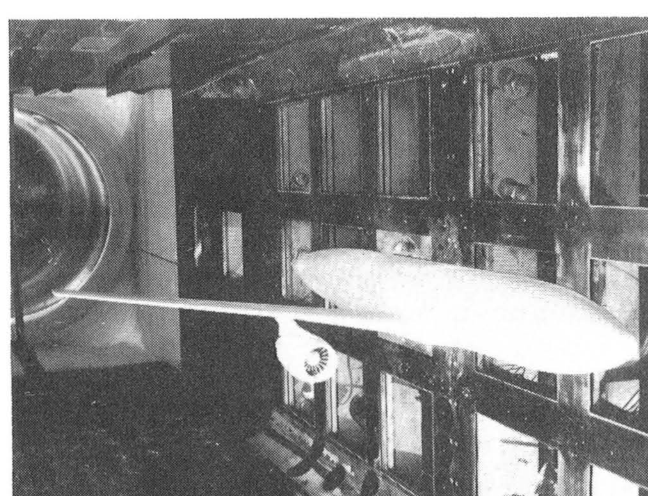


Combined Loads Orbiter Test

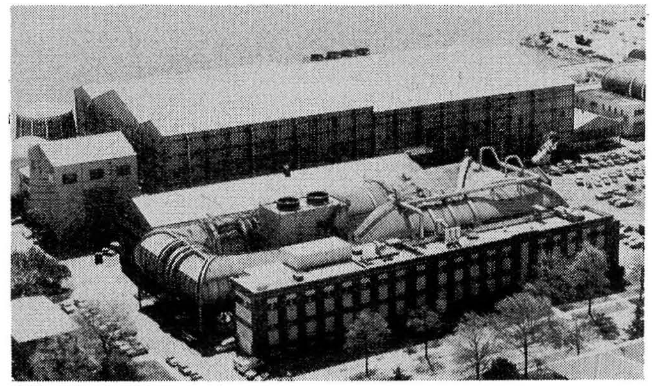
Propulsion/Airframe Integration Investigation

The proper integration of the aircraft propulsion system is one of the most important aspects of the development of a high performance and economic aircraft. A propulsion/airframe integration investigation is currently under way at Langley to determine the most favorable turbofan nacelle-pylon design for a proposed energy efficient transport aircraft configuration. The large engine size relative to the local wing chord of this configuration, due to a high aspect ratio supercritical wing, increases the difficulty of engine installation compared to aircraft of the past.

A semispan powered engine model transport configuration was tested while mounted on the side wall of the Langley 8-Foot Transonic Pressure Tunnel. A balance located at the center of the fuselage is used to measure the aerodynamic forces on the model. High-pressure air to drive the model engine is piped through the fuselage, through the supercritical wing, and down the pylon to the engine. When a powered engine such as this is used, it is possible to obtain the effect of the jet engine wake on the lift and drag of the model. Several tests were conducted with mixed flow and separate flow turbofan nacelles installed at various longitudinal and vertical positions relative to the wing. The effect of symmetrical, cambered, cant angle and area ruled pylons on interference drag has been determined. These data indicate that a sizable decrease in drag may be obtained by cambering and area ruling the pylon of the separate flow turbofan nacelle as a result of a reduction in the compressibility effects associated with engine installation.



Propulsion/Airframe Integration Tests



Transonic Dynamics Tunnel

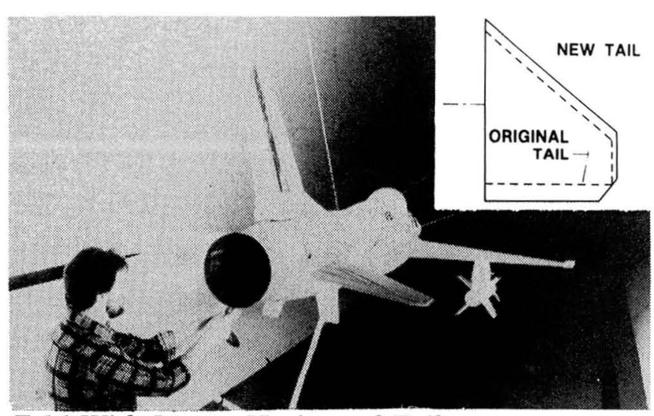
Conversion of the original Langley 19-Foot Pressure Tunnel into the Transonic Dynamics Tunnel was started in the late 1950's to satisfy the need for a large transonic wind tunnel dedicated specifically to work on dynamics and aeroelastic problems associated with the development of high-speed aircraft. Since the facility became operational in 1960, it has been used almost exclusively for clearing new designs for safety from flutter and buffet, evaluating solutions to aeroelastic problems, and researching aeroelastic phenomena at transonic speeds.

The tunnel is a slotted-throat, single-return, closed-circuit wind tunnel that has a 16-ft-square test section. The stagnation pressure can be varied from slightly above atmospheric to near vacuum. The Mach number can be varied from 0 to 1.2. Both test section Mach number and density are continuously controllable. The facility can use either air or Freon 12 as the test medium. Freon is usually used because it has several advantages over air as a test medium for dynamically scaled aeroelastic model testing. The tunnel has a Freon reclamation system so that the gas can be purified and reused.

The facility is equipped with many features uniquely suited for dynamic and aeroelasticity testing. These include a computerized data acquisition system especially designed to rapidly process large quantities of dynamic data, a means of rapidly reducing test section Mach number and dynamic pressure to protect models from damage when aeroelastic instabilities occur, a system of oscillating vanes to generate sinusoidal variations in tunnel flow angle for use in gust response studies, and special mount systems which enable simulation of airplane free-flight dynamic motions.

F-16 With Larger Horizontal Tails Shown Flutter Free

In order to improve the stability and maneuverability of the F-16 airplane, new horizontal tails are being built which are 30% larger than the old production tails. As part of the flutter clearance program for the modified F-16 airplane, the 1/4-scale full-span, cable-mounted F-16 flutter model equipped with the larger tails was tested in the Transonic Dynamics Tunnel (TDT). The F-16 airplane has a dual hydraulic system that drives each horizontal tail independently. The model was tested with a stabilizer pitch spring that simulated



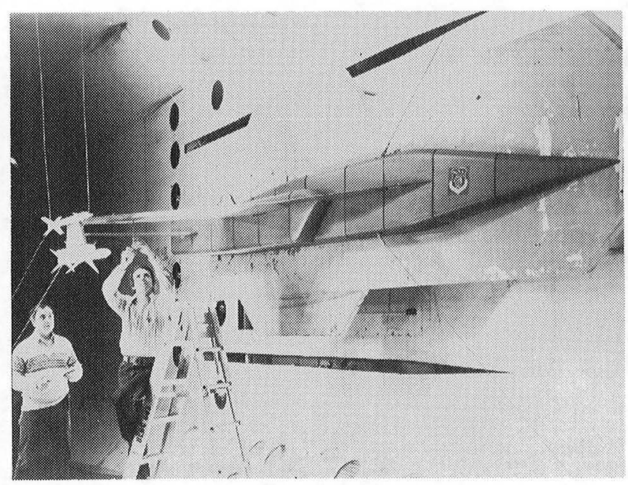
F-16 With Larger Horizontal Tail

the dual hydraulic actuator stiffness. This configuration was shown to be flutter free to speeds at least 20% above the limit speed envelope. The model was also tested with the pitch spring stiffness reduced to simulate the failure of a single hydraulic actuator on each horizontal tail. This configuration was shown to be flutter free to speeds greater than 15% above the limit speed envelope. The result satisfies the flutter margin requirement for the failed configuration.

Fighter Aircraft Flutter Suppression Systems

Fighter aircraft are required to carry a large variety of external wing-mounted stores. Some store configurations cause reductions in flutter speed (or dynamic pressure) and, as a consequence, place restrictions on the aircraft's operating envelope. To avoid such restrictions, considerable research by NASA, the Air Force, and industry has gone into development and wind-tunnel testing of methods of active flutter suppression. The basic approach is to sense motion of the wing using several accelerometers, send the accelerometer signals to a computer which implements a control law, and then feed the control law output to one or more hydraulically actuated control surfaces which move to provide aerodynamic damping to suppress flutter.

A recent test in a series of NASA/Air Force Wright Aeronautical Laboratory (AFWAL) cooperative research programs on wing/store flutter suppression, conducted in the Langley Transonic Dynamics Tunnel, employed a Northrop-built semi-span aeroelastic YF-17 model. The control laws that previously had been implemented using an analog computer were implemented with a digital computer. This was done because on aircraft it is desirable to use the more versatile existing digital computers. Damping tests were performed at a constant Mach number with variable dynamic pressure. The results indicated that the dynamic pressure at which flutter occurs is doubled by using this active flutter suppression system.

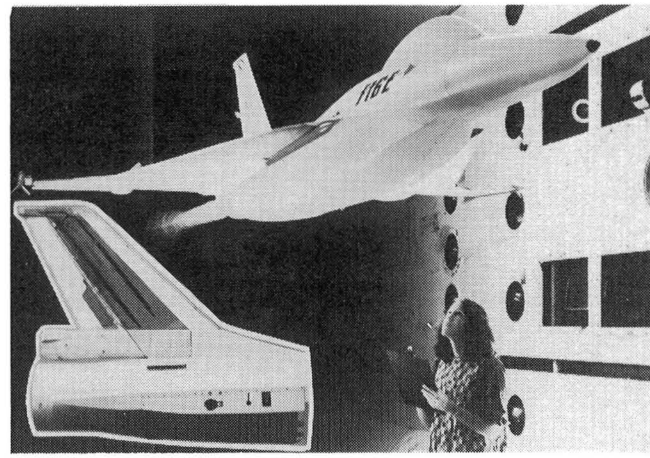


Fighter Aircraft Flutter Suppression

F-16E Vertical Tail Flutter Test

A new supersonic cruise version of the F-16 fighter airplane, designated the F-16E, is being developed by General Dynamics. It features an advanced technology wing that offers significantly improved aerodynamic performance over the present F-16 airplane. These F-16E airplanes will use substantially the same basic vertical tail as the Norwegian version of the F-16.

As part of the flutter clearance for these F-16E airplanes, transonic flutter tests of a 1/4-size model were made in the Transonic Dynamics Tunnel to demonstrate the flutter clearance of the F-16E vertical tail configuration. The purpose of these tests was to determine if the aerodynamic flow associated with the new wing could affect the flutter of the vertical tail sufficiently to reduce the flutter margin of safety required for flight. The model fuselage was mounted to a sting that extended forward into the simulated engine exhaust duct. The vertical tail model closely represented the dynamic characteristics of the full-scale article, but the model wings and fuselage were somewhat overstiff. The result of the flutter tests indicated that the vertical tail was free from flutter up to speeds greater than 20 percent above the limit flight speed envelope.

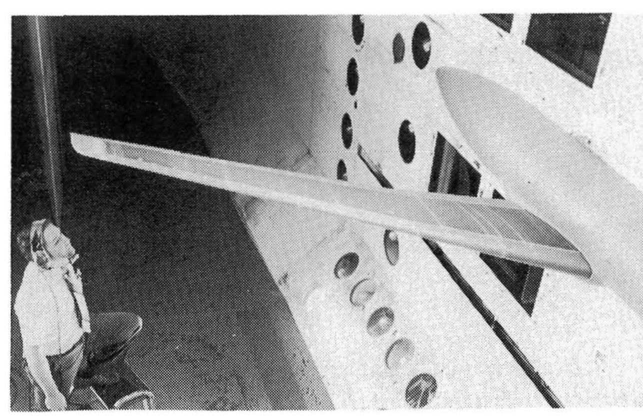


F-16 Vertical Tail Flutter Test

Supercritical Wing With Oscillating Control Surfaces

The design of active control systems for energy efficient transports with supercritical wings requires an understanding of both steady and unsteady transonic aerodynamics. Although considerable effort is being placed on developing methods for predicting unsteady transonic aerodynamics and significant progress has been made, no theoretical method has been developed to the point that it can be used to predict unsteady transonic loads reliably.

A series of tests on a semispan supercritical wing model equipped with leading-edge and trailing-edge oscillating control surfaces were conducted in the Transonic Dynamics Tunnel to provide a data base for use in designing active control systems and for use in validating transonic unsteady aerodynamic theories. The model has a sidewall-mounted half-body fuselage and the wing is instrumented with 252 static pressure orifices and 164 dynamic pressure gages. Measured chordwise distributions of lifting pressure magnitude and phase angle were obtained over a range of Mach numbers and Reynolds numbers. Model test variables included wing angle of attack, control-surface mean deflection angle, control surface oscillating deflection angle and frequency, and

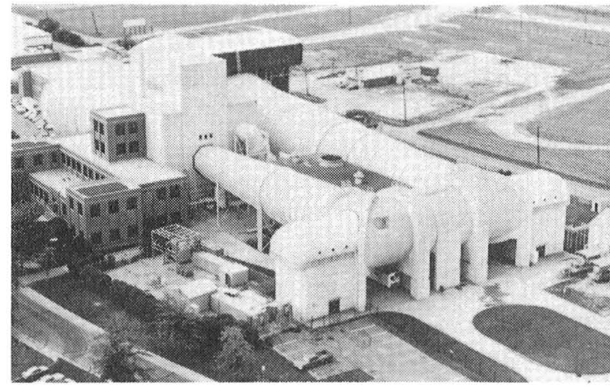


Instrumented Supercritical Wing

phasing between oscillating leading-edge and trailing-edge controls when used together.

The experimental results showed that unsteady lifting pressures generated by oscillating control surfaces are substantial. Inboard oscillating surfaces have significant influence on unsteady lifting pressure far outboard on the wing. Comparisons of measured data with calculated results obtained using subsonic lifting surface theory indicate a need for better prediction methods, particularly at transonic speeds.

16-Foot Transonic Wind Tunnel

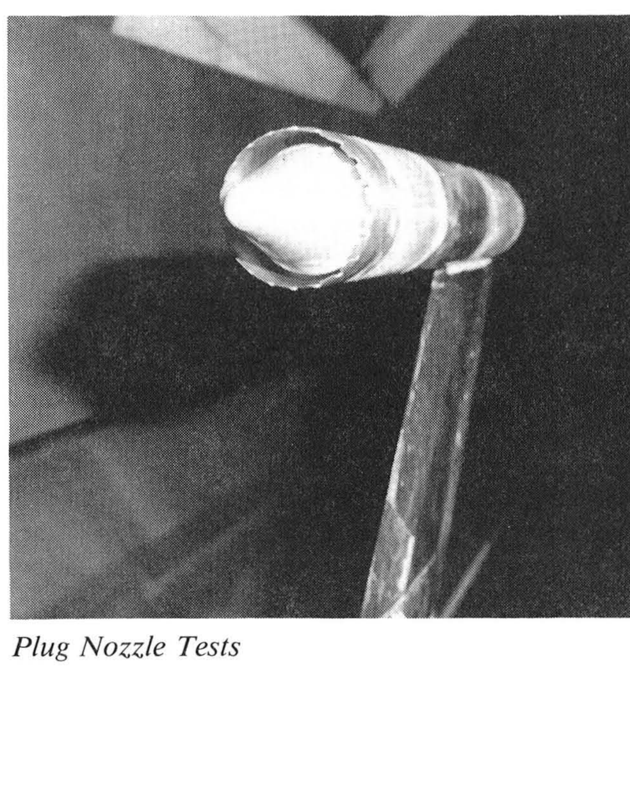


The 16-Foot Transonic Wind Tunnel is a closed circuit, single return, continuous flow, atmospheric tunnel. Speeds up to Mach number 1.05 are obtained with the tunnel main drive fans; speeds from Mach number 1.05 up to 1.30 are obtained with a combination of main drive and test-section plenum suction. The slotted octagonal test section measures 15.5 feet across the flats. The tunnel is equipped with an air exchanger with adjustable intake and exit vanes to provide some temperature control. This facility has a main drive of 60,000 hp. A 36,000 hp compressor provides test section plenum suction.

This tunnel is used for force, moment, pressure, flow visualization, and propulsion-airframe integration studies. Model mounting consists of sting, sting-strut, and fixed strut arrangements. Propulsion simulation studies are made utilizing dry, cold, high-pressure air.

Supersonic Exhaust Plug Nozzles

Tests have been conducted in the Langley 16-Foot Transonic Tunnel to determine the thrust-minus-drag performance of a series of annular plug nozzles mounted on an isolated pylon nacelle model. The models were designed to determine exhaust system aerodynamic performance characteristics at transonic speeds and to provide a data base for conducting meaningful trade studies with other types of exhaust systems. Exhaust flow was simulated by high pressure air supplied through the model strut. The experiment was conducted at Mach numbers up to $M = 1.2$. The angle of attack was maintained at 0° and the nozzle exhaust pressure ratio was varied up to 10. Model geometry variables include shroud boattail angle (12° to 28°), plug angle (10° to 20° half angle), and shroud exit extension length (0 to 76 percent of shroud external diameter). Forces and moments acting on the model were recorded by six-component strain



Plug Nozzle Tests

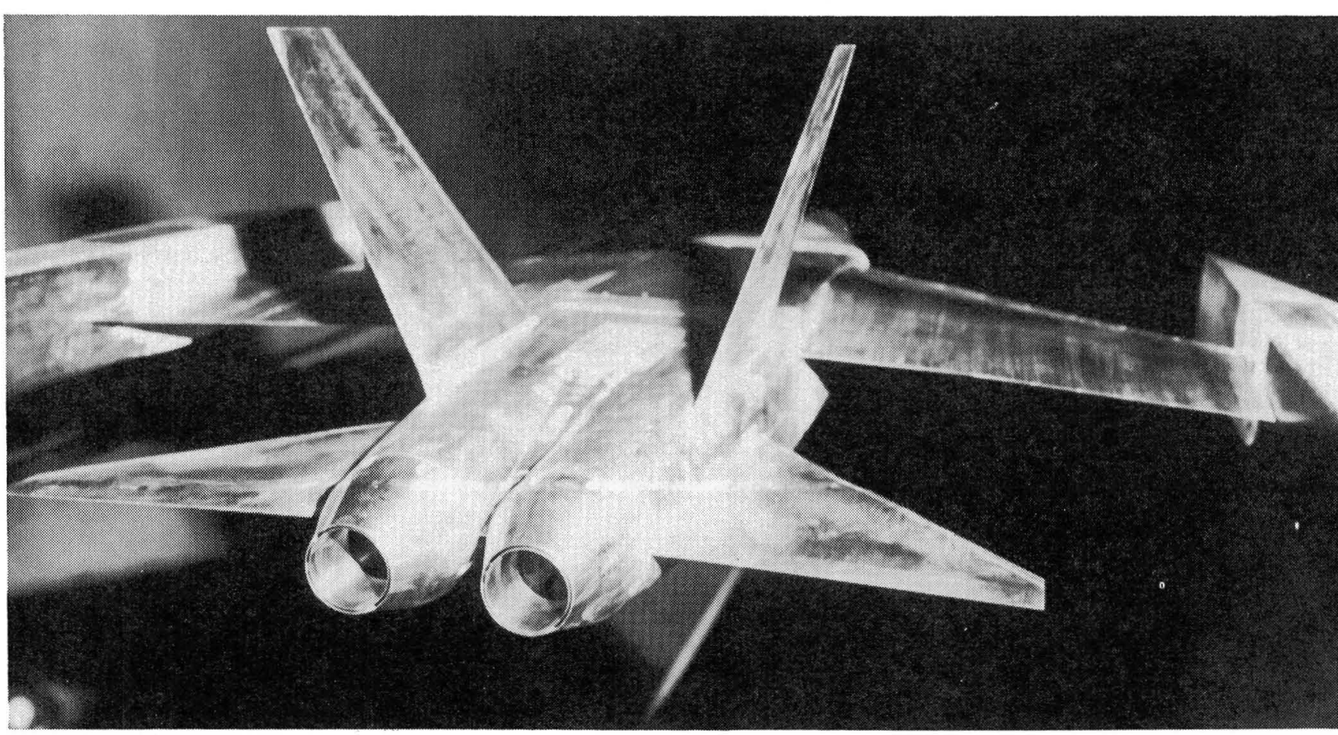
gage balance, and internal flow characteristics were recorded by pressure instrumentation.

Preliminary results at the subsonic cruise test condition indicate that the plug nozzle having the 15° half angle gave the highest thrust-minus-drag performance level. Extending the shroud exit position downstream from 0 to 76 percent of shroud external diameter reduced the performance by about 7 percent.

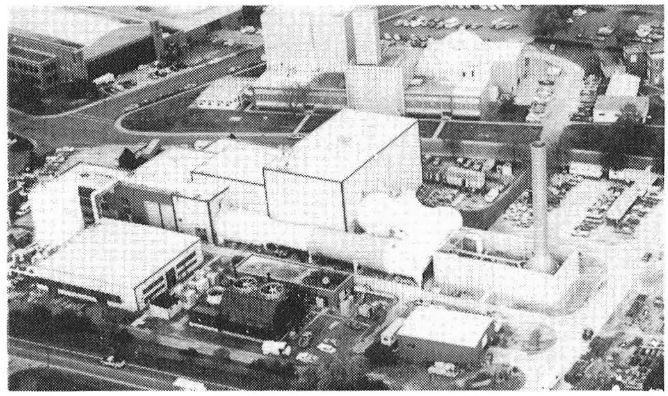
Twin-Engine Supercruise Nozzles

A research program has been conducted in the Langley 16-Foot Transonic Tunnel to determine the effect of axisymmetric nozzle design and control surface location on twin-engine nozzle/afterbody (afterbody includes horizontal and vertical tails) aerodynamic characteristics. Two variable-geometry nozzles, one designed for subsonic cruise (short nozzle length) and the other designed for supersonic cruise (long nozzle length), were simulated by three fixed-geometry test nozzles for each design. The nozzle test variables were nozzle length, power setting, or expansion ratio. The effect of vertical and horizontal tail axial location on nozzle/afterbody performance was also investigated with each test nozzle. The nozzle/afterbody aerodynamic characteristics were determined by force balance measurements. External nozzle pressures were measured on each configuration.

Comparison of nozzle configurations showed that nozzle/afterbody drag was generally decreased subsonically by increasing nozzle length. Supersonically, nozzle/afterbody drag was decreased by increasing expansion ratio. Although increasing nozzle length generally decreases nozzle/afterbody drag, nozzle weight would also be increased; thus, a trade-off study would be required to obtain the optimum nozzle length for an actual airplane design. Axial location of the vertical and horizontal tails was found to have only minimal effects on nozzle/afterbody aerodynamic characteristics.



Twin-Engine Supercruise Nozzle Tests



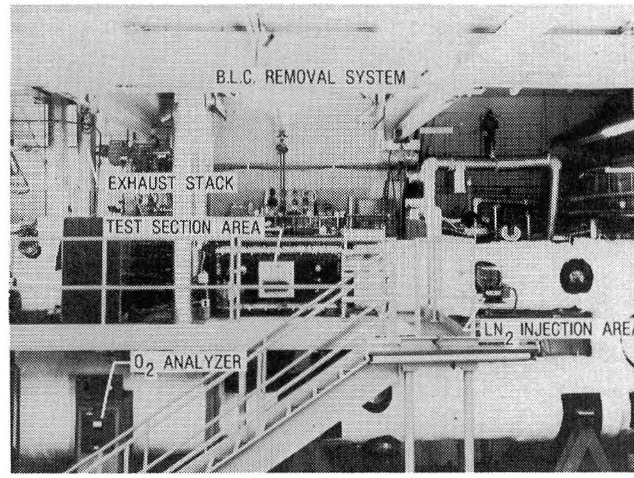
National Transonic Facility

The most difficult aerodynamic regime for aircraft designers to understand is the transonic region where speeds near Mach 1 (760 mph at sea level) are attained. At these speeds, the flow around aircraft is distorted by shock waves and the resulting turbulence decreases the lift and increases the drag in such complex patterns that designers cannot accurately predict the results. To develop a test facility that would allow full-scale testing of aircraft at such speeds would be very costly and require an enormous power supply.

NASA Langley's approach to this problem is to use nitrogen gas at high pressures and ultralow, cryogenic temperatures to simulate the transonic flow about full-sized aircraft. The principle that allows this simulation is that even if the sizes, speeds, and altitudes of two aircraft are very different, the aerodynamic properties of the flow about them are identical if the so-called Reynolds number (a parameter describing the flow which is a function of aircraft size and speed, as well as the density and viscosity of the flow) and the Mach number are the same for the two aircraft. By employing ultralow temperatures, the viscosity of the nitrogen gas is greatly reduced and Reynolds numbers will be achieved in a new tunnel, using small models, which will be identical to that characteristic of the airflow about full-sized aircraft in the real, more viscous atmosphere.

The new wind tunnel, the National Transonic Facility (NTF), is a cryogenic fan-driven transonic wind tunnel designed to provide full-scale Reynolds number simulation in the critical flight regions of most current and planned aircraft. It will operate at Mach numbers from 0.2 to 1.2, stagnation pressures from 1 to 9 bars, and stagnation temperatures from 340 to 80 kelvin. The maximum Reynolds number capability will be 120 million at a Mach number of 1.0 based on a reference length of 0.25 meters. Construction of the facility will be completed in mid-1982, with checkout and initial calibration by early 1983.

0.3-Meter Transonic Cryogenic Tunnel



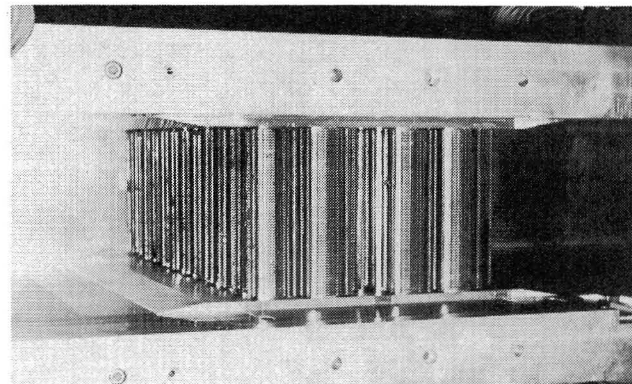
The Langley 0.3-m Transonic Cryogenic Tunnel (TCT) is a continuous flow fan driven transonic tunnel which uses nitrogen gas as the test medium. It is capable of operating at Mach numbers up to about 0.85, stagnation pressures up to 6 atmospheres, and stagnation temperatures from 340 to about 80 K. At the maximum test condition, a Reynolds number (based on a model chord of 15.24 cm) of 50×10^6 can be achieved. In its present configuration, a two-dimensional slotted wall test section is installed. The test section is 20 cm wide, 60 cm high and the slotted top and bottom walls have a 5 percent open area ratio. It is equipped with motorized model support turntables and a traversing wake survey probe, both of which are computer controlled.

This facility was first placed in operation in 1973 as a pilot three-dimensional tunnel for use as a proof of concept demonstration of the cryogenic test technique. The successful demonstration of that technique in the 0.3-m TCT played a major role in the decision by NASA to build the National Transonic Facility. In its present mode of operation, the TCT is used for routine airfoil testing at high Reynolds numbers, as a test bed for components and instrumentation for the NTF, and for advanced cryogenic testing techniques.

NTF Cooling Coil Test

A cooling coil is included in the design of the NTF circuit to provide economical testing at ambient temperature when liquid nitrogen is not used. The coil remains in the tunnel circuit for all testing conditions, including cryogenic operation. Consequently, it is necessary to investigate the aerodynamic effects of the coil at high Reynolds number conditions.

A model of the NTF cooling coil was tested in the 0.3-m TCT test section up to full-scale NTF Reynolds numbers. The primary purpose of the test was to determine the pressure drop across the



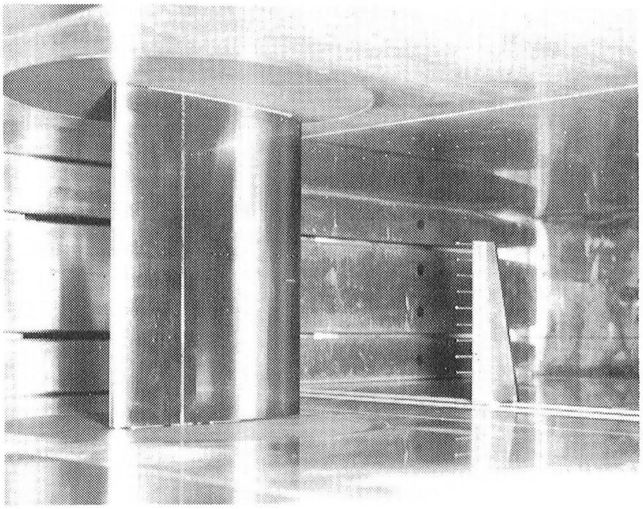
NTF Cooling Coil Test

coil. Although the tubes used to construct the model were actual NTF hardware, there was no coolant flow through the tubes for these tests and the data obtained are for the case of zero heat transfer. The pressure drop measured across the cooling coil at conditions corresponding to the maximum that would be encountered in the NTF is about 9 times the local dynamic pressure.

Onset of Airfoil Condensation in Cryogenic Tunnel

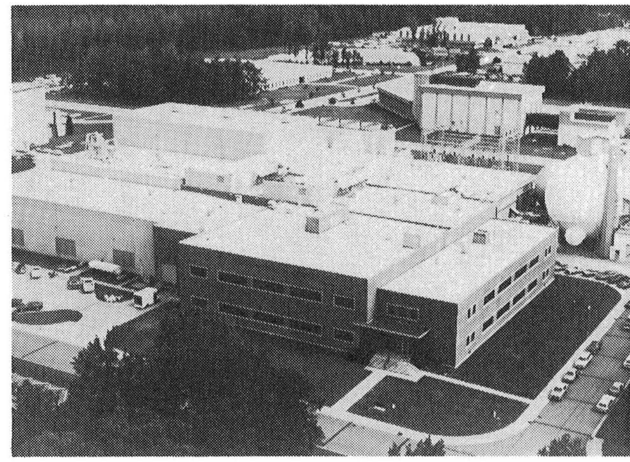
Cryogenic wind tunnels including the National Transonic Facility obtain their high Reynolds number capability by cooling the test gas; the lower the operating temperature, the greater the Reynolds number. Consequently, it is desirable to operate cryogenic tunnels as cold as possible to maximize Reynolds number through temperature reduction. The lower temperature limit, however, is fixed by the onset of condensation effects occurring either locally over the test model or in the tunnel test section itself. Although the saturation temperature (where condensation becomes possible) is known for the nitrogen test gas, the actual temperature at which condensation occurs is not easily predicted due to the time-dependent nature of the process. The objective of this test was to determine the onset of condensation effects in the pressure distribution of a supercritical airfoil.

The procedure followed was to test the airfoil in the 0.3-m TCT with constant free-stream Mach number, angle of attack, and airfoil Reynolds number. While holding these values constant, operating temperatures were reduced from starting values above those where saturation occurred locally over the model down to temperatures below where saturation occurred in the section itself. The results were then analyzed to determine at what temperature condensation effects occurred. The process was repeated for different free-stream Mach numbers, angles of attack, and airfoil Reynolds numbers. The results indicate that the onset of condensation is a function of local Mach number over the airfoil. Thus, the minimum operating temperature for cryogenic wind tunnels is dependent on the geometry of the model being tested. The data will be used to validate computer routines for theoretically predicting the onset of condensation.



Tests for Onset of Airfoil Condensation

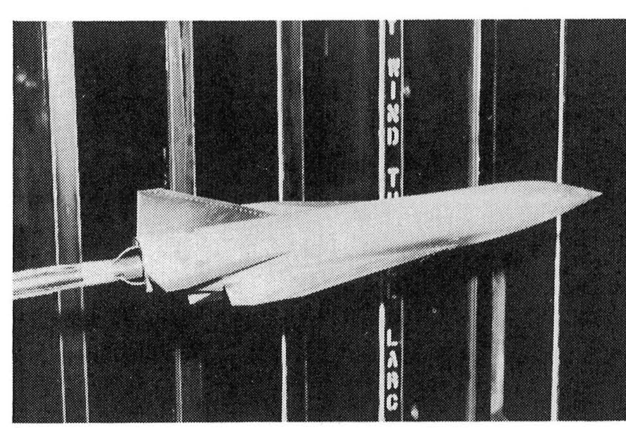
Unitary Plan Wind Tunnel



Immediately following World War II the need was recognized for wind-tunnel equipment to develop advanced airplanes and missiles. The military and the National Advisory Committee for Aeronautics (NACA) developed a plan for a series of facilities which was approved by the U.S. Congress in the Unitary Wind Tunnel Plan Act of 1949. This plan included five wind-tunnel facilities, three at NACA laboratories and two at the Arnold Engineering Development Center. The Langley Unitary Plan Wind Tunnel was among those three built by NACA. The Unitary Plan Wind Tunnel is a closed circuit, continuous flow, variable density type tunnel with two 4-foot by 4-foot by 7-foot test sections. The low range test section has a design Mach number range from 1.5 to 2.9 and the high range test section from 2.3 to 4.6. The tunnel has sliding block type nozzles which allow continuous variation in Mach number while on-line. The maximum Reynolds number per foot varies from 6×10^6 to 11×10^6 depending on Mach number. The tunnel is used for force and moment, pressure distribution, jet effects, dynamic stability, and heat-transfer studies. Flow visualization data are available in both test sections including schlieren, oil flow, and vapor screen.

Hypersonic Surface-to-Air Missile Tests

A 0.15 scale model of the airbreathing second stage of a hypersonic surface-to-air missile was tested in the Unitary Plan Wind Tunnel to determine longitudinal and lateral-directional stability of the second stage. The model could be tested without the simulated scramjet engine, with a constant area duct version of the engine or with the engine including actual compression surface. The latter configuration represents the vehicle in the power-off descent mode of the flight profile.



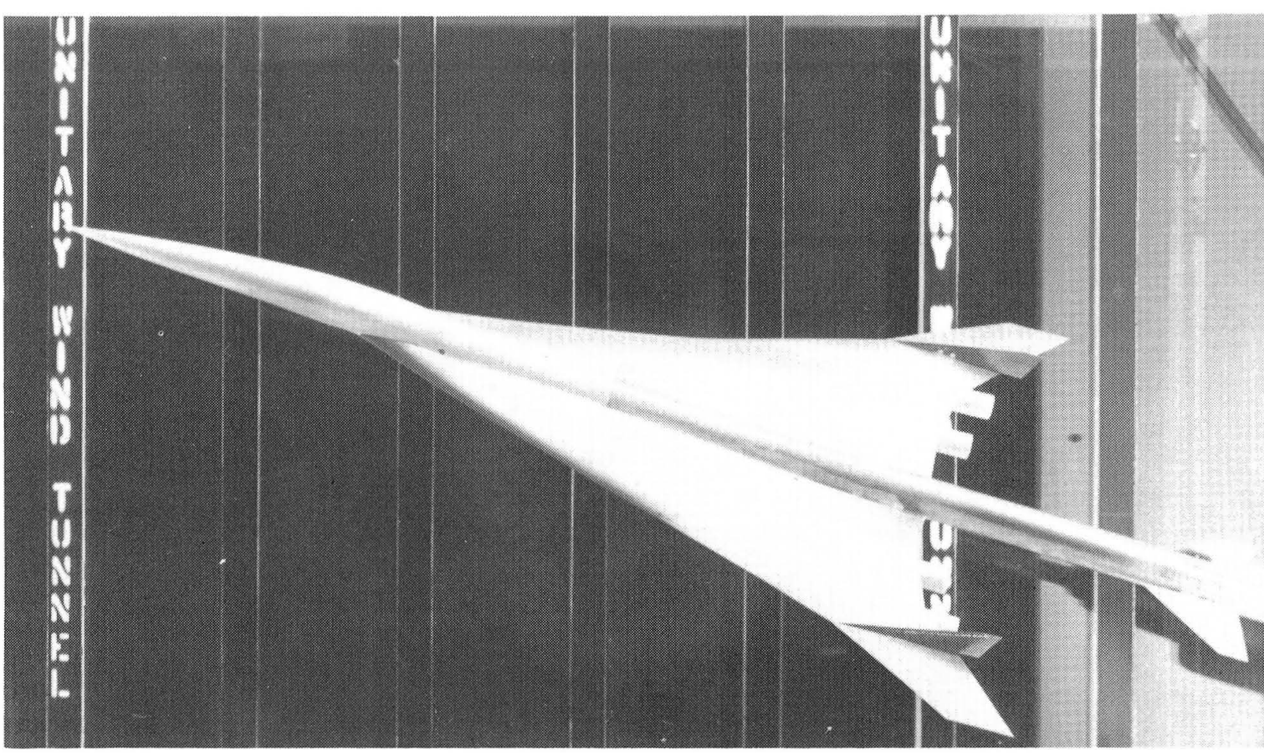
Hypersonic Surface-to-Air Missile Tests

The results showed that the model appeared to have pitchup problems at high angles of attack due to loss in wing or canted fin effectiveness, but this problem may not be serious because the full-scale vehicle operates at very high dynamic pressures and, therefore, flies at very low lift coefficients.

Advanced Supersonic Transport 200 Series

A baseline model consisting of a five-abreast passenger-seating advanced supersonic transport (AST) model was tested in the Unitary Plan Wind Tunnel. The configuration for the AST 200 series utilized advanced structural concepts that allowed the wing-fuselage to be extensively blended. The wing camber design and engine nacelle integration were performed using the latest numerical methods. Analytical evaluations indicated that the aerodynamic performance of this configuration should exceed that of any AST previously tested. Along with the baseline fuselage, two additional fuselages were tested to verify the predicted aerodynamics performance. The fuselages provided for a four- and six-abreast seating arrangement that formed a "family" with the baseline configuration. These fuselage arrangements could be manufactured from the baseline by the addition or deletion of lateral spacers to form a "family" of configurations, similar to that done for subsonic transports by utilizing longitudinal spacers. Longitudinal spacers are not feasible for supersonic aircraft because of the area ruling required for low wave drag. Theoretical analysis indicated that the lateral spacer concept for a supersonic family would result in little change for the cruise aerodynamics.

Models that were 1.06 percent of full scale were tested in the Mach range from 1.7 to 2.86. Both longitudinal and lateral aerodynamic force characteristics were measured. The preliminary results verify the analytical predictions and show that the cruise aerodynamic characteristics are approximately constant for the family of aircraft.



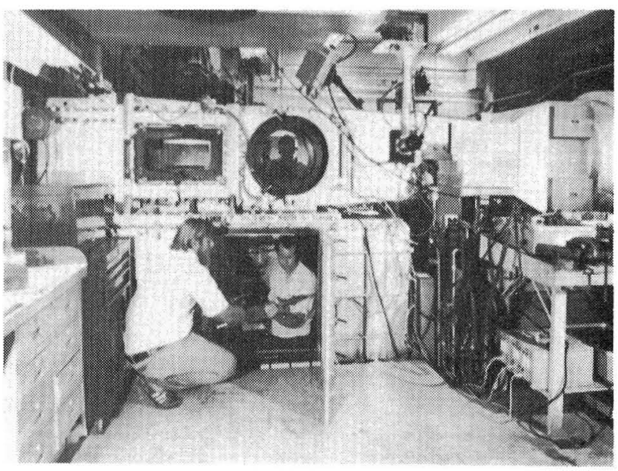
Advanced Supersonic Transport 200 Series

Hypersonic Facilities Complex

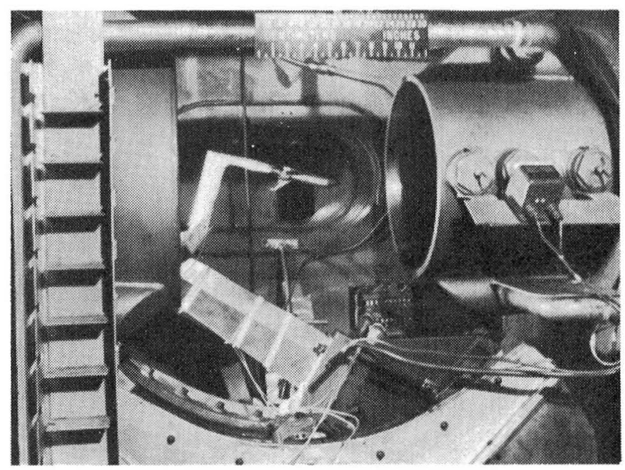
The Hypersonic Facilities Complex is used for studying heat transfer, aerodynamics, and viscous interactions of winged reentry vehicles, planetary probes, or hypersonic cruise aircraft and missiles. The Complex includes the 20-Inch Mach 6 Tunnel, the Hypersonic Helium Tunnel, the Hypersonic Nitrogen Tunnel, the Continuous Flow Hypersonic Tunnel, the Hypersonic CF_4 Tunnel, and the impulse facilities which include the Expansion Tube and a Shock Tube.

Tests conducted in the Continuous Flow Tunnel at Mach number 10 and the 20-Inch Tunnel at Mach number 6 are used to study Mach number effects in air. The CF_4 Tunnel also operates at Mach 6, so the effects of density ratio across normal shocks can be delineated. The Helium Tunnel ($M = 20$) and the Nitrogen Tunnel ($M = 19$) provide entry Mach numbers over a range of Reynolds number. The Expansion Tube provides real gas effects at entry velocities while the Shock Tube is used to measure radiative and gas kinetic properties in planetary entry.

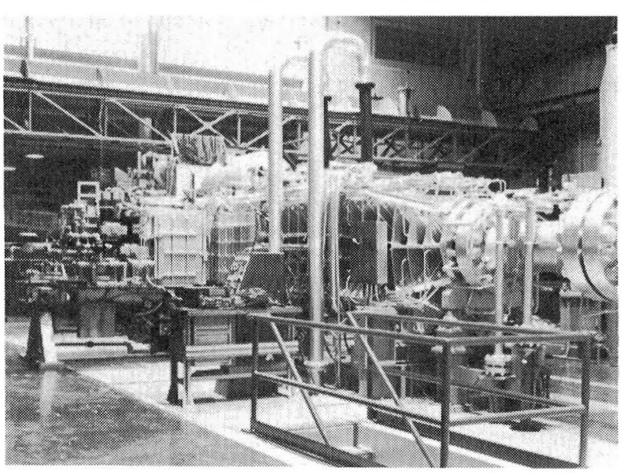
20-INCH MACH 6 TUNNEL



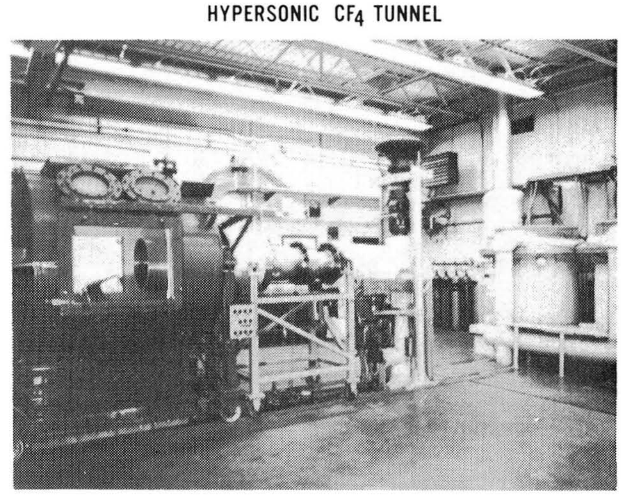
HYPERSONIC NITROGEN TUNNEL



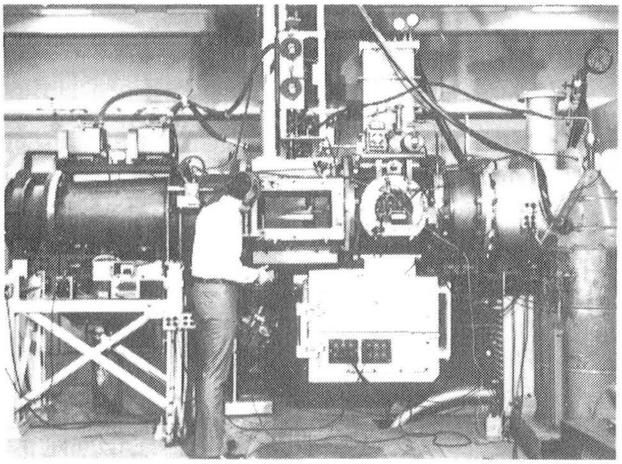
CONTINUOUS FLOW HYPERSONIC TUNNEL



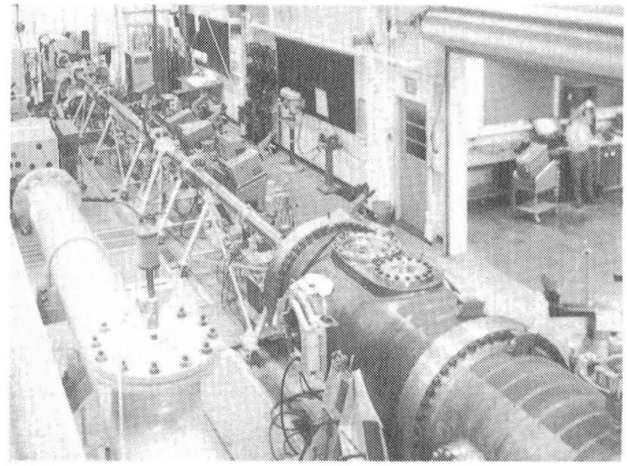
HYPERSONIC CF_4 TUNNEL



HYPERSONIC HELIUM TUNNEL



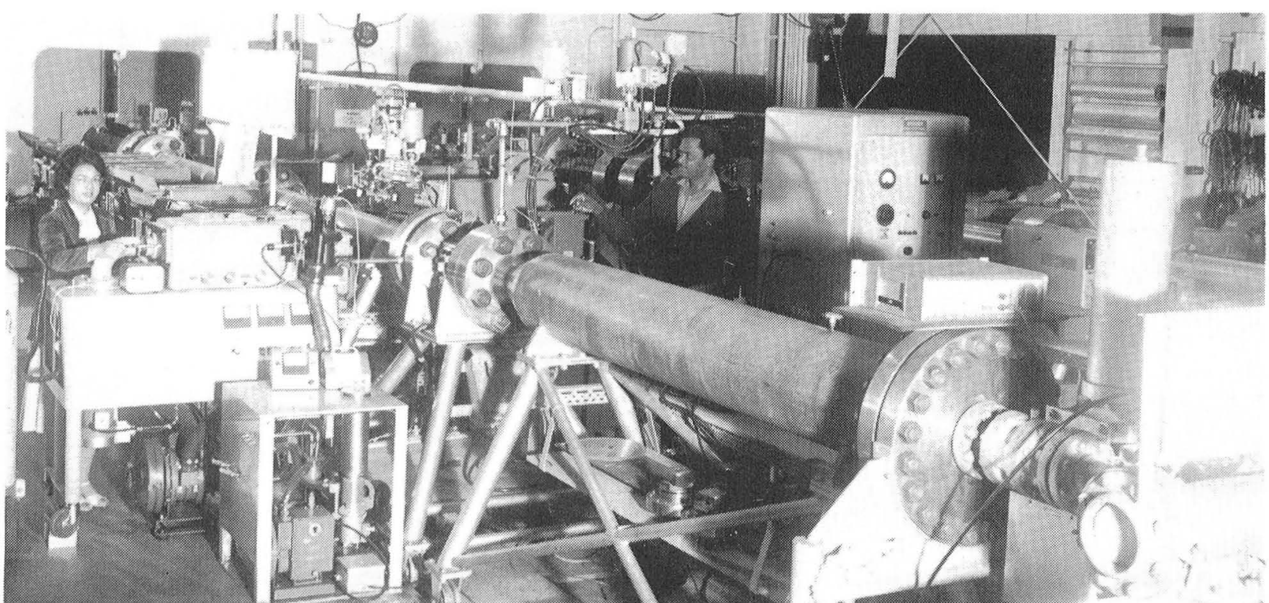
LANGLEY EXPANSION TUBE



Hypersonic Facilities Complex

First Measurement of C_3 Ultraviolet Band System

A study was conducted in the Langley 6-Inch Shock Tube to measure the radiative absorption of carbon and hydrocarbon species in the vacuum ultraviolet wavelength range. This study was part of a general investigation of the absorption properties of those species which are prevalent in the ablation gases emitted from phenolic carbon heat shields. In the planned Galileo mission to Jupiter, the Galileo probe will enter the planet's atmosphere at extremely high velocity, causing intense radiative heating and thus rapid ablation of the probe heat shield. In assessing the heating rate and required heat shield thickness, it is important to account for the portion of the incident radiation which is absorbed by the gases being emitted from the heat shield.



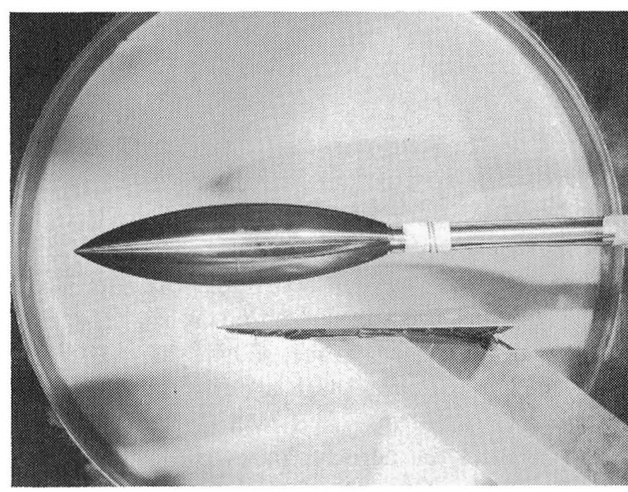
6-Inch Shock Tube Tests

In the present study, the hydrocarbon species of interest were produced by shock-heating acetylene-argon or methane-argon mixtures. Using absorption spectroscopy, the wavelength range from 130 to 300 nanometers was surveyed. A strong absorption band was found which extended from about 150 to 190 nm. By varying the concentration of species, it was determined that the absorption band was due to triatomic carbon, C_3. Inclusion of this information into radiative heating predictions for the Galileo probe reduces the expected probe heating temperature by about 6-8%.

Interference Lift Between a Separately Mounted Body and Wing

Recently several new concepts employing favorable aerodynamic interference have been proposed for improving the aerodynamic efficiency of high-speed aircraft concepts. Among them, the parasol wing arrangement shows promise of delivering substantial benefits of increased lift from wing/body interference. Models were built with wing and body separately mounted to eliminate interference due to the wing support, and were then tested in the 20-Inch Mach 6 Tunnel. The proper design of the wing support is vital to preserving any benefits that may be gained by this unique body-wing arrangement.

Force and pressure data were obtained on the body, as well as pressure data on the surface of the wing adjacent to the body at zero angle of attack for a range of body/wing separations. Flow visualization techniques (oil flow, vapor screen, and schlieren) were used to examine the viscous interaction between the wing and body and to observe the flow field about the wing/body arrangement. Results indicate that increased lift is obtained on the wing, but the viscous effects due to flow separation and shock impingement on the body cancel this benefit as evidenced by increased drag on the body. Because the body and wing are separately mounted, these data will be of interest to investigators examining the impingement of shock waves on cylindrical bodies. The vapor screen and oil flow data have already been requested by several investigators studying store separation at high speed.

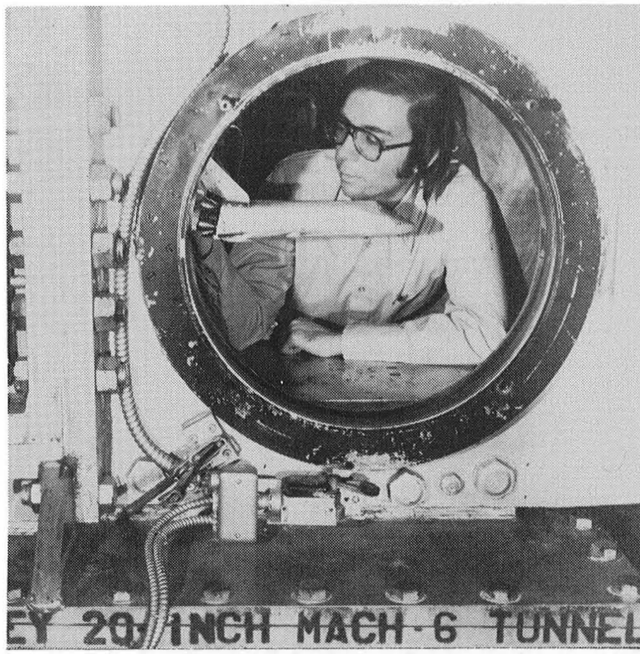


20-Inch Mach 6 Tunnel Tests of Lift Interference

Verification of Shuttle Orbiter Entry Aerodynamics

A series of force and moment tests on a 0.01-scale Space Shuttle Orbiter model in the Langley 20-Inch M-6 Tunnel was completed. These tests were designed to verify the hypersonic lateral-directional stability characteristics of the Orbiter and to provide tunnel-to-tunnel and model-to-model repeatability before the first flight. Earlier tests on a 0.004-scale Orbiter model in the same tunnel revealed a critical lateral-directional instability for near-flight conditions. To identify the source of the measured instability, the hypersonic data were re-investigated with the 0.01-scale model, particularly to determine the effects of Reynolds number variation.

The Langley tests complemented two others at AEDC which used the same model as well as a 0.02-scale Orbiter to achieve the desired Reynolds number match. Over 300 runs were conducted including those designed to verify tunnel flow quality. Results from this series of tests compared favorably with lateral-directional stability results from AEDC and with the Orbiter Aerodynamic Design Data Book for the highest test Reynolds number, demonstrating that the instability is not present at flight Reynolds numbers and therefore increasing the confidence level in the entry hypersonic aerodynamics for the Orbiter.

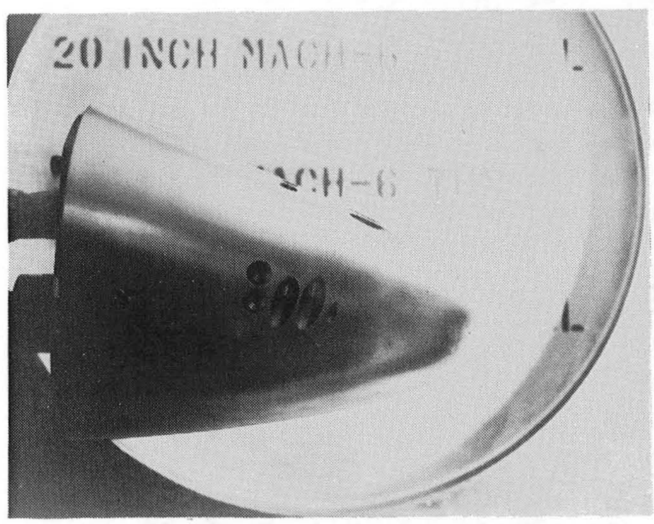


Shuttle Orbiter Entry Aerodynamics

Shuttle Entry Air Data System

The Shuttle Entry Air Data System (SEADS) is an across-the-speed-range air data system to be installed on the Space Shuttle Orbiter. The SEADS consists of an array of flush pressure orifices installed in a baseline geometry orbiter nose cap. The nose cap orifices are supplemented by six forward fuselage static pressure orifices. This system will be used to determine angles of attack and sideslip and free stream conditions, and will support flight research studies in aerodynamics and aerothermodynamics. In support of the development of SEADS, a 0.04-scale model of the Orbiter's forebody was tested in the Mach 10 Continuous Flow Hypersonic Tunnel, the 20-Inch Mach 6 Tunnel, and the Unitary Plan Wind Tunnel. The model was instrumented with 72 pressure orifices including the 20 in the SEADS and 17 to match instrumentation currently on the forward fuselage of the Orbiter. The remaining orifices were located to assist in data analysis and correlative comparisons with previous wind-tunnel Orbiter tests.

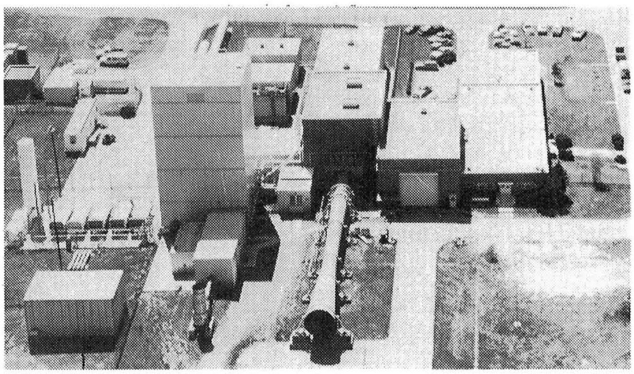
Data from the tests are being used to support software development for SEADS data reduction.



Shuttle Entry Air Data System

Since the model was also instrumented to match current instrumentation on the Orbiter, comparisons were made between Shuttle flight data, wind-tunnel data, and analytical predictions by the Langley HALIS (High Alpha Inviscid Solution) computer code. The results were in excellent agreement.

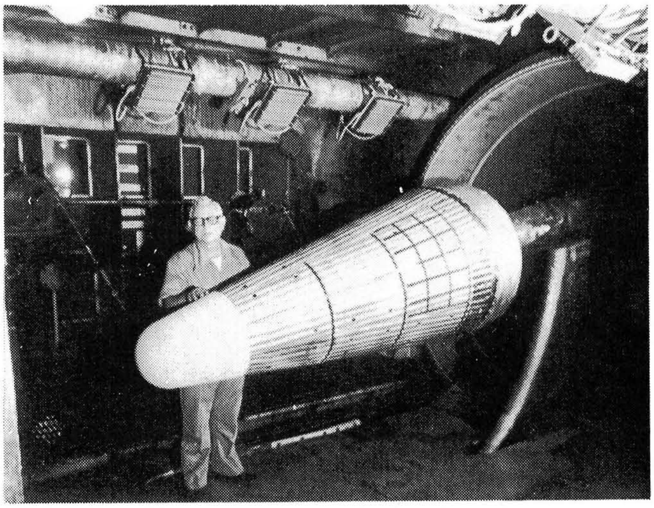
8-Foot High-Temperature Structures Tunnel



The 8-Foot High-Temperature Structures Tunnel (8' HTST) is a blowdown-type facility which achieves the required energy level for flight simulation by burning methane in air under pressure and using the resulting combustion products as the test medium with a maximum stagnation temperature near 3800 °R. The nozzle is an axisymmetrical conical-contoured design with an exit diameter of 8 ft. Model mounting is semispan or sting with insertion after the tunnel is started. A single-stage air ejector is used as a downstream pump to permit low-pressure or high-altitude simulation. The Reynolds number ranges from 0.3 $\times 10^6$ to 3 $\times 10^6$/ft with a nominal Mach number of 7. The run time ranges from 20 to 180 sec. The tunnel is used for studying detailed thermal loads and flow phenomena of high-speed and entry vehicle structural components.

Aerodynamic Heating on Corrugated Surfaces

The thermal protection system (TPS) for space transportation vehicles should be reusable and should require a minimum amount of repair. Previous studies have demonstrated the capability of two-dimensional metallic TPS surfaces with the next logical step being to determine the performance of a curved or three-dimensional surface. As a result, aerodynamic heating tests were made in the Langley 8-Foot High Temperature Structures Tunnel on a large 10.2° half-angle blunted cone model having corrugation-stiffened metallic panels distributed over the cone surface. The tests were made at a nominal Mach number of 6.7, at total temperatures of about 1850 K, and over a model angle of attack range from 0 to 10 degrees.



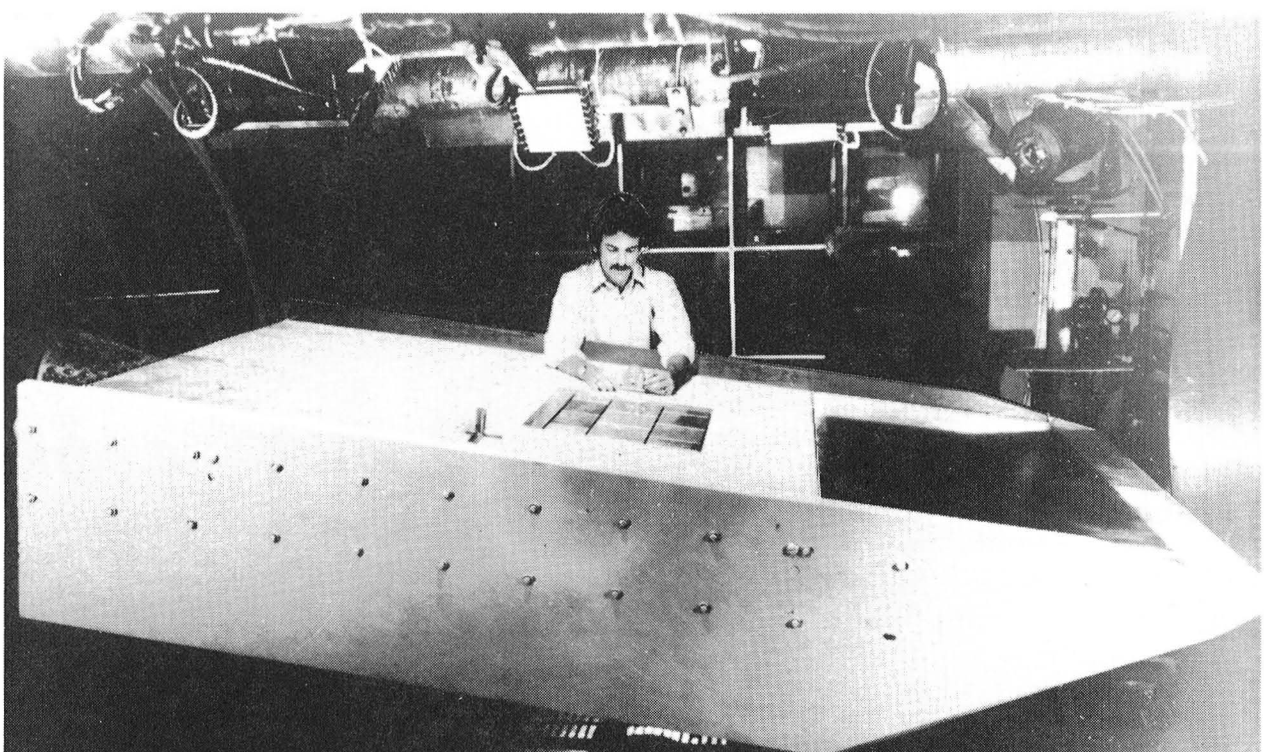
Aerodynamic Heating on Corrugated Surfaces

The results indicate that the heating over corrugated three-dimensional surfaces can be reasonably predicted by the theory for smooth surfaces; any discrete effects of the corrugations are probably washed out by flow disturbances due to panel overlap and other surface discontinuities. Any effects of the corrugations, however, are small and there appear to be no excessive penalties imposed due to the use of corrugated surfaces.

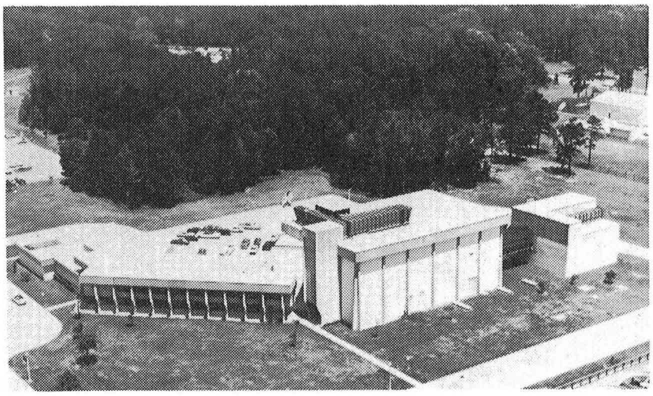
Flow Angularity Effects on Tile/Gap Impingement Heating

Results from analyses performed on the Space Shuttle thermal protection system (TPS) indicate that air flow in the thermal expansion gaps between the TPS tiles contributes approximately 20 percent of the heating to the Shuttle structure. Therefore, experimental investigations of tile/gap impingement heating were undertaken at Langley. Previous aerothermal tests on Shuttle type tiles in the LaRC 8-Foot High-Temperature Structures Tunnel identified the effects of boundary layer and gap geometry on the flow impingement heating rate on the tile's forward face at the end of a longitudinal gap aligned with the flow ("T" gap). The present study extends the previous effort to include the effect of impingement heating due to flow angularity with respect to the longitudinal gap. In addition, the boundary layer state (laminar or turbulent) and thickness, Reynolds number, and gap width were also varied. The determination of localized impingement heating is important because it affects the tile coating, whereas the overall tile heating affects the Shuttle's structural integrity. To obtain the overall tile heating distribution, a highly instrumented thin-wall metallic tile was tested in the 8' HTST.

Test results indicate impingement heating at the end of the "T" gap is significantly higher than tile surface heating at flow angles of 0° but reduces with increasing flow angle. As reported from previous tests for gap widths less than 0.070 inches (Space Shuttle tile gaps are nominally 0.045 inches in width), laminar flow over a tile array produces basically two-dimensional gap flow but turbulent flow produces primarily three-dimensional gap flow at the "T" junction. The turbulent boundary layer allows a larger energy transfer into the gaps, which produces the higher impingement heating for flow angles near zero. The effect of flow angularity on impingement heating from the present tests will be incorporated into an empirical relationship, developed from previous tests, which accurately predicts the effects of gap geometry over a range of boundary layer conditions.



Tests on Tile/Gap Impingement Heating



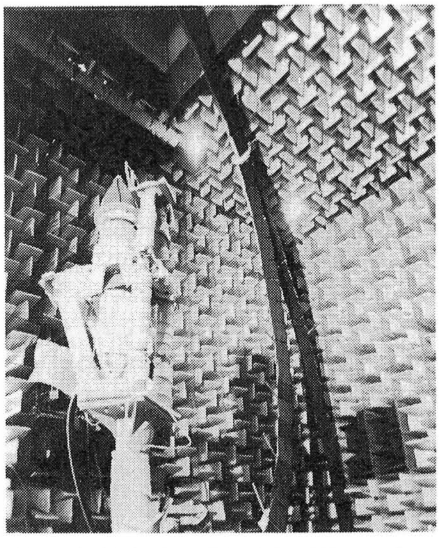
Aircraft Noise Reduction Laboratory

The Langley Aircraft Noise Reduction Laboratory consists of the Anechoic Flow Facility, Reverberation Chamber, Transmission Loss Facility, Anechoic Noise Facility and the Coannular Jet Noise Apparatus. The Anechoic Flow Facility has a test chamber lined with sound-absorbing wedges and is equipped with a low-turbulence low-noise test flow to allow aeroacoustic studies of aircraft components and models. The test flow, which is provided either by horizontal high-pressure or vertical low-pressure air systems, varies in Mach number up to 0.5. In contrast, the Anechoic Noise Facility is equipped with a very high pressure air supply used solely for simulating nozzle exhaust flow.

The Transmission Loss Facility has a source room and a receiving room joined by a connecting wall. The test panel, such as an aircraft fuselage wall, is mounted in the connecting wall for sound transmission loss studies. The Reverberation Chamber diffuses the source noise and is used for measuring the total acoustic power spectrum of the test source. The Coannular Jet Noise Apparatus has two coannular supersonic jets for studying turbulence evolution in the two interacting shear flows which are typical of high-speed aircraft engines.

Acoustic Shielding by Jet Flows

Acoustic shielding by jet flows is an attractive approach for jet engine propulsive noise reduction. Current pursuit of this concept in industries includes jet-by-jet shielding and thermal acoustic shielding. For optimal applications of the concept and accurate predictions of the noise benefits, the underlying physics of this shielding phenomenon must be identified. Measurements were made in the Anechoic Flow Facility to compare to an analytical model. The noise field of a "point source" shielded by a jet flow was measured by varying the frequency of the source, the jet velocity and the jet density. The jet density variation was achieved by using air and helium as the jet medium to simulate both cold and heated exhaust flows.



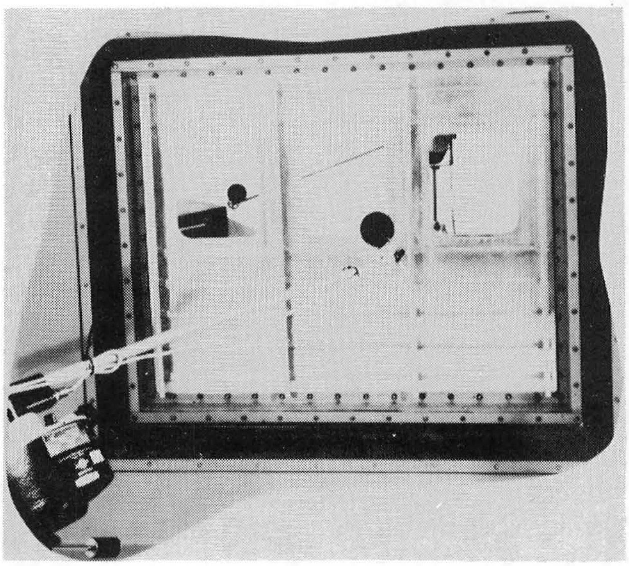
Anechoic Flow Facility Test of Acoustic Shielding by Jet Flow

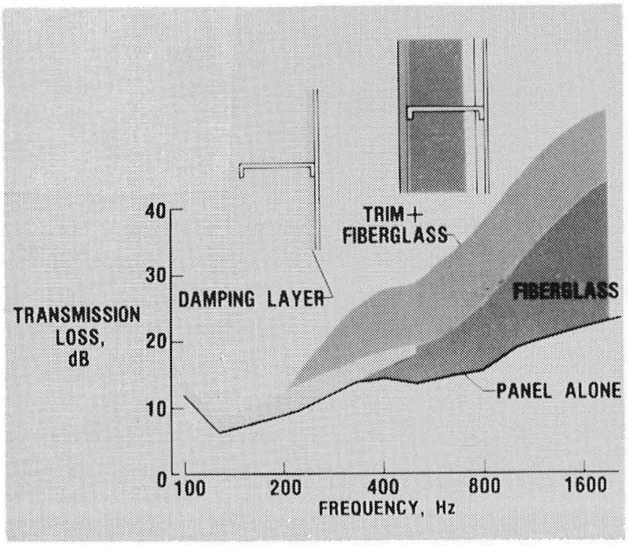
The experimental results indicate that flow shielding is a pure acoustic phenomenon in that the total radiated acoustic power is conserved. The primary mechanisms for observed noise reduction in the shadow zone of the shielding jet are wave refraction and diffraction. There is good agreement between the test data and the analytical predictions, indicating that acoustic shielding of jet engine noise by jet flows can reduce perceived noise levels by approximately 3 dB in the shadow zone.

Fuselage Sidewall Acoustic Treatments

In many aircraft, high interior noise levels result from the transmission of exterior noise through the fuselage sidewall. It is customary to add acoustic treatments to the sidewall to reduce the noise; however, weight constraints limit the amount of treatment that can be added. To aid in the search for low-weight but effective fuselage designs and acoustic treatments, tests were conducted in the Transmission Loss Facility for several panel/treatment combinations. Noise was generated in the source room and the space-averaged noise level in the source and receiving room was determined using the rotating boom microphones. Noise transmitted through the panel was obtained by subtracting the microphone signals.

The results indicate that the effectiveness of each treatment varies with frequency. For example, the "trim and fiberglass" combination is most effective at higher frequencies and provides over 40 dB of transmission loss in the frequency range of importance for conventional transports. Thus, the frequency spectrum of the input and transmitted noise should be considered in designing an appropriate acoustic treatment. Furthermore, none of the treatments provided much additional transmission loss at low frequencies of 100 to 300 Hz. Unfortunately, propeller tones in many light aircraft lie in this frequency range and dominate the noise spectrum. Transmission Loss Facility tests are continuing in an effort to seek acoustic designs and treatments that are more effective (but still lightweight) in this frequency range.

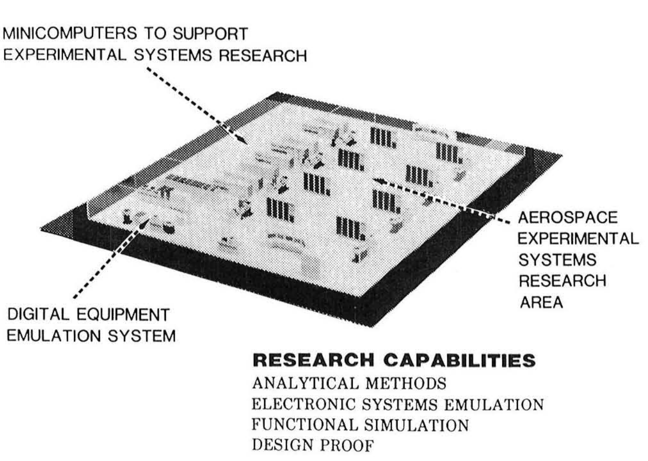




Acoustic Sidewall Tests

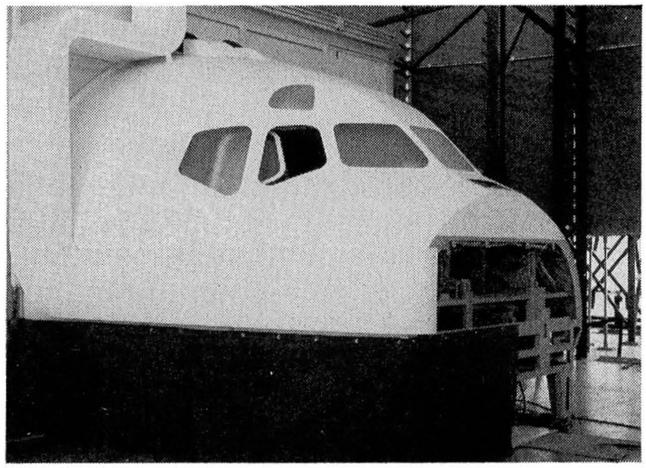
Avionics Integration Research Laboratory–AIRLAB

The United States leads the world in the development and design of commercial transport aircraft and is the major producer of tactical and strategic military aircraft and space vehicles. To maintain this leadership role throughout the 1990s and beyond will require the incorporation of the latest advances in digital systems theory and electronics technology into fully integrated aerospace electronic systems. This will necessitate new systems that can dramatically improve performance, lower production and maintenance costs, yet at the same time maintain a high, measurable level of safety for passengers and flight crews. The establishment of the Avionics Integration Research Laboratory (AIRLAB) at Langley Research Center is NASA's response to these needs. The AIRLAB will address major research problems associated with identifying and developing methods for systematically evaluating highly reliable, fully integrated digital aviation electronics and control systems. Using state-of-the-art computational resources, AIRLAB will provide a unique setting where participants will study, evaluate, simulate, and demonstrate the safety, reliability, performance, and economics of fully integrated avionics systems.



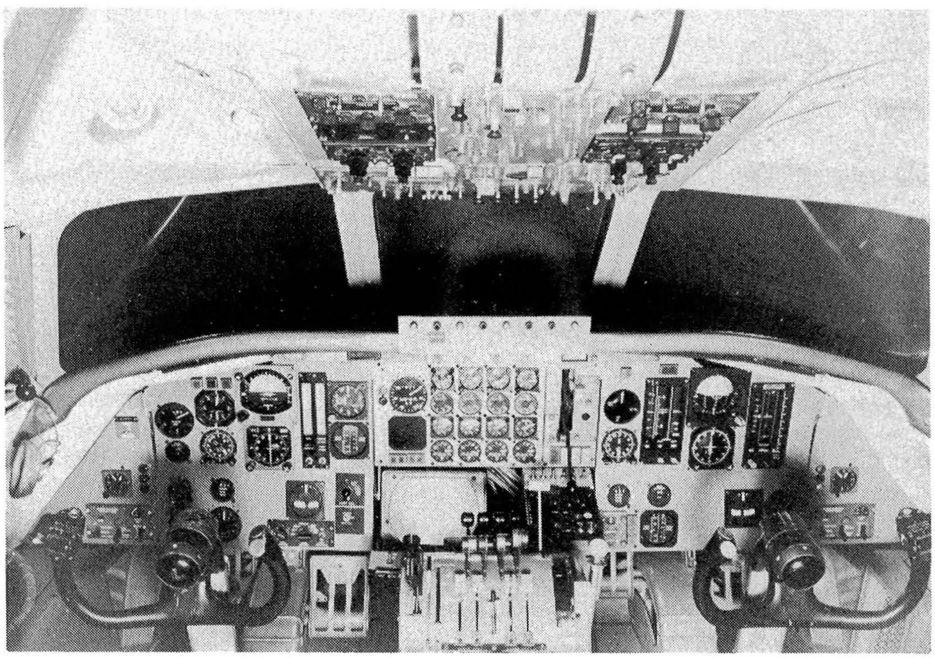
Schematic of AIRLAB

AIRLAB research activities will include (1) the development of methodologies and processes required to fully integrate avionics and control systems for future aerospace vehicles; (2) the study and evaluation of candidate integrated system architectures; and (3) the evaluation of advanced technology in a system context. These areas include research in system theory, fault-tolerant computing, data distribution systems, system design methodologies, and logical processes for intelligent operating systems. Research efforts will result in advanced integrated avionics system concepts, including high reliability, fault tolerance, high dispatch and functional reliability, and improved maintenance procedures; creditable data for use by the airframe industry, including logical, systematic definitions of systems concepts, design feature trade-offs, implementation technology, and design assessments; and experimental systems for proof of concept. AIRLAB will also be used to perform validation research to confirm that safety, performance, reliability, and economic specifications for a particular system are actually met.



DC-9 Simulator

The DC-9 Simulator located in Building 1220 will consist of a fixed-base DC-9 cockpit, signal distribution console, two X-Y plotters, and electronic cabinets. The existing DC-8 cockpit is now being upgraded into a complete DC-9 configuration, which will allow dedicated DC-9 simulations in early 1983. Three stations are available in the cockpit for a Captain, First Officer, and a Flight Engineer. Flight control responses for elevator, aileron, and rudder are simulated by forces from hydraulic servo systems. Throttle controls for four engines are provided on the center console. The simulator has communication receivers for VOR/ILS (VHF Omnidirectional Range/Instrument Landing System) and for ADF (Airborne Direction Finder) navigation. A collimated color visual out-the-window display is provided at the Captain's station. The Adage Company Graphics Display can be made available via closed circuit television. The Area Navigation System is available to provide horizontal and vertical steering signals to simulate an aircraft following a predetermined, three-dimensional flight path. Research applications have included simulation studies of large multi-engine jet aircraft in Air Traffic Control situations, handling qualities, and stability augmentation requirements for supersonic transport configurations.

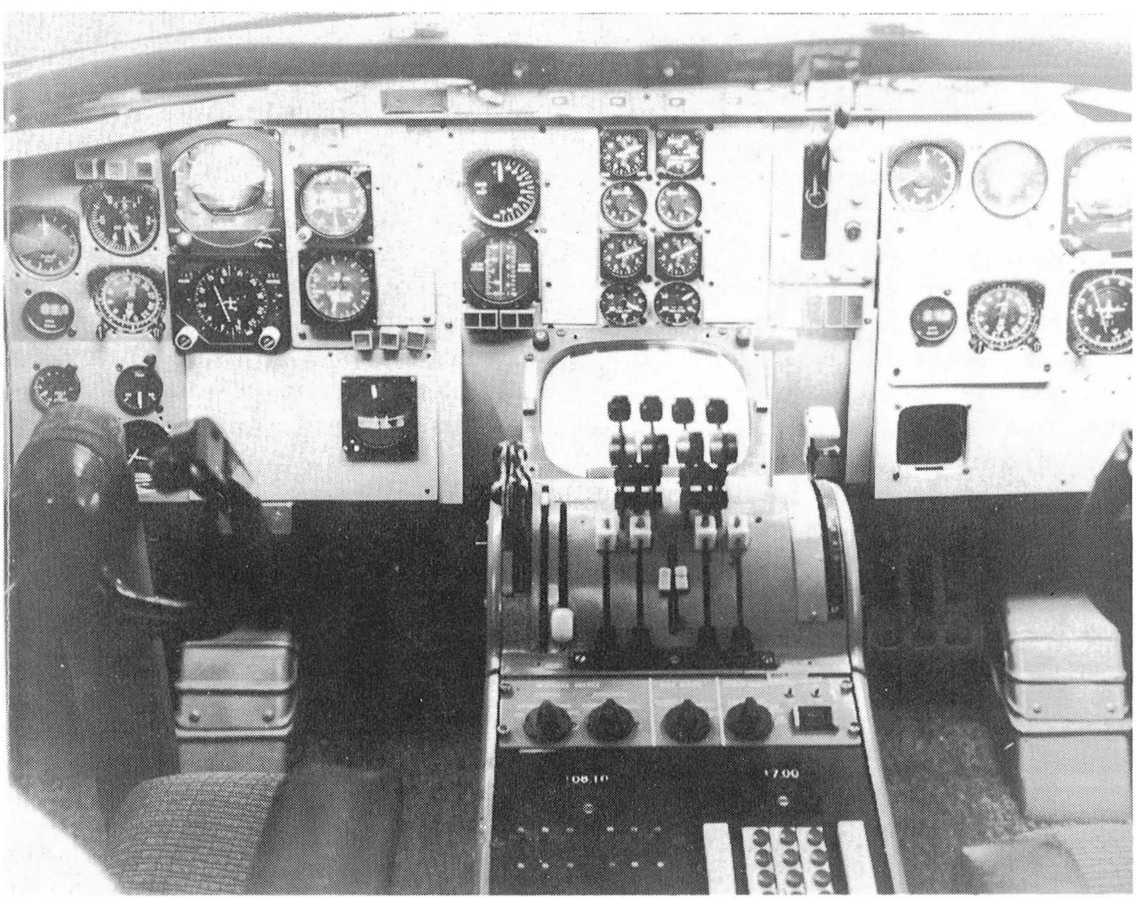


Interior View—DC-9 Simulator

CDTI Traffic Sensor Noise Study

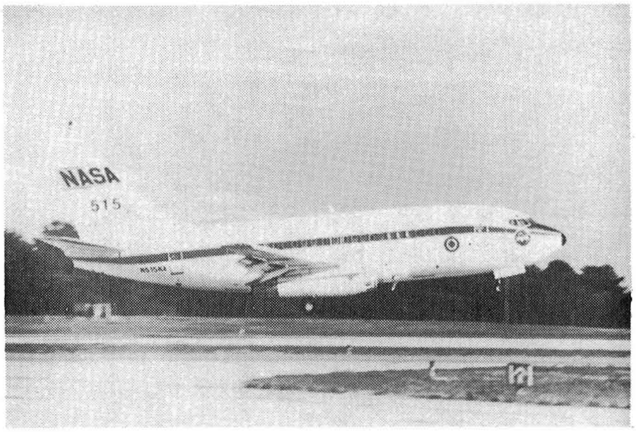
The Cockpit Display of Traffic Information (CDTI) concept provides a pilot with a graphic display of the positions of nearby traffic relative to his own aircraft. The basic information required to drive the display is the relative altitude, range, and azimuthal bearing of the traffic. One method of obtaining this information is through an onboard traffic sensor which utilizes the transponder responses of other aircraft to determine their location. This method is one capability of the proposed Threat Alert and Collision Avoidance System (TCAS) under investigation by the FAA. Random sensor noise present in such a system will result in errors in the displayed range and azimuthal bearing of the target aircraft.

A study to determine the effect of traffic-sensor noise on a pilot's ability to perform a self-spacing task was recently completed in the original DC-8 Transport Simulator. The true positions of the traffic were perturbed in both range and azimuth by random noise and displayed to the pilot on a scope located in the weather radar position. The evaluation task involved simulated instrument approaches into the Denver terminal area while maintaining self-separation on a lead aircraft. Both the separation performance data and pilot subjective ratings and comments were obtained from this study. A preliminary analysis of the separation performance data indicates that pilot performance was acceptable for the self-spacing task even for the highest noise levels tested.



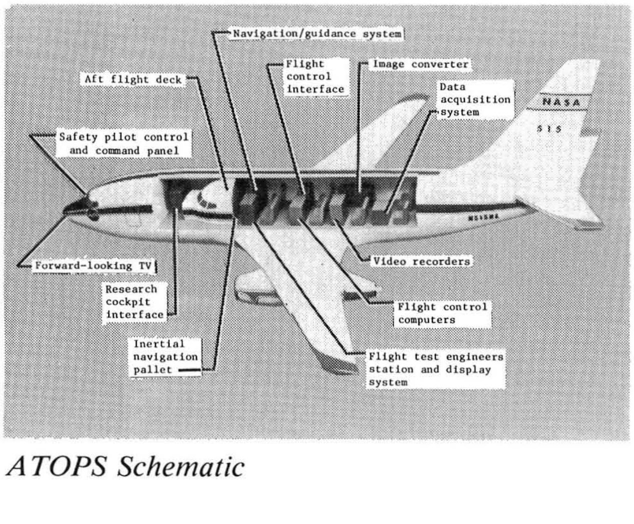
CDTI in Simulator

Advanced Transport Operating Systems (ATOPS) and ATOPS Simulator

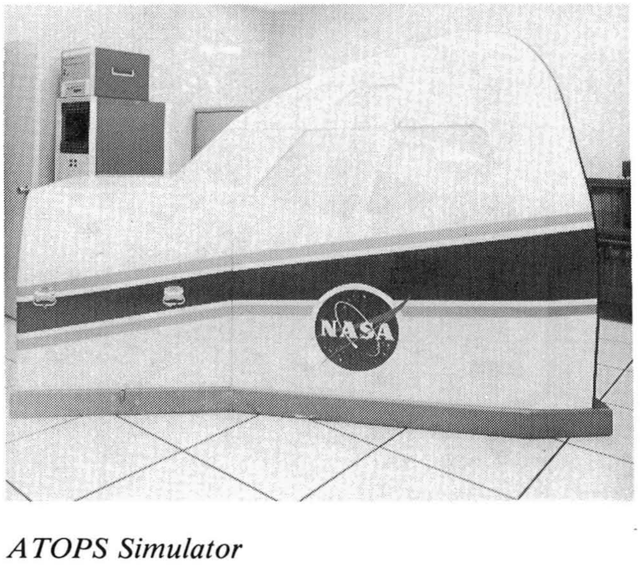


The "Advanced Transport Operating Systems" is a Boeing 737-100 airplane (formerly called the Terminal Configured Vehicle, or TCV) used to conduct flight tests for new concepts of airborne systems and candidate operational flight procedures. The airplane is equipped with a special research flight deck, located about 20 feet aft of the standard flight deck. An extensive array of electronics equipment and data recording systems is installed throughout the former passenger cabin. The airplane can be flown from the aft flight deck in a fly-by-wire mode, using advanced electronic displays and automatic control systems that are all-digital and can be reprogrammed for research purposes. Two safety pilots in the front flight deck are responsible for all phases of flight safety and most traffic clearances. Two research pilots usually fly the airplane from the aft cockpit during test periods. The only airplane systems that cannot be controlled from the aft flight deck are the landing gear and the speed brakes. The safety pilots can take control of the airplane at any time by overpowering the aft flight deck controls or by disengaging the aft flight deck.

The ATOPS Program is concerned with identifying, developing and evaluating displays, controls and crew procedures that will be required for more efficient and acceptable flight operations in high-density terminal areas in an improved air traffic control environment of the future. The ATOPS airplane, with its advanced control systems, displays and data monitoring equipment, is well suited to flight test and exploit the full operating potential of the Microwave Landing System (MLS) now being developed by the FAA to replace the Instrument Landing System (ILS) presently in use at all major airports in the world and at many smaller airports.

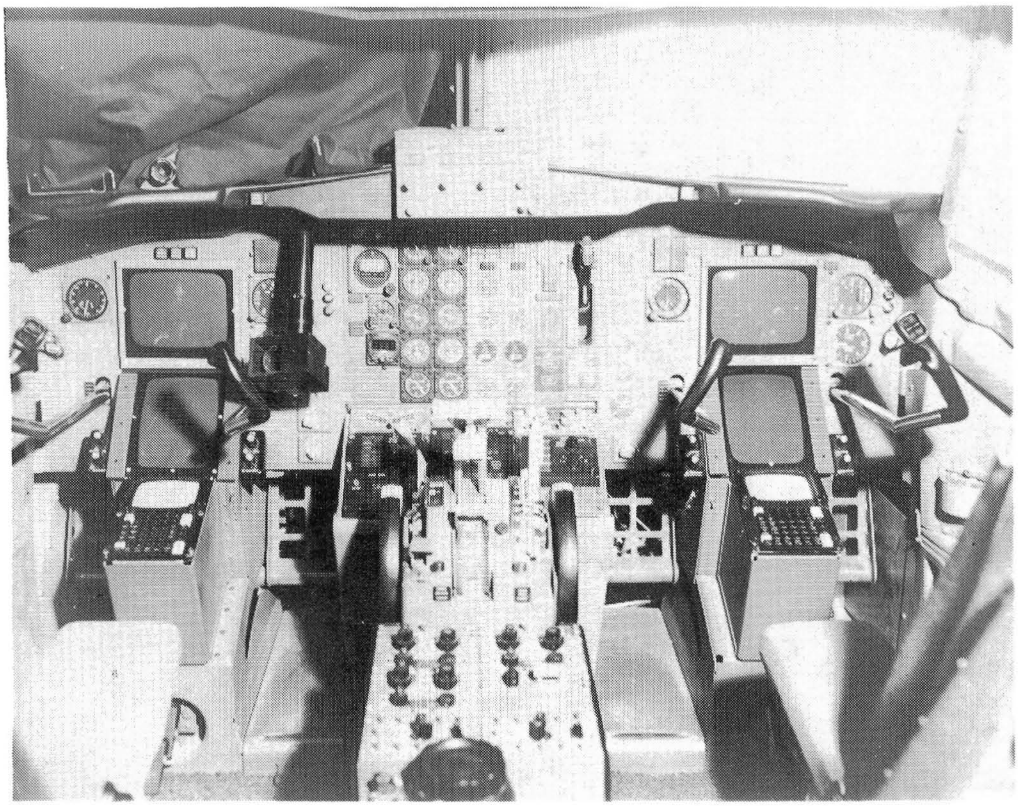


ATOPS Schematic



ATOPS Simulator

The ATOPS Simulator is a duplicate of the aft deck cockpit in the Boeing 737-100 aircraft. The Simulator provides the means of ground-based simulation in support of the ATOPS research program. The cockpit contains duplicate flight instruments, interchangeable cathode ray tubes for the electronic attitude indicator and multiple function indicator, computer driven throttles and flaps, a simulated navigation control display unit and a simulated control mode panel. The Simulator allows proposed control laws or display techniques to be verified and validated prior to testing in the aircraft.



ATOPS Simulator—Interior View

Digital Integrated Auto Land System (DIALS)

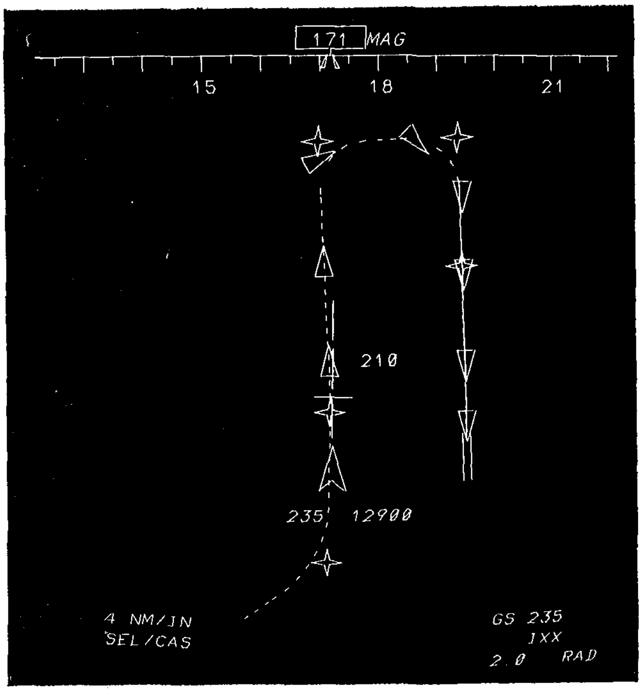
DIALS was designed to explore and demonstrate the improved capabilities made possible by the use of modern control and direct digital methodologies with the next generation terminal area guidance system, the Microwave Landing System (MLS) for improved terminal area operations and safety of transport aircraft. DIALS added the capability to capture and track steep glideslopes (up to 5 degrees), as well as conventional 3-degree glideslopes, provided a flexibility which can increase the efficiency of terminal area operations, improved noise abatement in terminal area communities and provided a means of vortex avoidance when following large aircraft. The ability to "capture the localizer" (runway centerline) with low overshoots in adverse weather conditions increases terminal area efficiency by enhancing the independent operation of close parallel runways. Capturing the localizer and glideslope simultaneously with quick settling time to perform close-in captures (approximately 2 n.mi.) provides new flexibility in aircraft scheduling and sequencing for air traffic control operations.

DIALS was flight tested on the ATOPS research aircraft to determine its performance. Automatic final approaches and landings were successfully tested for conventional and steep glideslopes (3°, 4.5°, and 5°), and for various localizer intercept angles (20° - 50°) under various wind conditions. The average overshoot of the runway centerline was 24.2 ft with a standard deviation of 25.7 ft. The flight tests also included decrab maneuvers in crosswinds up to 12 knots. Simultaneous glideslope and localizer captures were successfully tested for close-in captures. The settling times starting at capture initiation were found to be approximately 40 seconds for the localizer and 10 seconds for the glideslope. The mean touchdown sink rate for ten automatic (hands-off) landings was 2.4 ft/sec with a standard deviation of 0.7 ft/sec.

Self-Spacing With a CDTI

One of the more attractive applications of a Cockpit Display of Traffic Information (CDTI) is self-spacing wherein a pilot can acquire and maintain a specified interval on a lead aircraft. The assumption behind this application is that pilots using CDTI should be able to achieve more consistent spacing performance than the present air traffic control concept, and hence runway throughput could be increased. Along with this benefit comes the question of whether or not dynamic oscillations would occur, similar to the "accordian" effect seen with a queue of automobiles in stop-and-go traffic.

In order to explore this potential problem, an experiment was conducted using the ATOPS simulator in which the navigation display was converted to a CDTI by presenting the position of other aircraft in the area. Queues of up to nine CDTI-equipped aircraft, each one self-spacing on the preceding aircraft, were built by recording successive approaches flown in the simulator and using this recording as the source for traffic data on each subsequent approach. The aircraft crews were rotated to ensure that the pilots had no prior knowledge of the aircraft they would be following. The results of the experiment showed no evidence of any dynamic instability tendencies which had been anticipated. These tests, as well as related experiments, suggest that more consistent spacing performance might be obtained using a CDTI than is being achieved with today's air traffic control concept.



Self-Spacing With a CDTI

General Aviation Simulator

The General Aviation Simulator (GAS) consists of a flight quality general aviation aircraft cockpit mounted on a two-degree-of-freedom motion platform. The cockpit is a reproduction of a twin-engine propellor driven general aviation aircraft with programmable control loading for the wheel and "thru-the-panel" column, and spring loaded rudder pedals as well as a full complement of instruments, controls and switches including radio/navigation equipment. A collimated image visual system provides a 60°-field-of-view out-the-window color display. The visual system can accept inputs from a model board system, computer generated graphics and a target aircraft/horizon scene. Research applications of the GAS include the evaluation of systems for approach/landing displays; stability/control of free wing aircraft; system studies for single pilot operation; and evaluation of techniques for mechanical gust control.

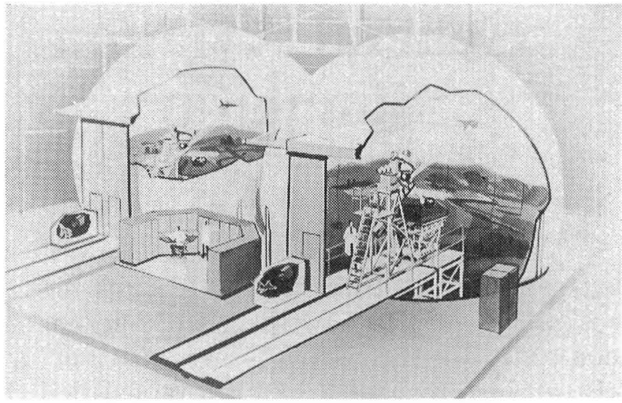
Automatic Terminal Approach System Simulator Study

An effort is under way at LaRC to improve the pilot/machine interface with aircraft automation so as to reduce the workload and increase the safety and utility of single pilot Instrument Flight Rules operations. An automatic terminal approach system (ATAS) was conceived as a means of improving this critical interface. The ATAS can automatically fly a published instrument approach by using stored instrument approach data to automatically tune aircraft avionics and control an aircraft's autopilot. The ATAS will execute the missed-approach procedure at the completion of the approach unless the pilot takes over to land.

A simulation study was performed to determine the feasibility of an ATAS, determine pilot acceptance, and examine pilot interaction with such a system. A generic ATAS system was simulated in the Langley GA Simulator. Seven pilots each flew four instrument approaches with the ATAS and four approaches with a baseline heading select autopilot mode. The ATAS runs resulted in lower flight technical error, lower pilot workload, and fewer blunders than with the baseline autopilot, and the ATAS status display enabled the pilots to maintain situational awareness during the automatic approaches.

ATAS Installed in GA Simulator

Differential Maneuvering Simulator

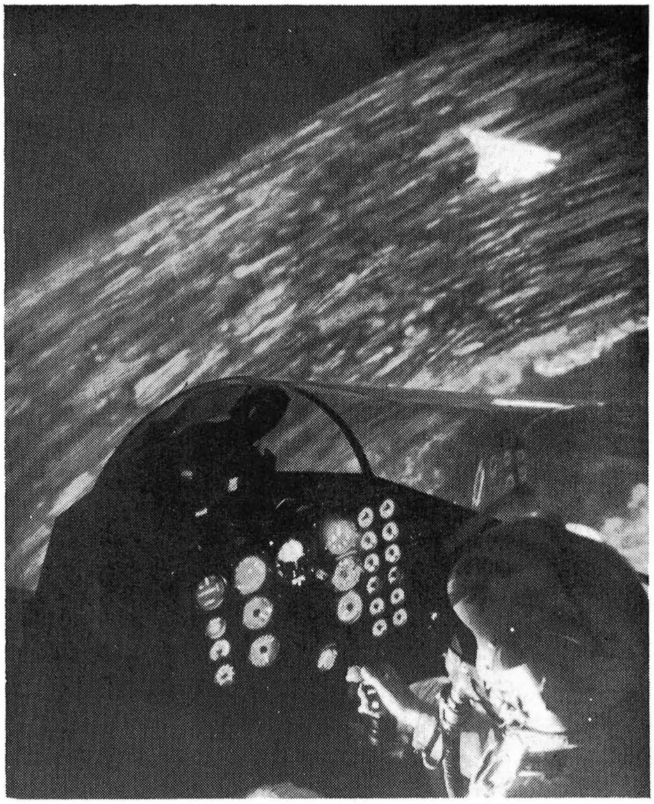


The Langley Differential Maneuvering Simulator (DMS) provides a means of simulating two piloted aircraft operating in a differential mode with a realistic cockpit environment and a wide angle external visual scene for each of the two pilots. The system consists of two identical fixed-base cockpits and projection systems, each based in a 12.2-m-diameter (40 ft) projection sphere. Each projection system consists of a sky-Earth projector to provide a horizon reference and a system for target-image generation and projection. The internal sky-Earth scene provides reference in all three rotational degrees of freedom in a manner which allows unrestricted aircraft motions. The sky-Earth scene has no translational motion. The internal visual scene also provides continuous rotational and bounded (300 ft to 45,000 ft) translational reference to a second (target) vehicle in six degrees of freedom. The target image presented to each pilot represents the aircraft being flown by the other pilot in this dual simulator. Each cockpit provides essential instruments and displays along with a wide angle heads-up display. Kinesthetic cues in the form of a G-suit pressurization system, G-seat system, cockpit buffet and programmable control forces are provided to each pilot consistent with his aircraft's motions. Research applications include high-angle-of-attack spin susceptibility studies, evaluation of evasive maneuvers for various aircraft, and evaluations of the effect of parameter changes on the performance of several baseline aircraft.

Improved High-Angle-of-Attack Control System for F-14

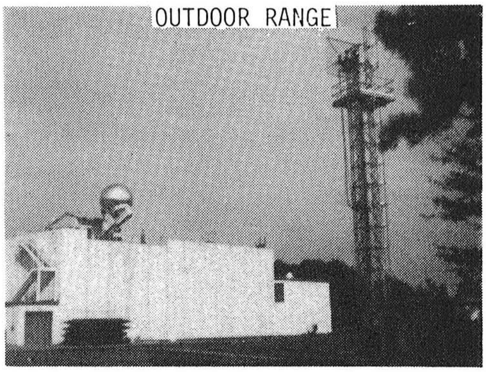
A major thrust of the Military Stall/Spin Program conducted at the NASA Langley Research Center is to investigate the high-angle-of-attack flight characteristics of current and advanced fighter airplane designs using piloted simulations conducted on the Langley Differential Maneuvering Simulator (DMS). These simulations are used to predict the high-angle-of-attack flying qualities, to assess the departure and spin susceptibility, and to determine the automatic flight control laws needed.

During 1981 a multi-year piloted simulation program to develop effective high-angle-of-attack automatic flight control laws for the F-14 airplane was completed. The simulated airplane was flown by research test pilots, contractor test pilots, and Navy test pilots through a wide range of air combat maneuvering to identify the critical flight conditions and to develop control system concepts to provide maximum maneuverability and a high level of departure and spin resistance. The simulation study resulted in the development of high-angle-of-attack automatic control laws for the F-14 airplane which provide automatic control coordination, suppression of wing rock, and automatic spin prevention. This control system developed in the Langley DMS has now been validated in airplane flight tests conducted at the NASA Dryden Flight Research Facility.

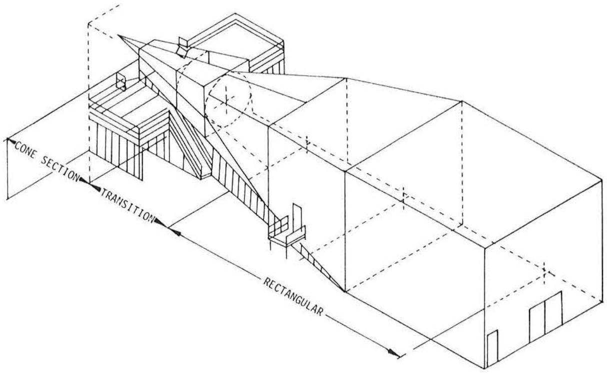


DMS Studies of F-14 Control System

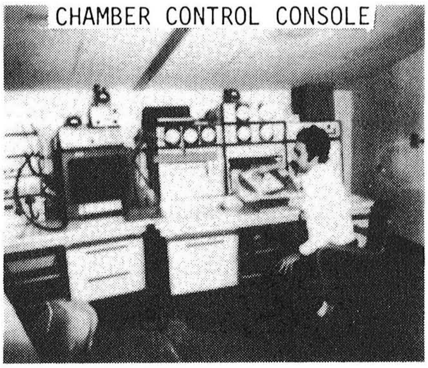
Vehicle Antenna Test Facility



The Vehicle Antenna Test Facility (VATF) is a research facility used to obtain data for new antenna designs and antenna systems and to provide antenna performance data in support of various research programs. The VATF consists of two indoor radio frequency (RF) Anechoic Test Chambers and an outdoor antenna range system. The anechoic chambers which are RF-shielded provide simulated free-space conditions for measurements from 100 MHz to greater than 40 GHz. The anechoic chambers, shaped like pyramidal horns to avoid specular reflections of the walls, are over 100 feet long and have test area cross sections approximately 30 feet by 30 feet.



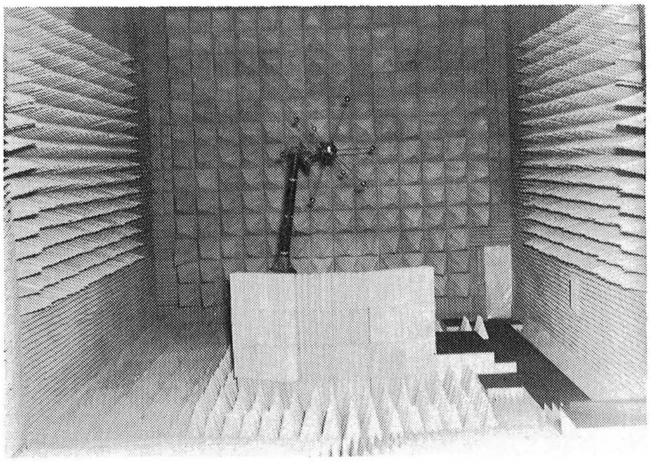
Antennas and aircraft models measuring up to 8 feet can be evaluated in the facility. Complete spherical radiation characteristics of antennas can be measured automatically using digital techniques and magnetic tapes for data storage. The measured data stored on tape can then be processed to provide antenna directivity, polar or rectangular plots of the radiation patterns and false color volumetric plots of the radiation density.



The outdoor antenna range system is available for use when the antenna or test model size or frequency precludes the use of the anechoic chambers. The outdoor range consists of two remote transmitting towers that are spaced 150 feet and 350 feet from the test positioner mounted on the VATF roof. The VATF has several electronic laboratories with extensive measurement capability needed to support the design of unique antennas for aircraft and missiles prior to their evaluation in the antenna chambers or on the outdoor antenna range system.

Maynard Daughter Payload

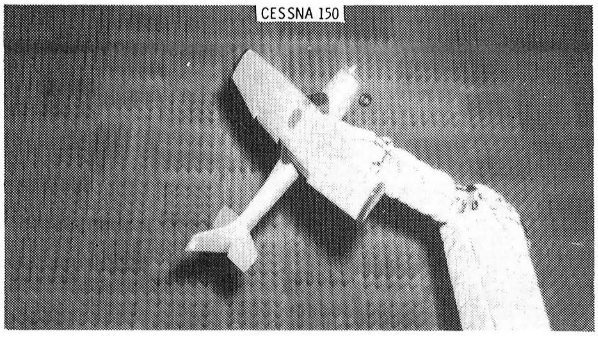
Antenna systems were designed and their performance measured at two frequencies in the VATF for a Wallops Flight Center Project, called the Maynard Daughter payload. The Maynard class of payloads is designed to measure ion densities in the upper atmosphere. The nature of the measuring technique dictated very rigid antenna design criteria and the impact of the antenna systems on the surface configuration of the Maynard Sphere/Arm Assembly had to be minimized. The final designs for the two antenna systems consisted of an array of eight cavity-backed slots and an array of four low-profile monopole antennas. Antenna radiation patterns, gain and impedance measurements obtained in the VATF's Low Frequency Anechoic Test Chamber verified that the antenna systems satisfied the design requirements. Successful launches of the payload/antenna systems from Wallops Island; Norway; and Poker Flats, Alaska, indicate the antenna systems performed as required.

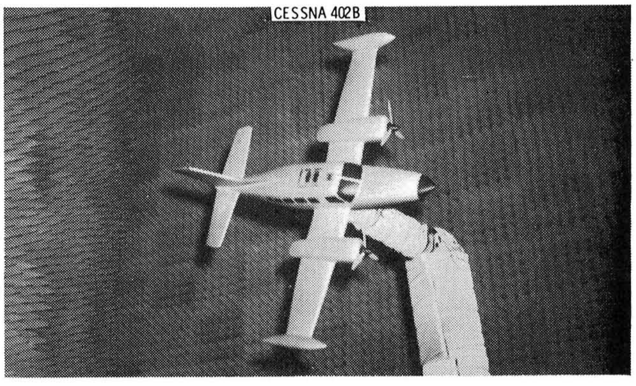


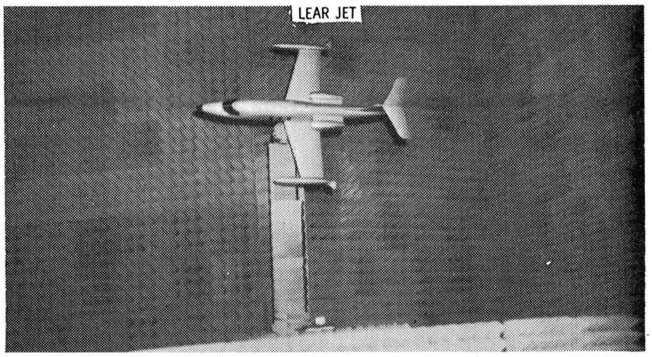
Wallops Payload in VATF

Antenna Technology for General Aviation Aircraft

Antenna tests were performed in the VATF to develop new antenna designs and to optimize antenna performance for two airborne antenna systems on general aviation aircraft. Tests were conducted for the Microwave Landing System (MLS) and the Emergency Locator Transmitter (ELT) System. Several one-seventh scaled models of different general aviation aircraft such as Lear Jets and Cessnas were used to conduct the tests. The measurements provided data for optimizing the MLS antenna locations and performance on Ransom Airline's de Havilland DASH 7 aircraft,





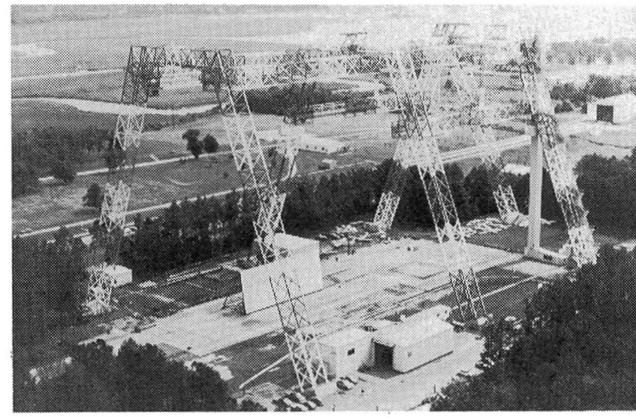


Antenna Systems Tests for GA Aircraft

the first commuter airline to utilize the new MLS. The MLS, scheduled to replace the existing outdated Instrument Landing System (ILS) currently in use at commercial airports, operates at a frequency of approximately 5 GHz and the measurements were performed at 35 GHz with the scale models to provide equivalent full-scale data. The full-scale frequency is equal to the scale model frequency multiplied by the scale factor.

Data were also obtained for optimizing the locations of ELT antennas on several scale model general aviation aircraft. The ELT system is used by downed aircraft to transmit signals to search aircraft or to the Search and Rescue (SAR) Satellite to aid in the rescue operations. Antenna performance was measured on the scale models at 850.5, 1701.0 and 2845.5 MHz representing full scale frequencies of 121.5, 243, and 406.5 MHz, respectively. A dual frequency ELT antenna designed at NASA LaRC to operate at 121.5 and 406.5 MHz was scaled and evaluated on one of the test models. Flight antennas are being supplied by NASA Langley to NASA Goddard Space Flight Center for mounting on general aviation aircraft and flight evaluation with the SAR Satellite.

Impact Dynamics Research Facility

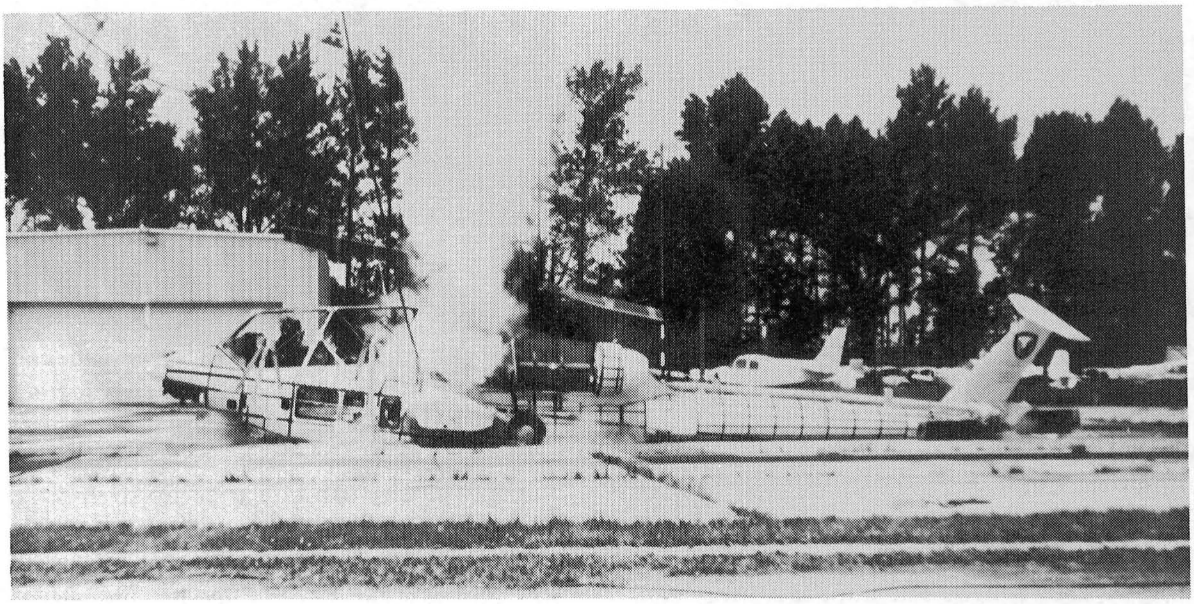


This facility, originally used for simulating lunar landings by the astronauts during the Apollo Program, has been modified to simulate crashes of full-scale aircraft under controlled conditions. Simulation is accomplished by swinging the aircraft by cables, pendulum-style, into the concrete impact runway from an A-frame structure approximately 400 ft long × 230 ft high. The impact runway can be modified to simulate other crash ground environments such as packed dirt with trees. The impact runway has been modified in the past by using soil to meet a specific test requirement. The aircraft is suspended by swing cables from 2 pivot points 217 ft off the ground. It is then pulled back along an arc to a predetermined height by a pullback cable from a movable bridge on top of the A-frame, released from the pullback cable, and allowed to swing, pendulum-style, into the ground. An instant before impact, the swing cables are separated from the aircraft by pyrotechnics. The length of the swing cables regulates the aircraft impact angle from 0° (level) to approximately 60°. Impact velocity can be varied up to approximately 65 mph, governed by the pullback height, and to 90 mph with rocket assist. Variations of aircraft pitch, roll, and yaw can be obtained by changes in the aircraft suspension harness attached to the swing cables. Onboard instrumentation data are obtained through an umbilical cable attached to the top of the A-frame. Data are transmitted by hard wire to the control room at the base of the A-frame. Photographic data are obtained by ground cameras and cameras mounted on top of the A-frame. Maximum allowable weight of the aircraft is 30,000 lb with limitations that might be imposed by other dynamic parameters.

YAH-63 Attack Helicopter Crash Test

A YAH-63 attack helicopter was crash-tested at the Impact Dynamics Research Facility as a joint effort of NASA and the Army Aviation Research and Development Command Applied Technology Laboratory at Fort Eustis, Virginia. The objective of the test was to acquire engineering data relative to the dynamic behavior of the structure and such major components as crashworthy fuel tanks, landing gear struts and crew seats in an actual crash environment. In addition to the structural aspects of the test, additional experiments on board included an inflatable body and head restraint system, an accident information retrieval system, a deployable Navy flight incident recorder and crash position locator, and an emergency locator transmitter.

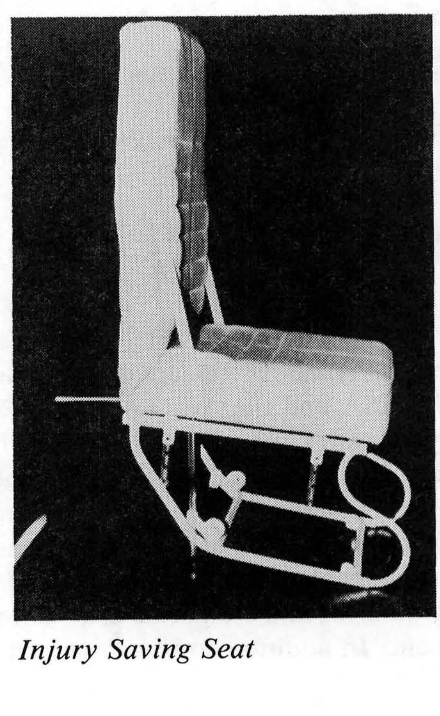
For the test, the aircraft struck the concrete at a nominal 50 feet per second along a −55° flight path at a pitch angle of +10°. The white "smoke" which dominates the scene is vaporized hydraulic fluid from the landing gear struts. The struts were designed for crashworthiness by equipping each of them with load-limiting "blow off" valves such that crash energy is absorbed by venting the contained hydraulic fluid through a restrictive orifice. The majority of the experimental crashworthy systems functioned as designed, including the AH-64 production crew seat, which stroked fully, and the Navy's deployable recorder/locator signal which was received by an observer aircraft.



Army YAH-63 Crash Test

Injury Prevention With Modified Seat

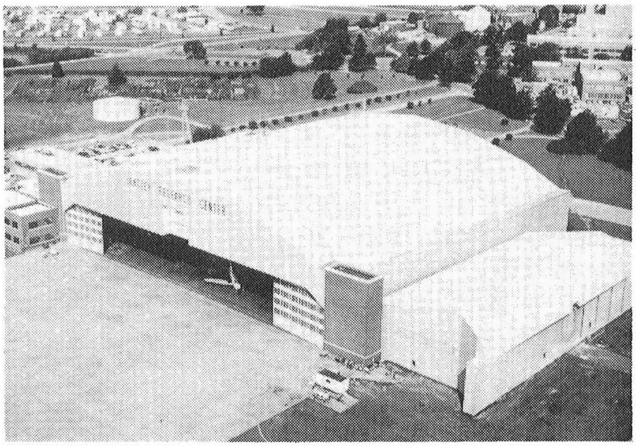
Several of the large missionary organizations such as JAARS (Jungle Aviation And Radio Service) have a strong interest in reducing injuries from light aircraft accidents. A cooperative effort has begun between NASA and JAARS toward crash protection utilizing existing technologies and the Impact Dynamics Research Facility at Langley.



Injury Saving Seat

The NASA Langley recommendations given to JAARS for seat and restraint modifications were metal seat pans, double shoulder harness, lap belt with metal-to-metal hardware, and a foam seat material orginally developed for astronauts' couches. In February of 1981, a Cessna 206 modified by JAARS with the above seat and harness changes crashed in Papua, New Guinea. The aircraft stalled and impacted with a high vertical sink rate which caused extensive airframe damage and significant deformation of both the floor and seat pan areas. Following the accident, the pilot was in excellent condition with no cuts, abrasions, sprains, or broken bones, and the right seat pilot suffered only stiffness of the neck for several days. Four accidents, under similar circumstances but with unmodified seats, resulted in broken backs and other injuries, with one fatality.

JAARS designed an 'S' frame seat for attenuation protection. A recent LaRC crash test of a light twin airplane, impacting at 60 miles per hour, had one of JAARS' 'S' frame seats with temper foam cushions installed in the pilot's section of this aircraft. Data showed the anthropomorphic dummy on the 'S' frame seat received only 20 G's while the anthropomorphic dummy in the standard co-pilot seat received 30 G's at a sink rate of 23 feet per second.



Flight Research Facility

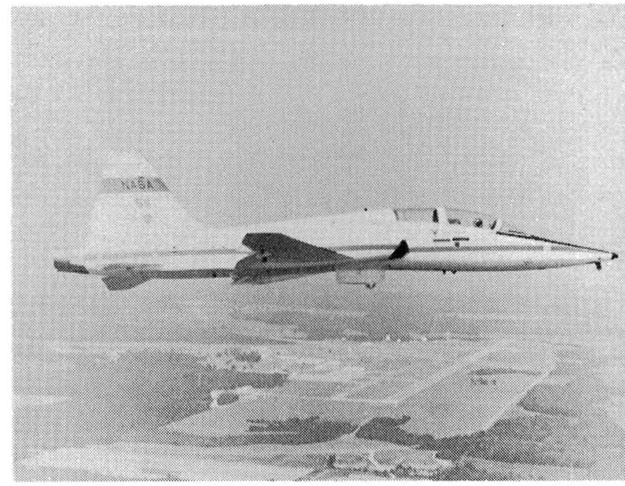
Twice the current number of twenty aircraft could easily be housed in the huge hangar of the Flight Research Facility. Door dimensions will allow entry of a Boeing 747. Such features as floor air and electrical power services; radiant floor heating to eliminate corrosion-causing moisture; a modern deluge fire suppression system; energy saving lighting; modern maintenance spaces; and entry doors and taxiways on either side of the building make this structure equal or superior to any hangar in the country. Extensive and modern maintenance equipment makes possible repairs on aircraft ranging in sophistication from modern metal and composite airliners, fighters, and helicopters to fabric-covered light airplanes. Surrounding the hangar are ramp areas with a load bearing capacity sufficient to handle the largest wide body jet now flying. The high power turnup area can also handle a wide variety of aircraft.

The present array of research and research support aircraft includes an airliner, military fighters, trainers, a bomber, experimental one-of-a-kind designs, an agricultural airplane, a corporate jet, helicopters and single and multiengine light airplanes. This variety enables research to be carried out over a wide range of flight conditions, from hover to Mach 2 and from the surface to 60,000 feet. Research pilot currency in this wide spectrum of aircraft is important in doing credible inflight experiments as well as flight simulator assessments. A variety of research is conducted in such areas as terminal traffic flow, Microwave Landing System (MLS) approach optimization, airfoil properties, single-pilot IFR, engine noise, turbulence research, natural laminar flow, winglet studies, stall/spin, and severe storm hazards.

As an example, the severe storm hazards research is focused on conducting flight tests with the F-106B to obtain data defining severe storm characteristics with regard to lightning and turbulence. Daytime thunderstorm penetration flights were made in the thunderstorm season of 1981 with the aid of both ground-based and airborne radar near the National Severe Storms Laboratory in Oklahoma and also near Langley and Wallops Flight Center. The F-106B, which was experimentally configured, flew between 15,000 and 30,000 feet altitude, conducting 111 storm cell penetrations and receiving 10 lightning strikes. Data describing the wave forms of the lightning strike current to the aircraft and the aircraft electromagnetic response to the strike were collected. With future aircraft using more digital electronic systems for control and guidance, as well as composite structural materials, the need for understanding the physics of aircraft-lightning interactions has become more important. In addition to the lightning hazard characterization, wind and turbulence data are also being measured by the aircraft as it penetrates storm cells. These data will be compared with wind and turbulence measurements by advanced ground-based weather radars using the Doppler principle to develop criteria for vectoring aircraft through and around severe storms when these new radars become part of the future National Aerospace System.

ADMINISTRATIVE AIRCRAFT

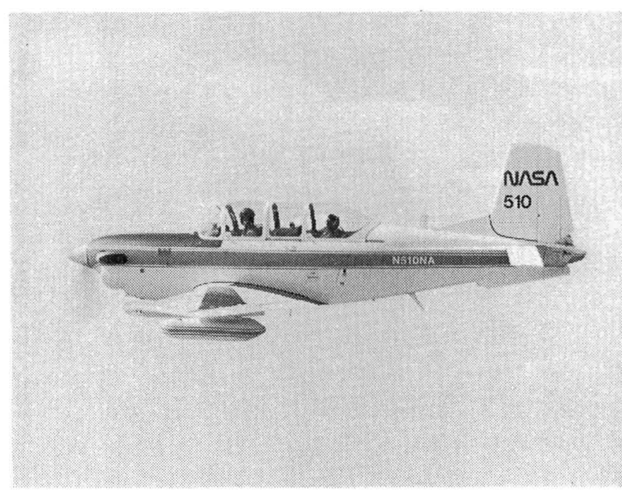
NASA 1, GRUMMAN GULFSTREAM G-159



NASA 511, NORTHROP T-38A, TALON

RESEARCH SUPPORT AIRCRAFT

NASA 505, CESSNA U3A

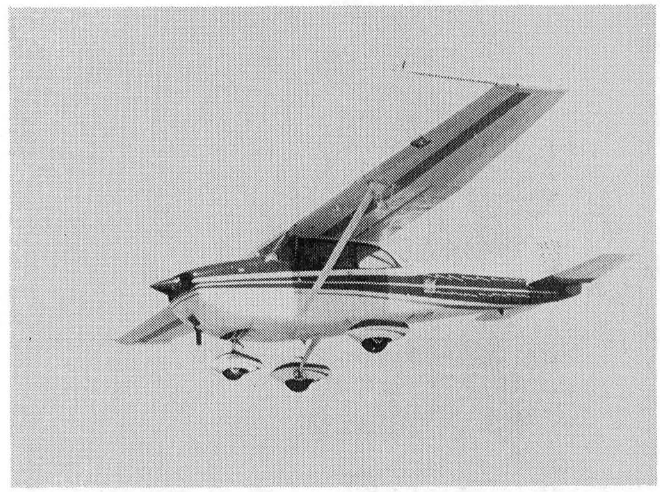


NASA 510, BEECH T-34C, MENTOR

NASA 506, BEECH B-80, QUEEN AIR

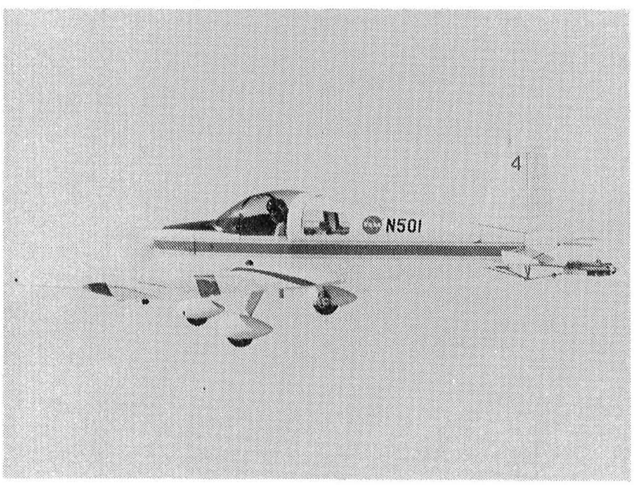
NASA 530, BELL 204B

NASA 540, BELL OH-58A, KIOWA



NASA 507, CESSNA C-172K, SKY HAWK

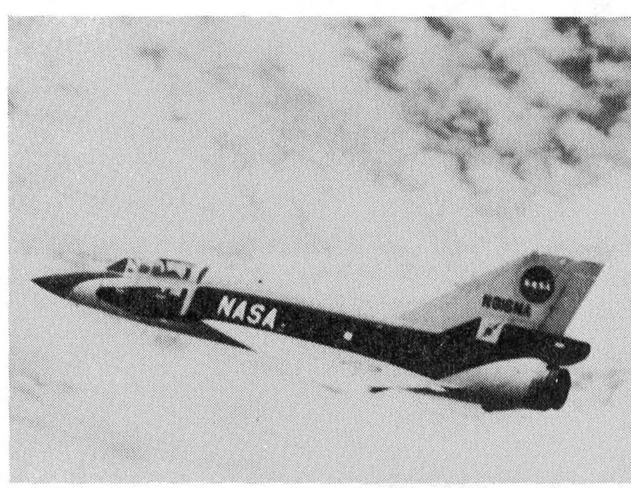
RESEARCH AIRCRAFT



NASA 501, MODIFIED AA-1, YANKEE

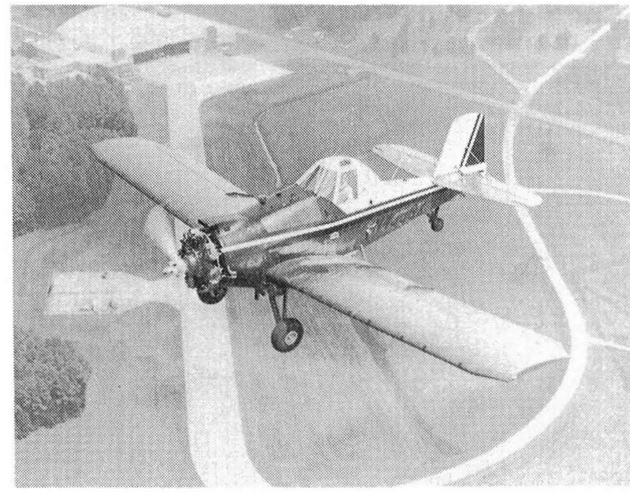
NASA 519, MODIFIED PIPER PA-28R-200, ARROW

NASA 504, MODIFIED BEECH C-23, SUNDOWNER



NASA 816, GENERAL DYNAMICS F-106B

NASA 515, BOEING 737-100



NASA 517, AYRES S2R, THRUSH

NASA 508, DEHAVILLAND DHC-6, TWIN OTTER

NASA 512, GRUMMAN OV-1B, MOHAWK

Some Langley Research and Support Aircraft

NASA 509, NORTH AMERICAN T-39A, SABRELINER

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15. Supplementary Notes

16. Abstract

The role of the Langley Research Center is to engage in the basic and applied research necessary for the advancement of aeronautics and space flight, to generate new and advanced concepts for the accomplishment of related national goals, and to provide research advice, technological support, and assistance to other NASA installations, other government agencies, and industry. This report highlights some of the significant tests which were performed during calendar year 1981 in Langley test facilities, a number of which are unique in the world. The report illustrates both the broad range of the research and technology activities at the Langley Research Center and the contributions of this work toward maintaining United States leadership in aeronautics and space research. Other highlights of Langley research and technology for 1981 are described in "Research and Technology—the 1981 Annual Report of the Langley Research Center." Further information about both reports is available from the Office of the Chief Scientist, Mail Stop 103, Langley Research Center, Hampton, Virginia 23665 (804-827-3316).

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