NASA/TM-2007-214622

2006 Engineering Annual Report

Albion Bowers, Patrick Stoliker, and Everlyn Cruciani
NASA Dryden Flight Research Center
Edwards, California

August 2007

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HIGH-TEMPERATURE SENSOR APPLICATIONS FOR GROUND-TESTING OF C-17 ENGINE

Summary

The Flight Loads Laboratory (FLL) at the NASA Dryden Flight Research Center (DFRC) is attempting to acquire high-temperature dynamic strain measurements by using optical-strain sensors as a secondary experiment on the C-17 (The Boeing Company, Chicago, Illinois) engine "17th Stage Bleed-Air Duct Redesign Verification" tests. The Extrinsic Fabry-Perot Interferometer (EFPI) optical sensors have been used successfully during ground tests to measure static strains to temperatures as high as 1850° F, but no measurement attempt has been made in a combined high-temperature/vibration environment. In addition to optical strain measurements, valuable application experience and data will be generated from thermal-sprayed, free-filament, wire resistive strain gages in this hostile environment.

Objective

This task, performed under a NASA and Air Force Flight Test Center alliance agreement, allows DFRC to acquire valuable experience in applying wire resistive strain gages and also investigates the feasibility of using fiber optic strain measurements under very harsh conditions. In 1997 under the X-33 (Lockheed Martin Corporation, Bethesda, Maryland) program, the FLL installed dynamic high-temperature wire resistive strain gages for measuring combined thermal-acoustic loads on Inconel honeycomb thermal protection system (TPS) panels at the NASA Glenn Research Center in Cleveland, Ohio. The sensors provided good data with minimal failures to temperatures of 1550 °F at acoustic levels to 159 db. Unfortunately, this unique FLL measurement capability has not been further characterized since these initial tests.

Polyimide-coated optical Fiber-Bragg gratings have been used under combined acoustic loads and temperatures to 500 °F, but it is unclear whether the construction of the optical EFPI sensor will endure under similar conditions to even higher temperatures. The focus of the fiber optic research experiment on the C-17 engine during ground testing is to primarily look at sensor survivability issues such as fracturing of the gold-coated fiber, sensor head construction, and the attachment to the substrate. If successful, these tests could lead to an increased effort to develop flight-hardened fiber-optic systems, a technology area currently lacking in research. The commercial EFPI ground system used for these tests can sample at 1 KHz, but the manufacturer is currently working to increase the sample rate to 100 KHz. If the sensor survives and the sampling rate is eventually increased, valid high-temperature measurements can be achieved using a single optical sensor to obtain both dynamic and static strains.

Approach

Thermal-sprayed wire resistive strain gages, seen in figure 1, were attached to the 17th stage case wall near the right and left boss flanges and also on the 10th stage manifold per test requirements dictated by The Boeing Company and Pratt & Whitney (East Hartford, Connecticut). The U.S. Air Force-required tests will be conducted by Boeing in an attempt to isolate a possible unbalance of blades during bird strikes and the resulting vibration fatigue effects on 17th stage bleed-air duct. The instrumented areas are expected to exceed 1000° F during several engine runs.

Axial and circumferential gages were attached using thermal spray procedures described in NASA document DEI-R-011 (unpublished, internal report). Both plasma spray and Rokide flame-spray processes were used to roughen the surface, electrically insulate, and encapsulate the strain gages. Similar methods were used to attach the optical EFPI strain sensors. The optical sensors, seen in figure 2, are not required for the C-17 engine tests, but were integrated to provide research data on the sensor survivability.

Since two different types of high-temperature wire resistive gages are currently preferred for both a dynamic and static (temperature-compensated) measurement, the use of a single EFPI to accomplish the same measurement would be of great benefit. These tests will give NASA insight into the survivability of optical-strain sensors in harsh environments and the installation procedural changes required to improve this technology.

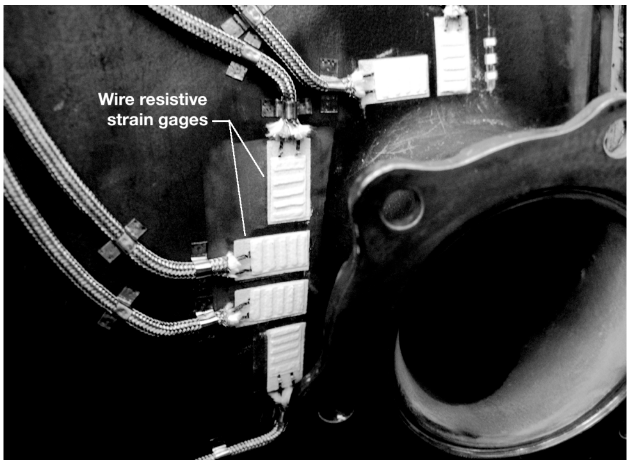

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Figure 1. Left hand 17th stage bleed air boss instrumented with six strain gages and one optical EFPI.

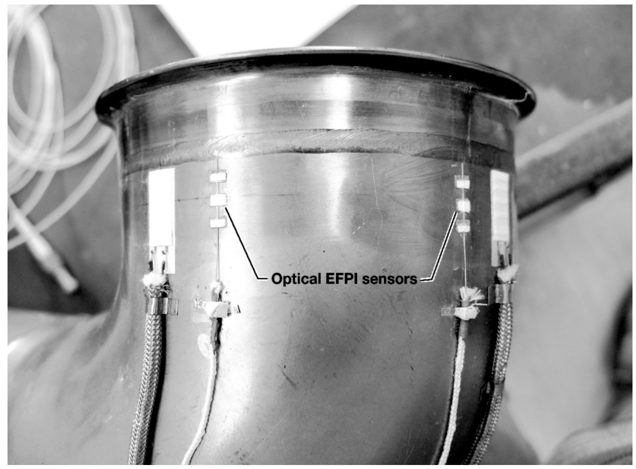

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Figure 2. Example of 10th stage duct instrumentation illustrating the optical EFPI strain sensor.

Status

Currently, developed attachment techniques for the EFPI sensor have been investigated only in high-temperature environments. Laboratory evaluations of the attachment integrity and sensor performance were done in static environments to 1850 °F on both Inconel and ceramic composite substrates. The attempt to measure under high-temperature dynamic conditions using the EFPI sensor is high-risk at this time since no coupon level work in this area has been initiated. Unfortunately, test opportunities that provide the needed test conditions, such as the upcoming C-17 engine ground tests, are rare.

Plans have been initiated to examine this sensor in similar thermal/dynamic laboratory conditions, but have not been implemented as of this time. Failure of the EFPI sensor during the C-17 engine ground tests does not exclude its eventual use for this type of application. It will, however, provide data that will guide future laboratory testing required to improve its ruggedness.

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PROJECT ORION – ABORT FLIGHT TEST

Summary

The Project Orion-Abort Flight Test (AFT) project involves the NASA Johnson Space Center (JSC) (Houston, Texas), the Dryden Flight Research Center (DFRC) (Edwards, California), the Langley Research Center (LaRC) (Langley, Virginia), the Kennedy Space Center (KSC) (Kennedy Space Center, Florida), the Glenn Research Center (GRC) (Cleveland, Ohio), the Project Orion prime contractor, Lockheed Martin Corporation (Bethesda, Maryland), and subcontractors. This project will flight-test the new Orion systems that are responsible for safely removing astronauts from potentially fatal launch and boost phase situations. Proposed tests include pad abort, seen in figure 1, as well as booster-launched flight tests. The AFT flight tests are important to the design effort and timely launch of manned Orion. The goals of the AFT project are to:

1. Provide characterization of emerging design of the critical Launch Abort System (LAS), as well as recovery, power, and avionics systems to improve and perfect the Orion final design.
2. Flight-demonstrate and characterize the performance of the same systems.

Objective

Abort Flight Test objectives will include allowing the Abort Flight Test Project to conduct flight tests, gather data, and disseminate data. This will demonstrate satisfactory performance and operation of the launch abort system, as well as the crew module landing and recovery system, seen in figure 2.

Approach

In methods similar to the Apollo unmanned flight tests, the AFT project plans to flight-test with the latest design of systems available. The pad abort configuration will be flight-tested first. Subsequently, in each ascent abort, a booster will launch the flight-test article to predetermined flight conditions. The Orion systems are expected to change and improve, perhaps as often as each flight. Abort launch will be initiated manually or by predetermined flight conditions, but otherwise the abort events are expected to follow sequenced commands. Results will be provided for design improvement purposes to responsible entities.

Ride-along experiments (for example, TPS, antennas, landing systems, etc.) will be included only where sanctioned by the AFT project and only if it does not jeopardize AFT mission objectives.

Status

On August 14, 2006, JSC successfully conducted tests on one proposed parachute design.

In September 2006, AFT personnel decided to use Lockheed Martin avionics and power systems rather than separate NASA-developed systems to help accelerate Orion system development. In that same meeting, NASA decided to establish a simulation lab at DFRC.

Contacts

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REENTRY THERMAL ANALYSIS OF A GENERIC CREW EXPLORATION VEHICLE WINDWARD WALL STRUCTURES

Summary

Reentry heat transfer analysis was performed on the windward wall structures of a generic composite Crew Exploration Vehicle (CEV) capsule, seen in figure 1, protected by the Apollo thermal protection system (TPS) (ablative material). The Apollo low Earth orbit reentry trajectory was used to calculate the input reentry heating rates. In the thermal analysis computer program used, the TPS ablation effect was not included; however, the results from the non-ablation heat transfer analyses were used to develop an approximate virtual ablation method to estimate the ablation heat loads and the TPS secession thicknesses. Depending on the severity of the heating-rate time history, the virtual ablation period was found to last for 93–117 s, and the ablation heat load was estimated to be in the range of 85–87 percent of the total heat load for the ablation period. The TPS recession thickness was estimated to be in the range of 0.08–0.11 in. The TPS thickness range of 0.717–0.733 in was found to be adequate to keep the composite sub-structural temperatures at the limit of 300 °F. Plots of maximum composite temperatures (at touchdown) as functions of TPS thickness h can be seen in figure 2.

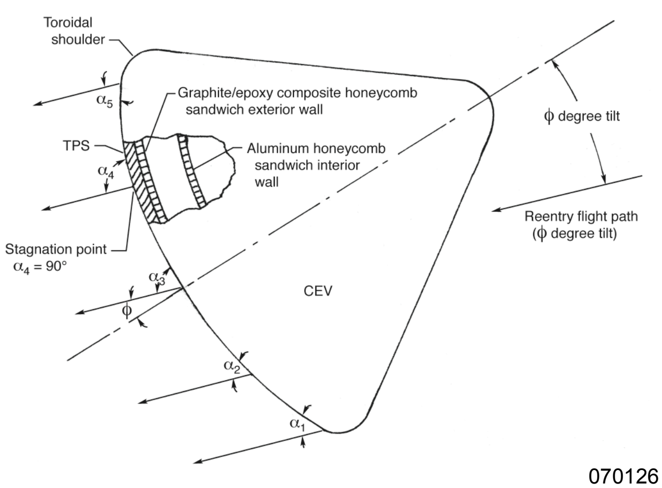

Figure 1. CEV reentry flight at zero-degree angle of tilt.

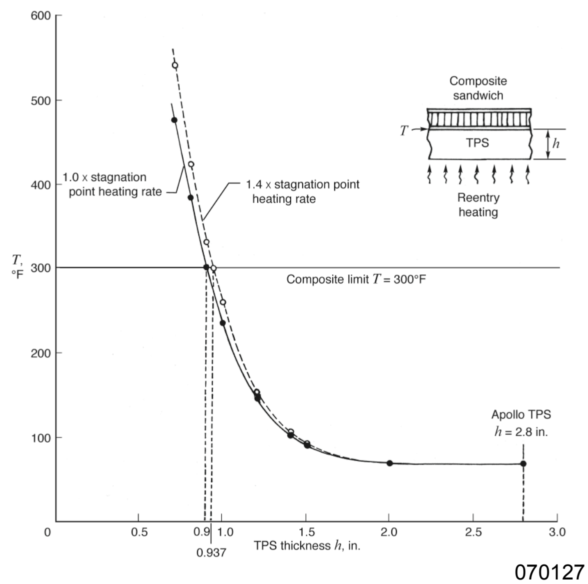

Figure 2. Plots of maximum composite temperatures (at touchdown) as functions of TPS thickness *h*; 1.4 x stagnation point heating rate.

Objective

The objective of this analysis was:

- To investigate the heat-shielding performance of ablative TPS during the low Earth orbit reentry flight.
- To develop virtual ablation method to estimate the ablation heat loads and the TPS secession depths.
- To find the minimum TPS thicknesses to ensure the CEV composite structural temperatures did not exceed the limit of 300 ºF.

Approach

The Apollo low Earth orbit reentry trajectory was used to calculate the reentry heating rates. The NASA Dryden Flight Research Center in-house aerodynamic heating program (TPATH program) was used to calculate the zero-tilt and 18º-tilt stagnation point heating rates. The structural performance and resizing (SPAR) finite-element computer code was then used to calculate the transient temperature distributions in the CEV windward wall structures.

Status

A virtual ablation method (graphical method) was successfully developed for the estimations of the ablation heat loads and the TPS recession thicknesses for CEV low Earth orbit reenty heating. Other more severe lunar-return reentry heating rates will be calculated and input to the SPAR finite-element thermal model to examine the TPS recession thicknesses, minimum required TPS thickness, and substructural thermal response to the change of the lunar-return reentry heating rates.

Contacts

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FURTHER DEVELOPMENT OF KO DISPLACEMENT THEORIES FOR DEFORMED SHAPE PREDICTIONS OF COMPLEX STRUCTURES

Summary

Formulated Ko displacement theories are further developed for nonuniform cantilever beams, seen in figure 1, under bending, torsion, and combined bending and torsion loading. The displacement equations are expressed in terms of strains measured at multiple equally spaced strain-sensing stations on the lower (or upper) surface of the beam. The bending and distortion strain data can be input to the displacement equations for the calculations of slopes, deflections, and cross-sectional twist angles of the beam for generation of the deformed shapes of the entire beam. The displacement equations developed were successfully validated for their accuracy by the finite-element analysis. The displacement theories developed could also be applied to calculate the deformed shapes of simple beams, plates, aircraft wings, and fuselages. The displacement equations with the associated strain sensing system using fiber-optic sensors form a powerful tool for in-flight deformed shape monitoring of the flexible aircraft wings. The calculated displacement data could ultimately be visually displayed before the eyes of a ground-based pilot to monitor the in-flight deformed shape of unmanned aircraft wings.

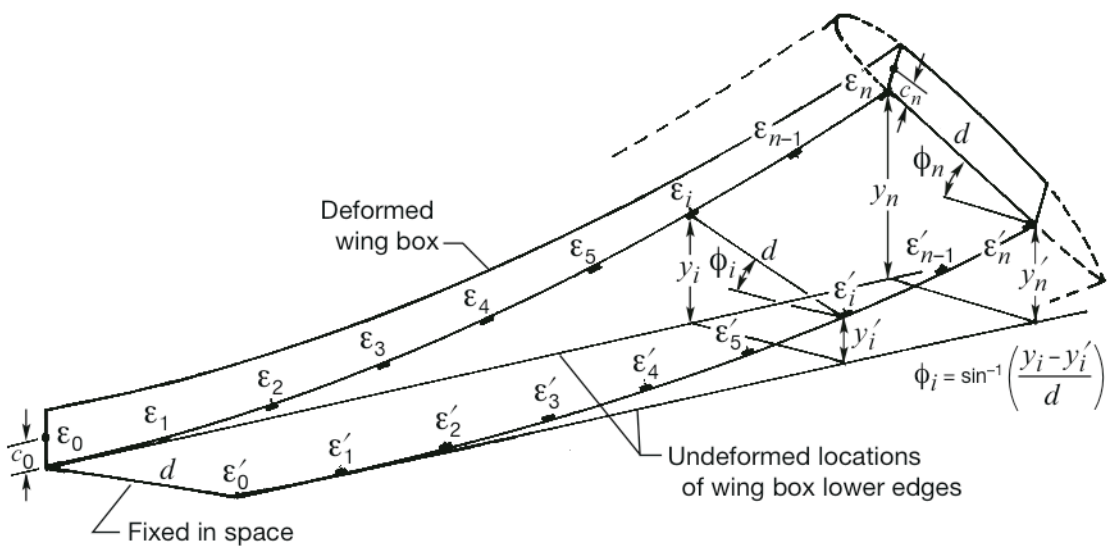

Figure 1. Tapered cantilever wing box.

Objective

By installing multiple strain sensors at discrete sensing stations on a cantilever wing, it is possible to use those strain sensor data as inputs to Ko displacement equations to calculate the deflections and cross sectional twists of the aircraft wings during flight. The purpose is to predict the in-flight deformed shapes of flying vehicles.

Approach

The formulation of the Ko displacement theory for the nonuniform beams is based upon the modified beam differential equation. Using a piece-wise linear assumption and dividing the beam domain into *n* sections, the beam slope and deflection equations for each beam section were then formulated in terms of measured strains at *n* + 1 strain sensing stations at the bottom (or top) of the beam

Status

The Ko displacement theory for nonuniform beams is being tested with the aid of structural performance and resizing (SPAR) finite-element computer program. Cases tested were 1) tapered tubular cantilever beams, 2) un-swept and swept tapered wing boxes, 3) trapezoidal plates, 4) unmanned aerial vehicle (UAV) wing. Partial results, seen in figure 2, show high accuracy of Ko displacement theory in the structural deformed shape predictions. The Ko displacement theory and the associated fiber optics strain sensing system form a powerful tool for monitoring the in-flight deformed shapes of the aircraft wings. This innovative method: Real-time Structural Shape Sensing Techniques using Surface Strain Sensor Measurements, is currently patent pending.

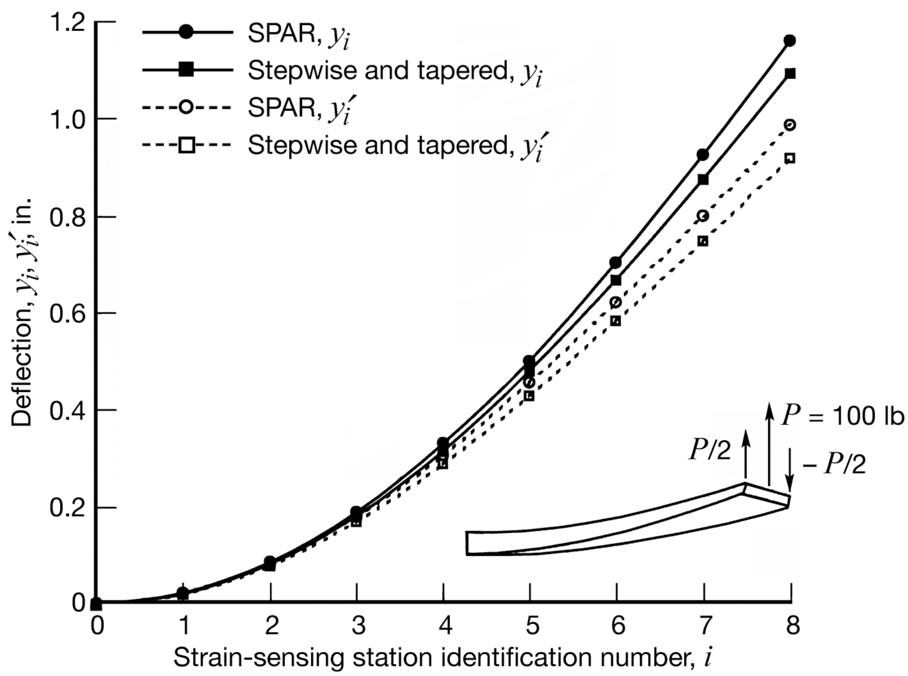

Figure 2. Wing-box deflections predicted from Ko displacement theory and calculated from SPAR finite-element code.

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MASS PROPERTY VERIFICATION OF THE UNMANNED AERIAL VEHICLE SYNTHETIC APERTURE RADAR

Summary

The NASA Unmanned Aerial Vehicle Synthetic Aperture Radar (UAVSAR) program has developed an external store flight-test article for repeat pass interferometry missions aboard an unmanned aerial vehicle (UAV). Most of the UAVSAR pod flight instrumentation, structure, and outer shell were developed and designed from scratch in collaboration with Total Aircraft Services (TAS) (Van Nuys, California), Jet Propulsion Laboratory (JPL) (Pasadena, California), and NASA Dryden Flight Research Center giving the uncommon opportunity to compare the computer aided design (CAD) model mass properties with the results of a final product measurement.

Objective

The mass properties of an external modification to an aircraft are important parameters for preflight flutter boundary estimates. For the UAVSAR program the longitudinal center of gravity (CG) and yaw inertia were the chosen parameters, based on a preliminary flutter analysis, to experimentally verify the CAD estimates. The objective of the test is to compare the measured mass property values with the CAD model values, update the FEM accordingly, and acquire a more accurate preflight flutter boundary prediction. The CAD model used for comparison was a provided finite-element model (FEM).

Approach

The experimental design used for both the longitudinal CG test and the yaw inertia test is based on established methods used in the aerospace industry for decades. The UAVSAR pod is designed with two suspension lugs located at the top of the pod for connecting to a standard MAU-12 ejection rack. The test setup for measuring the longitudinal CG consisted of suspending the UAVSAR pod in the air by two cables fitted with load cells and attached to the pod suspension lugs. The test setup for measuring the yaw inertia was based on NACA Technical Note No. 351 (ref. 1) which describes obtaining the yaw inertia measurement using the bifilar torsion pendulum method, requiring the longitudinal CG of the test article to be directly centered in the test structure while the suspended pod is given a torque in yaw. With the CG unknown, a test rig was designed and fabricated that would allow the attachment of the MAU-12 ejection rack to slide along a steel tube making it unnecessary to know the exact location of the longitudinal CG of the test article before the day of testing. This test rig is designed such that it can be used repeatedly on flight-test articles equipped for mounting to a MAU-12 (within weight and CG limits), with minimal to no modifications. The UAVSAR (pod #2) fixed to the test rig during the yaw inertia test is shown in figure 1.

The results of the test showed a large discrepancy between both the UAVSAR pod total weight and the longitudinal CG. The UAVSAR (pod #2) weight and yaw inertia were overestimated by 28.6 percent and 13.3 percent, respectively, and the longitudinal CG was forward of the predicted by 2.01 in and forward of the predicted structural CG envelope by 0.38in. The results are summarized in table 1. Many circumstances led to these discrepancies, but mainly the errors occurred because dummy weights were used to model the weight of the actual instrumentation yet not model the lateral and longitudinal CG of the instrumentation. The wire weight was neglected in the manufactured pod and overestimated in the CAD model. Lastly, the test itself will add slight error because of the linearization of the derived angular momentum equations and possible friction in the pivot points.

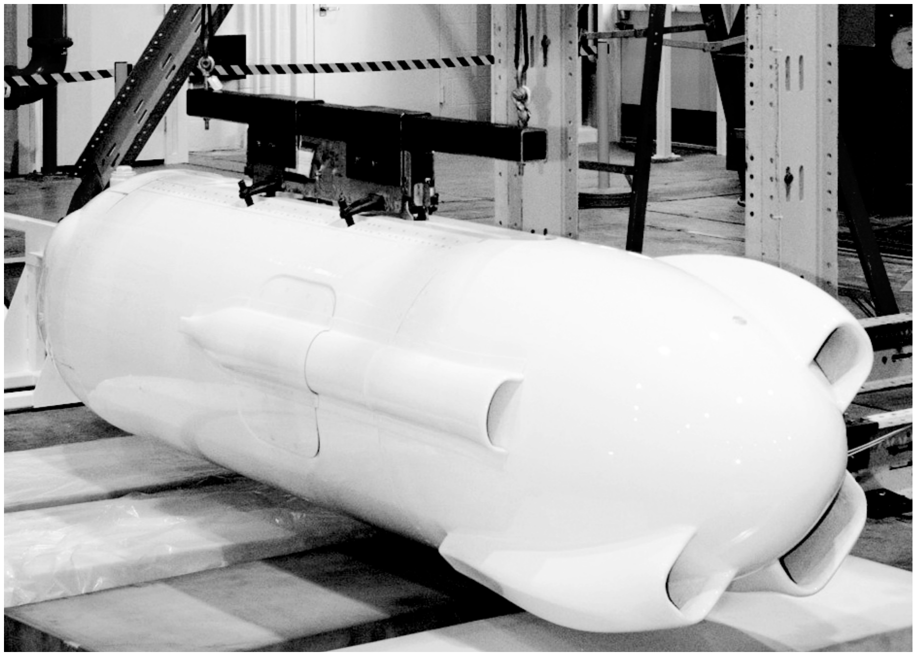
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Figure 1. UAVSAR yaw inertia test setup.

Table 1. Summary of longitudinal CG and yaw inertia test results on UAVSAR pod #2.

Mass Property Parameter	CAD Estimate	Test Results	Difference
Total Weight [lb]	1200	934	28.6%
Yaw Inertia [lb-in^2] (about Pod CG)	913,700	806,400	13.3%
Longitudinal CG [in] (Measured positive aft of MAU-12 center)	-0.87	-2.88	2.01in

Status

The UAVSAR pod #2 was approved for flight by the project structure engineers based on updated structural stress margins resulting from the enlarged CG envelope and flutter boundary estimates. As of February 2007, pod #2 is awaiting flight clearance by the Flight Readiness Review Board. The UAVSAR with actual SAR instrumentation (pod #1) is scheduled to be delivered March 2007, when these tests will be performed again. Testing pod #1 will improve the accuracy of the test results by omitting dummy masses and including the wire weight. A future test plan should include performing the yaw inertia test on an object with known yaw inertia, such as a simple beam, to discover any possible errors in the test setup avoidable in future testing. This error then needs to be compared with the error found in the CAD model to verify the necessity of this test method when CAD models are available.

References

1. Miller, M.P., *An Accurate Method of Measuring the Moments of Inertia of Airplanes*, NACA Technical Note No. 351, 1930.

Contacts

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AUTOPILOT INTERFACE COMPUTER FOR PLATFORM PRECISION AUTOPILOT

Summary

The NASA Unmanned Aerial Vehicle Synthetic Aperture Radar (UAVSAR) program is developing a Synthetic Aperture Radar (SAR) for ground measurements. The C-20A Gulfstream III (C-20A/GIII) (Gulfstream Aerospace, Savannah, Georgia) airplane is being used as an interim platform for the SAR prior to integration onto an Unmanned Aerial Vehicle (UAV). The Platform Precision Autopilot (PPA) is used to fly precision trajectories required by the SAR on the C-20A/GIII airplane. The embedded computer component of the PPA is referred to as the Autopilot Interface Computer (AIC). Design criteria for the AIC include low cost, high-performance, simple programming interface, environmental qualification, plus small size and low power for possible transfer to the UAV platform.

Objective

The AIC is required to host a precision control algorithm that calculates small corrections in pitch and roll for the C-20A/GIII airplane. These small corrections are needed to properly acquire science data for the SAR experiment on the airplane. Specifically, the control algorithm must maintain the airplane within 10 m of a predefined flight path. This is necessary to ensure, for example, radar data collected may be properly correlated with data collected from previous operations over the same area.

Approach

The AIC provides an Ethernet interface to the C-20A/GIII onboard inertial data, a serial interface to differential GPS on the SAR pod, a Controller Area Network (CAN) interface to an operator station on the airplane, and an analog interface to the Instrument Landing System (ILS) Interface System (I2S). The I2S is the component of the PPA that provides the pitch and roll corrections calculated by the AIC to the C-20A/GIII autopilot. The core of the AIC consists of a single board computer (SBC) hosting a Motorola (Schaumburg, Illinois) 565 microcontroller operating at 56 MHz. The SBC is available commercially at a low cost, and is qualified for operation in the flight environment. Additionally, an in-house-designed printed circuit board (PCB) is used to provide power and signal interfaces between the SBC and the AIC external interfaces.

The AIC software is developed in the Mathworks (Natick, Massachusetts) Matlab® and Simulink® Environment. It consists of Simulink® Block Diagrams which are translated into C code by the Mathworks Real Time Workshop (RTW) Embedded Coder and the Embedded Target for the Motorola MPC555 mircrocontroller. The Embedded Target allows for real time code execution on the MPC565 mircrocontroller, and provides Simulink® block drivers for each of the functional modules on the MPC565.

The AIC is shown in fig. 1.

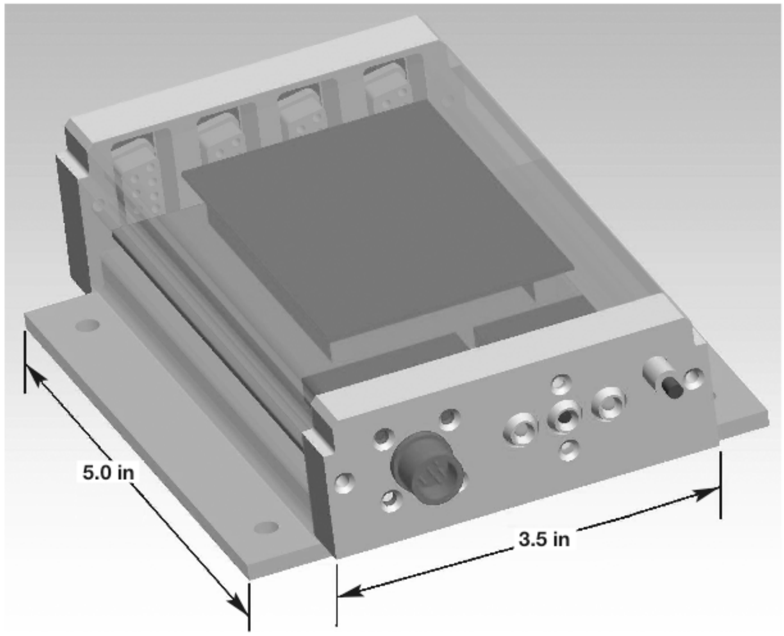

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Figure 1. Autopilot Interface Computer.

Status

The PPA system completed a flight readiness review in December 2006 and received approval for flight-testing in February 2007. Flight-testing of the precision autopilot will commence in early 2007 with a demonstration flight with the SAR planned later in 2007.

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THE C-20A/G-III PLATFORM PRECISION AUTOPILOT DEVELOPMENT

Summary

The NASA Unmanned Aerial Vehicle Synthetic Aperture Radar (UAVSAR) program is developing a Synthetic Aperture Radar (SAR) for ground measurements. A key element for the success of this program is a Platform Precision Autopilot (PPA). An interim vehicle, the NASA C-20A Gulfstream III (C-20A/G-III) (Gulfstream Aerospace Corporation, Savannah, Georgia) airplane, shown in figure 1, was selected to carry the radar pod and develop the PPA. The PPA interfaces with the C-20A/G-III airplane, making the onboard computer think it is flying an Instrument Landing System (ILS) approach. This technique retains the safeguards in the airplane autopilot. The PPA will enter initial flight-testing in early 2007.

ED07-0042-041

Figure 1. NASA C-20A/G-III with Synthetic Aperture Radar pod.

Objective

The PPA will enable repeat-pass flights within a 10 m tube for interferometric applications of the SAR being developed for the UAVSAR program. Flight lines are expected to be up to 200 km in length. The PPA must meet the 10-m tube requirement in conditions of light turbulence. The end product will be a “carefree” autopilot suitable for deployment and operation by the SAR scientists.

Approach

The PPA uses a Kalman filter to generate a real-time position solution with information from the C-20A/G-III airplane and a near real-time differential GPS unit located in the UAVSAR pod. The real-time navigation solution is used to compute commands (Guidance and Control modules) that, in turn, drive two modified ILS testers. The ILS tester units produce modulated RF signals fed to the onboard navigation receiver. These correction signals allow the C-20A/G-III autopilot to fly a simulated ILS approach that meets the requirements for UAVSAR operations. Figure 2 shows a block diagram of the system architecture.

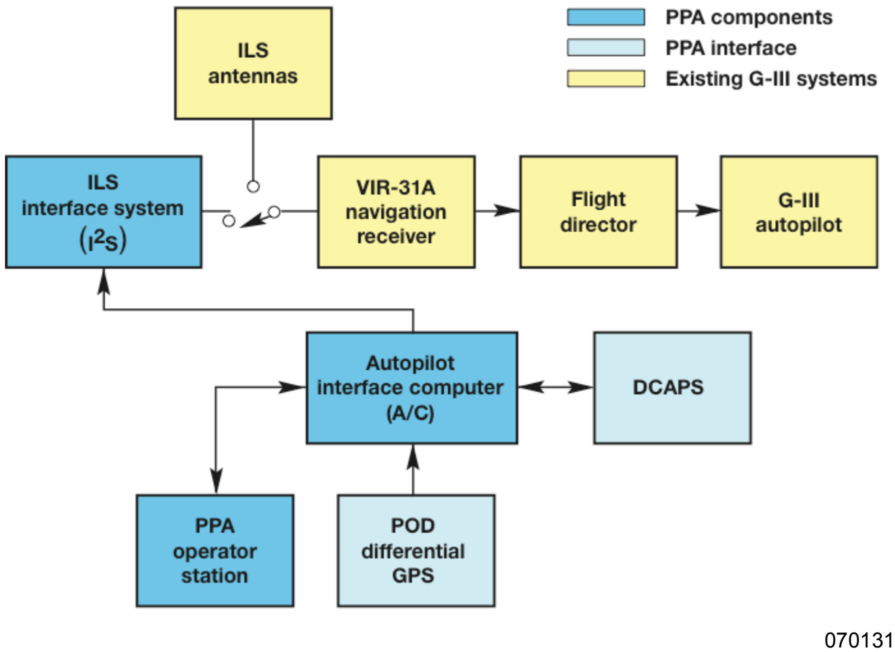

Figure 2. PPA System Architecture.

The PPA control approach is very similar to one used previously by the Danish Center for Remote Sensing for a similar SAR application (ref. 1).

The NASA Dryden Flight Research Center built a C-20A/G-III engineering simulation for development and evaluation of the PPA. A Monte Carlo capability was developed parallel with the C-20A/G-III simulation to examine the PPA performance in the presence of vehicle and atmospheric uncertainties. Perturbed parameters include uncertainties in aerodynamics, mass properties, system timing, atmospheric disturbances, and initial conditions. Figure 3 shows the 10-m tube precision autopilot tracking performance during a 500-run Monte Carlo analysis. The data below shows 6-min tracking the 10 m tube in the presence of light turbulence. The PPA meets the performance requirements in the simulation environment.

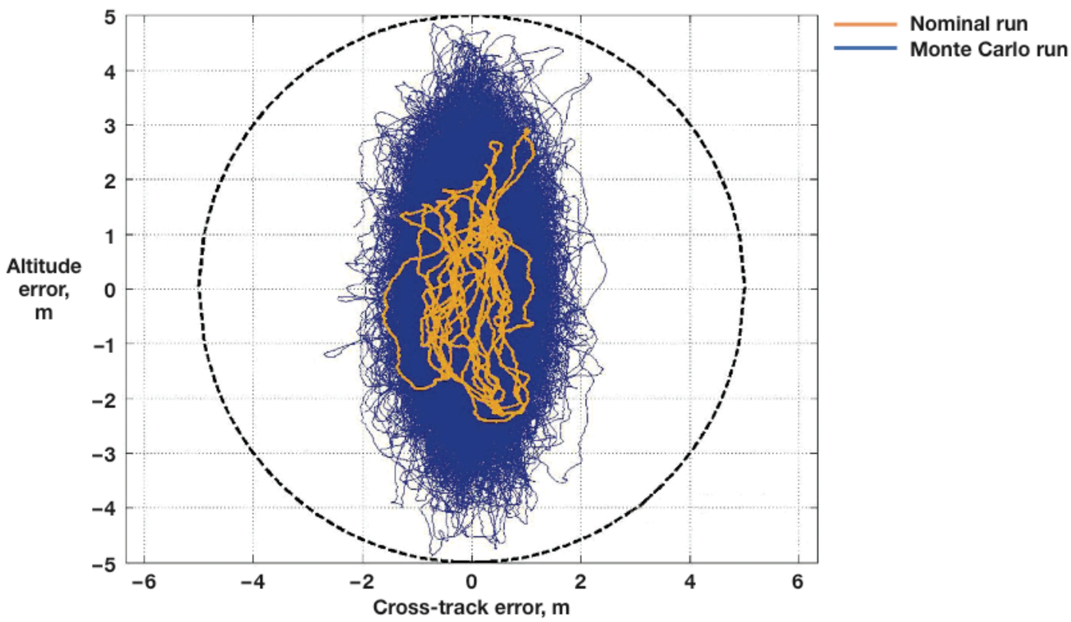

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Figure 3. C-20A/G-III Precision Autopilot 10-meter tube tracking performance.

Status

The PPA completed a flight readiness review in December 2006 and received approval for flight-testing in February 2007. Flight-testing of the precision autopilot will commence in early 2007 with a demonstration flight with the SAR planned later in 2007.

References

3. Soren Norvang Madsen, Niels Skou, Johan Granholm, Kim Wildt Woelders, and Erik Lintz Christensen, "A System for Airborne SAR Interferometry," *AEU International Journal of Electronic Communication* 50(1996) No. 2, 106-111.

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REAL-TIME STABILITY MARGIN ESTIMATION FOR THE X-48B BLENDED-WING BODY

Summary

A real-time stability margin (RTSM) estimation tool has been developed for in-flight robustness analysis of the X-48B (Boeing Phantom Works, St. Louis, Missouri) blended-wing body. The tool incorporates methods for generating excitation signals and the ability to analyze the open-loop frequency response during flight-testing. The excitation signals are mutually orthogonal and minimize the peak factors to provide multi-input excitation while avoiding excursions in flight condition. For in-flight analysis, an emphasis has been placed on comparison between flight data and simulation data in addition to estimated stability margins.

Objective

The objective of RTSM estimation is to improve envelope clearance efficiency and provide an early indication of potential modeling errors. Efficiency gains for envelope clearance are accomplished by reducing the time required for excitation. This is done by exciting multiple signals simultaneously and reducing the peak factor of the multisine excitations. Analysis of the open-loop frequency response, and comparison with the anticipated frequency response provide unique insight into developing modeling discrepancies and trends throughout the flight envelope.

Approach

A multisine signal is simply a sum of sinusoid signals, as indicated in eq. (1). The multisine signal is composed of sinusoids of various frequencies (ω_k), phases (ϕ_k), and relative power (P_k). To generate a mutually orthogonal set of multisine signals, the frequency components should vary linearly between the minimum and maximum frequency of interest. The relative power of each frequency component can be tuned to achieve a tailored power distribution for each channel of the multisine signal. The signal component phases are determined by minimizing the peak factor of each channel by use of an optimization routine, as described in reference 1.

$$u_j = \sum_{k=1}^{M} \sqrt{\frac{P_k}{2}} \cos(\omega_k t + \phi_k) \tag{1}$$

For the X-48B aircraft, multisine signals were developed to excite the roll, pitch, and yaw channels prior to the control allocation, as indicated in figure 1. The excitations vary in frequency between 1 rad/s and 75 rad/s, yielding a 19 s excitation signal. Each channel is composed of 25 individual frequencies. The power spectrum was tailored to increase vehicle response in a narrow bandwidth on a single channel. The peak factor of each channel is approximately 1.25.

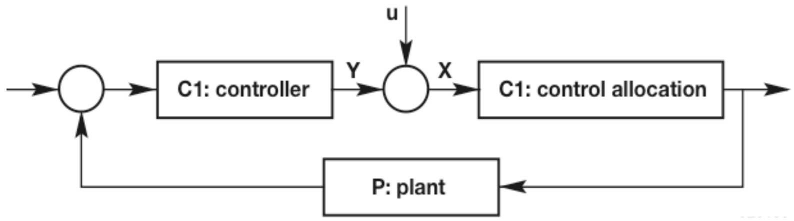

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Figure 1. Block diagram indicating the excitation signal (u) location.

In-flight robustness is realized by estimating the transfer function response of the open-loop system, seen in figure 1. The excitation signals (*u*), and transfer function inputs (*X*) and outputs (*Y*) are all vectors of length 3, representing the roll, pitch, and yaw channels. Equation (2) represents the open-loop transfer function of interest.

$$\frac{Y}{X} = C_1 P C_2 \tag{2}$$

The vehicle open-loop transfer function can be estimated given closed-loop time-histories of both signals X and Y during excitation of signal *u*. Recorded data from the telemetry system will be used by the analysis tools during flight. While the multi-channel excitation enables analysis of the system as a multi-input, multi-output (MIMO) system, more insightful data can be garnered by analyzing the system as three single-input, multi-output (SIMO) systems. The analysis tools produce a series of time history plots, power spectral density plots for the inputs and outputs, and nine Bode plots including coherence and stability margin information.

Status

Flights of the X-48B aircraft utilizing the RTSM excitation and analysis tools will be conducted in 2007.

Reference

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SOFIA TO FLY

Summary

The Stratospheric Observatory For Infrared Astronomy (SOFIA) is a 2.5 m, optical, infrared, submillimeter telescope mounted in a Boeing (Chicago, Illinois) 747 airplane, to be used for many basic astronomical observations performed at stratospheric altitudes. A ground mission facility will be created to support the aircraft and science missions. The observatory will accommodate installation of different focal plane instruments with in-flight accessibility provided by investigators selected from the international science community. The observatory objective is expected to have an operational lifetime in excess of 20 years.

Objective

The objective of this project is to deliver to the science community an infrared flying observatory and collect astronomy science data in the infrared spectrum.

Approach

A consortium of contractors and NASA centers is, at this time, building SOFIA. A 747 SP airplane is being modified by L-3 Communications in Waco, Texas to be the observatory platform. The Ames Research Center (ARC) in Mountain View, California, and the Dryden Flight Research Center (DFRC) in Edwards, California, are managing the various subsystems on the airplane. Day-to-day program management of the observatory will be conducted at DFRC.

The observatory has the telescope mounted in the back of the airplane with tracking and moveable doors to enable the telescope to observe the night sky. The telescope has an oil-bearing system to float it for shock and vibration that will keep it from being damaged. It also has a pneumatic system to limit shock and vibration. There is an upper rigid door (URD) that closes the cavity where the telescope is located. This URD, a lower flexible door (LFD), and a telescope aperture assembly (AA) move with the telescope to maintain a consistent observation window. The URD closes off the telescope cavity whenever the system is not in use. The cavity has an environmental control system to keep the telescope at a controlled temperature and humidity environment when the URD is closed.

A mission control and communications system (MCCS) is being developed to control the telescope, cavity doors, limited aircraft flight path, and communications between science stations while in observatory mode. The MCCS will control the telescope tracking system, the cavity doors, the aircraft flight management system, data collection, and communication with workstations.

Status

The airplane finished its first flight in April 2007. The MCCS is not operational at this time. The telescope for first flights will be in a safe mode with the oil-bearing floating system operational. The CECS will be operated in a limited capacity to help keep the telescope dry at altitude.

After functional check flights, the aircraft will be flown to DFRC for further system preparations for full operation. Full operation is several years from now.

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GORDON'S UNMANNED AERIAL VEHICLE TRAINER: A FLIGHT-TRAINING TOOL FOR REMOTELY-PILOTED VEHICLE PILOTS

Summary

In 2007, the NASA Dryden Flight Research Center (DFRC) will be flying the Ikhana Predator B (General Atomics Aeronautical Systems, Inc., San Diego, California) unmanned aerial vehicle (UAV) (fig. 1), and the X-48B (The Boeing Company, Chicago, Illinois) Blended-Wing Body (BWB) aircraft (fig. 2). These high-value UAVs require a remote pilot to fly the vehicle while looking through a forward-looking camera located in the vehicle nose during take off, pattern operations, and landing. These operations are high-risk to the vehicle because of the poor field of view, lack of peripheral vision and external cues, lack of sensory feedback, and system time delays for uplink command and video downlink. Recently, DFRC pilots have flown only piloted vehicles and were concerned with the lack of familiarity with operating a remotely-piloted vehicle (RPV) through a video camera.

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Figure 1. Ikhana Predator B.

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Figure 2. X-48 Blended-wing body.

Objective

Gordon's UAV Trainer (GUT) was developed as a training tool for training pilots to fly RPVs through a forward-looking camera in the vehicle nose. Two low-cost UAVs were configured for these operations and flown during 2006. With this tool, pilots are able to gain some experience flying "through a camera" on an inexpensive platform prior to flight with the research vehicles.

Approach

Low-cost aircraft were equipped with remote control equipment used in hobby aircraft. Rate gyros were added to the yaw and roll axes to dampen the aircraft response. The aircraft were instrumented with a video camera aligned with the X-axis of the vehicle. Each aircraft is remotely controlled from a van-mounted cockpit (fig. 3) that has basic stick, rudder, and throttle inputs. The pilot observes the aircraft attitude and position via onboard video and a moving map.

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Figure 3. GUT cockpit.

The video also displays air speed, altitude, engine RPM, and battery voltage that is added to the video signal before telemetry to the ground. The vehicle data is displayed on a moving map provided by a GPSFlight™ (Tuckwila, Washington) tracking system. The GPSFlight™ unit measures aircraft position, heading, altitude, velocity, and range from the pilot station.

System Architecture

Figure 4 shows the block diagram of the system architecture. With the exception of the pilot cockpit, the rest of the equipment may be replaced with commercial-off-the-shelf hardware.

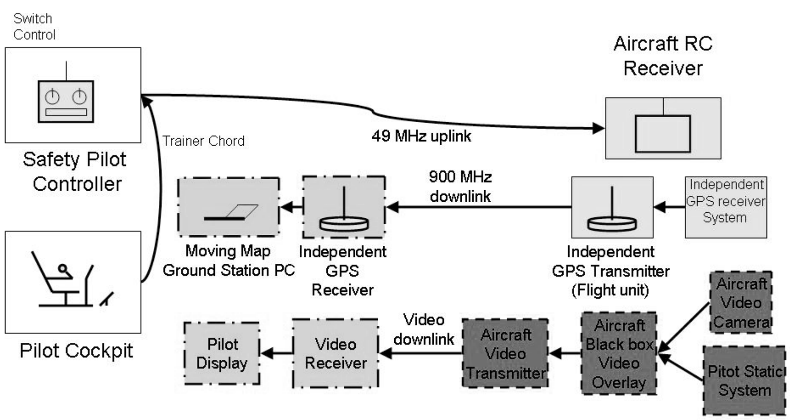

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Figure 4. GUT system architecture.

Status

The GUT system was flown successfully during 2006 in preparation for Ikhana training flights. The APV-3 (RnR Products, Milpitas, California) aircraft, seen in figure 5, was lost during a training flight because of a wiring problem between the ground cockpit and the safety pilot controller. The wiring problem has been fixed and the system is integrated into the DFRC utility vehicle UAV, seen in figure 6. The system is available as a training tool for Ikhana, BWB and other projects that require remotely piloted control.


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Figure 5. APV-3.


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Figure 6. DFRC utility vehicle UAV.

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IKHANA, THE FIRST "INTELLIGENT" UNMANNED RESEARCH TESTBED

Summary

The NASA Dryden Flight Research Center (DFRC) has acquired an MQ-9 Predator B unmanned aircraft system (UAS), seen in figure 1, to be used as a research testbed. The vehicle is manufactured by General Atomics Aeronautical Systems, Inc. (GA-ASI) (San Diego, California), and was procured through the assistance of the Air Force Flight Test Center at Edwards, California. A flight research computer will be integrated into the vehicle to conduct autonomous flight research experiments, such as collision avoidance, intelligent mission planning, or precision autopilot control. Modifications to the aircraft and ground control station (GCS) software are in development. The vehicle will then be able to provide duo role capability, conducting science mission flights in addition to research experiment flights. This is the first time that a large, high-altitude, long endurance UAS has been used in this manner.

ED-07-0038-052

Figure 1. NASA Ikhana UAS.

Objective

The objective of developing this platform is to support subsonic fixed-wing and airspace programs to validate potential solutions to fundamental technology barriers. There are two basic capabilities of the vehicle that can be used to accomplish this objective. First, there is an Airborne Research Test System (ARTS III) that can host research experiments and algorithms for autonomous aircraft control; mission validation flights of advanced sensor technologies can also be performed.

Approach

The NASA UAS vehicle is called Ikahna (ee-kah-nah). The name is derived from a Native American word of the Choctaw Nation meaning "intelligent, aware, or conscious." It is aptly named because the ARTS III computer can be programmed to fly the aircraft autonomously, allowing researchers to investigate, for example, collision avoidance or intelligent mission planning algorithms. The ARTS III computer is built by the Institute for Scientific Research (ISR) in Fairmont, West Virginia.

The heart of the ARTS III computer is a 1 GHz Power PC flight-critical processor, which is used to monitor all input and output signals between the Ikhana flight control computer and the host processors that contain the research experiment or algorithms. This is done to ensure that the autonomous commands are within the operational and performance envelope of the vehicle. There are two dedicated host processors in ARTS III that are also 1GHz Power PCs. The system features 24 analog-to-digital input channels, 3 digital-to-analog outputs, and 10 digital input/output channels. It also supports MIL-STD-1553A, RS422, ARINC 429, and Ethernet communication with other payloads or research systems. All three CPUs and associated hardware are contained within a 6U ruggedized VME chassis, shown in figure 2.

Figure 2. ARTS III.

Expanding the standard capabilities of the aircraft into a flexible research platform requires modification of the flight control computer and GCS software, currently being performed by GA-ASI. These changes are designed to achieve reversion to the standard MQ-9 flight characteristics if the research system is engaged. When commanded, the vehicle will disengage the ARTS III, attain level steady state flight, and return control back to the pilot.

Hardware modifications to the aircraft and GCS will be performed by NASA. Vehicle upgrades include an onboard data recorder, a GPS-based time code generator, and a flight termination system. The GCS will be outfitted with more engineering workstations and displays, along with higher data update rates than the standard MQ-9 GCS. In addition, command and control of the ARTS III will require a dedicated PC workstation.

When the development of this research platform is complete, a number of verification and validation tests will be conducted to ensure that the modifications are safe to fly. In a near term mission, the aircraft will be used extensively for surveying forest fires in the western states; provide the U.S. Forest Service with near real-time infrared images and data on the severity, location, and activity of the fires. Ikhana is also planning on flying a number of piggyback experiments during this timeframe; one is called Argus, and the other is called the Fiber Optic Wing Shape Sensing (FOWSS) experiment. The Argus system is a diode laser spectrometer that measures the mixing ratios of carbon monoxide, methane, and nitrous oxide concentrations in the atmosphere. These gases allow scientists to analyze smoke plumes and determine the origin of the fire. The FOWSS experiment consists of a laser system and fiber-optic sensors that will be used to measure the shape of the wings. The performance of the FOWSS experiment will be compared to a set of standard strain gage measurements taken from the wing as well.

This technology could potentially be used to avoid dangerous flutter conditions on large, flexible-wing vehicles.

Status

The fire missions, along with the piggyback experiments, will be conducted in the summer of 2007. Modifications to the aircraft and GCS hardware will occur after that time to accommodate the ARTS III and associated systems. The ARTS III units are undergoing environmental testing. Software for the Ikhana flight control computer, GCS, and ARTS III are currently in development.

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FIBER OPTIC WING SHAPE SENSING ON THE IKHANA VEHICLE

Summary

In June 2003, the Helios (AeroVironment, Inc., Monrovia, California) prototype unpiloted aerial vehicle (UAV) experienced significant pitch instability during low-altitude flight that lead to a catastrophic structural failure and in-flight breakup. One of the most significant lessons learned from the mishap investigation was the requirement to provide real-time measurement of wing shape. This measurement could ultimately be used as feedback into the flight control system for aeroelastic motion control. Wing shape sensing (WSS) is, therefore, an essential first step towards achieving the ultimate goal of actively controlling the wing shape during flight, reducing aerostructural loads and avoiding such failures in the future.

Calculating real-time wing shape is particularly challenging especially for lightweight, highly-flexible structures because of the stringent weight and volume requirements, both for structural sensors and supporting systems. A recent study assessed the viability of using conventional strain gage instrumentation and demonstrated that the wire weight alone represented a prohibitive weight penalty and was impractical to implement for many aerospace vehicles. Alternatively, lightweight and low-profile fiber optic wing shape sensors (FOWSS), in conjunction with computationally efficient algorithms, were viewed as a promising approach to providing very accurate wing measurement calculations for eventual input to the flight control system for aeroelastic motion control. Both inhabited and uninhabited aircraft will benefit from this technology.

Objective

The overall goal of the FOWSS flight validation test, Phase I, is to provide an advanced demonstration of real-time, fiber optic wing shape sensing technology. These flight tests will allow the NASA team to assess the technical viability of incorporating wing shape-sensing measurements into the vehicle flight control algorithms.

The Phase 1, FOWSS flight validation test consists of the following objectives:

1. Flight-validate fiber-optic sensor measurements and real-time, wing shape-sensing predictions on the NASA Ikhana vehicle [fiscal year 07 (FY07].
2. Validate fiber optic mathematical models and design tools (FY08).
3. Assess technical viability and, if applicable, develop methodology and approach to incorporate wing shape measurements within the vehicle flight control system (FY08–FY09).
4. Develop and flight-validate advanced approaches to perform active wing shape control using
    a. conventional control surfaces (FY08–FY10)
    b. active material concepts (FY09–FY11 and beyond)

Approach

The overall approach envisaged in this effort is to utilize lightweight fiber-optic sensors and miniaturized systems together with efficient structural algorithms to calculate wing shape in real time. Hundreds of localized strain measurements will be acquired and used as input to wing shape predictive algorithms. Independent strain gage sensors will be used to validate fiber optic measurements for the first time in flight. This effort builds upon the successful laboratory validation of these technologies in support of the Pathfinder Plus (AeroVironment, Inc., Monrovia, California) vehicle from 2005. The Ikhana airframe, an all-composite Predator-B (General Atomics Aeronautical Systems, Inc San Diego, California) system, is representative of modern flexible aircraft structures. The Ikhana Airborne Research Test System (ARTS) will host research control laws that could be used to actively control wing shape and/or loads. The research will be conducted in three phases, titled:

Phase I: In-Flight Validation of Fiber-Optic Sensors for Wing Shape Sensing (FY07)

Phase II: In-Flight Validation of Fiber Optic Wing Shape Sensing and Operational Loads Monitoring (FY08)

Phase III: Validated Methods for In-Flight Active Wing Shape Control (FY09-11)

Status

As seen in figure 1, the Fiber Optic Wing Shape Sensors (FOWSS) team made significant progress toward both analytical modeling and preparing for the experimental validation of the FOWSS system. The algorithms that convert the local in-plane strain measurements to global out-of-plane wing shape were modified for the Ikhana vehicle. A patent application, entitled “Method for Real-time Shape Sensing” was submitted in December 2006 (ref. 1). The FOWSS system design is complete and the majority of the flight-ruggedized system components, seen in figure 2, have been ordered. A plan for environmental testing of the system has been developed. An Objectives and Requirements Document (ORD) has been developed (in line with Dryden Flight Research Center processes) to define the flight objectives, instrumentation requirements, installation requirements, maneuver definitions, and ground monitoring needs. A preliminary instrumentation installation plan, seen in figure 3, has been defined that identifies the best approach for installing the FOWSS controller (including the laser), fiber-optic sensors, and strain gages.

$$y_i = \frac{(\Delta l)^2}{6}\sum_{j=1}^{i}\frac{1}{c_{j-1}}\left\{\left[3(2j-1)-(3j-2)\frac{c_{i-j+1}}{c_{i-j}}\right]\bar{\varepsilon}_{i-j}+(3j-2)\bar{\varepsilon}_{i-j+1}\right\}+y_0+i\Delta l\tan\theta_0$$

$$(i = 1, 2, 3, \ldots, n)$$

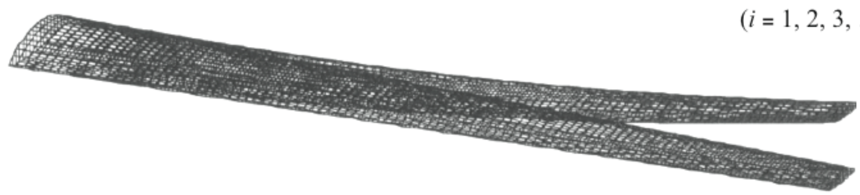

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Figure 1. Fiber-Bragg flight system.

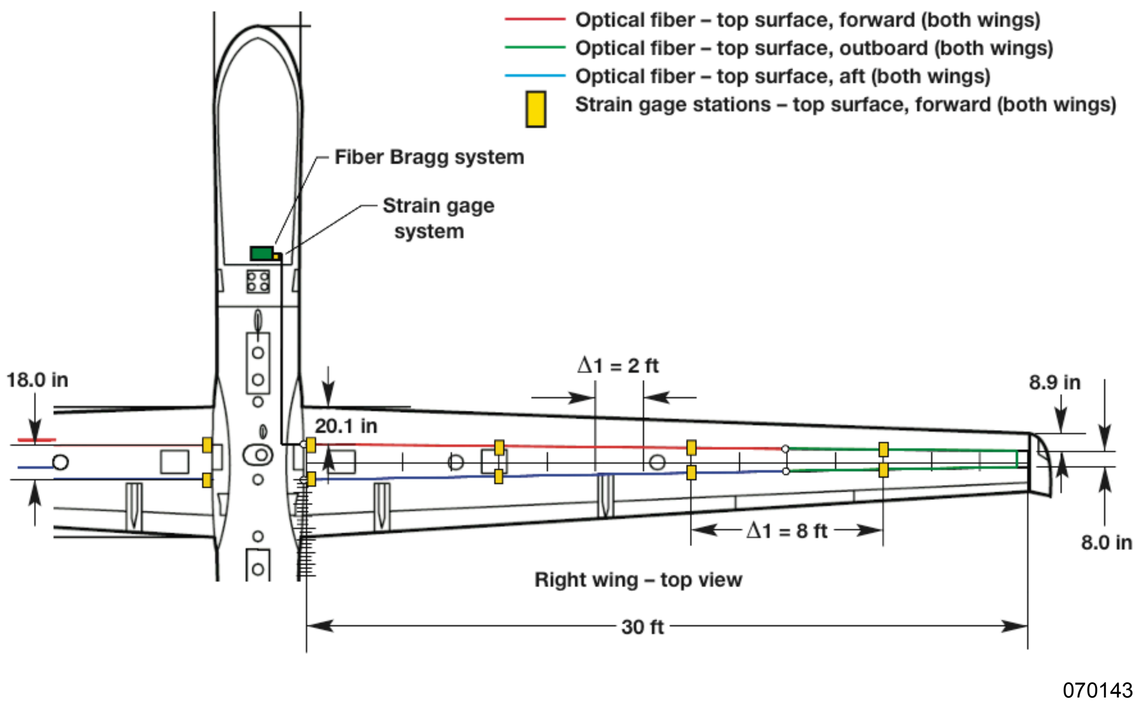

Figure 2. Finite-element models and analytical equations.

References

1. Ko, William L. and W. Lance Richards, *Method for Real-time Shape Sensing*, Patent Application DRC 006024, December 2006.

Contacts

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AEROELASTIC MODEL STRUCTURE COMPUTATION FOR ENVELOPE EXPANSION

Summary

Structure detection is a procedure for selecting a subset of candidate terms, from a full model description, that best describes the observed output. This is a necessary procedure to compute an efficient system description that may afford greater insight into the functionality of the system or a simpler controller design. Structure computation as a tool for black-box modeling may be of critical importance in the development of robust, parsimonious models for the flight-test community. Moreover, this approach may lead to efficient strategies for rapid envelope expansion, which may save significant development time and costs.

In this study, a least absolute shrinkage and selection operator (LASSO) technique is investigated for computing efficient model descriptions of nonlinear aeroelastic systems. The LASSO minimises the residual sum of squares by the addition of an ℓ_1 penalty term on the parameter vector of the traditional ℓ_2 minimisation problem. The use of LASSO for structure detection is a natural extension of this constrained minimisation approach to pseudolinear regression problems which produces some model parameters that are exactly zero and, therefore, yields a parsimonious system description. Applicability of LASSO for model structure computation for the NASA F/A-18 (McDonnell Douglas Corporation, St. Louis, Missouri, and Northrop Corporation, Newbury Park, California) airplane Active Aeroelastic Wing (AAW) using flight-test data is shown for several flight conditions (Mach numbers) by identifying a parsimonious system description with a high-percent fit for cross-validated data.

Objective

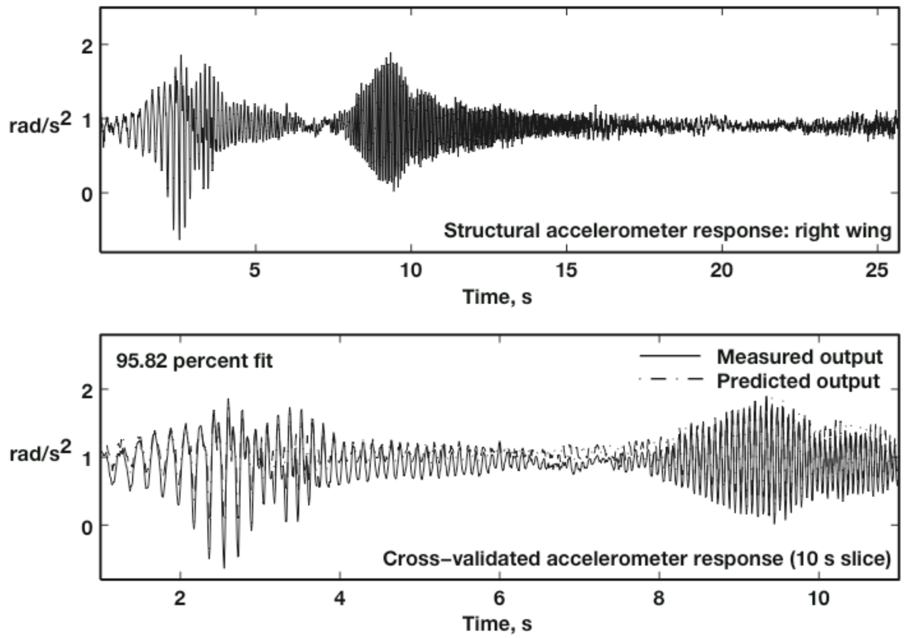
System identification, or black-box modeling, is a critical step in aircraft development, analysis and validation for flightworthiness. The development and testing of aircraft typically takes many years and requires considerable expenditure of limited resources. One reason for lengthy development time and cost is inadequate knowledge of an appropriate model type or structure to use for parameter estimation. Selection of an insufficient model structure may lead to difficulties in parameter estimation, giving estimates with significant biases and/or large variances. This often complicates control synthesis or renders it infeasible. The power of using structure detection techniques as a tool for model development (i.e. black-box modeling) is that it can provide a parsimonious system description which can describe complex aeroelastic behaviour over a large operating range. Consequently, this provides models that can be more robust and, therefore, reduce development time.

Moreover, when studying aeroelastic systems it may not be practical to assume that the exact model structure is well known a priori. In aerospace systems analysis one of the main objectives is not only to estimate system parameters but also to gain insight into the structure of the underlying system. Therefore, structure computation is of significant relevance and importance to modelling and design of aircraft and aerospace vehicles. Structure computation may indicate deficiencies in an analytical model and could lead to improved modelling strategies and also provide a parsimonious, black-box, system description for control synthesis.

Approach

Flight data was gathered during subsonic flutter clearance of the F/A-18 AAW. At each flight condition, the airplane was subjected to band-limited white inputs, with uniform distribution and zero-mean. The inputs correspond to collective and differential aileron, collective and differential outboard leading edge flap, rudder, and collective stabilator excitations in the range of $\pm$ 0.5 rad and 30 Hz bandwidth for 26 s. This report considered accelerometer data measured during the collective aileron sweeps at Mach 0.85 and 0.95, both at an altitude of 4,572 m (15,000 ft).

Figures 1(a) and 1(b) show the predicted output for a cross-validation data set for the identified structures.



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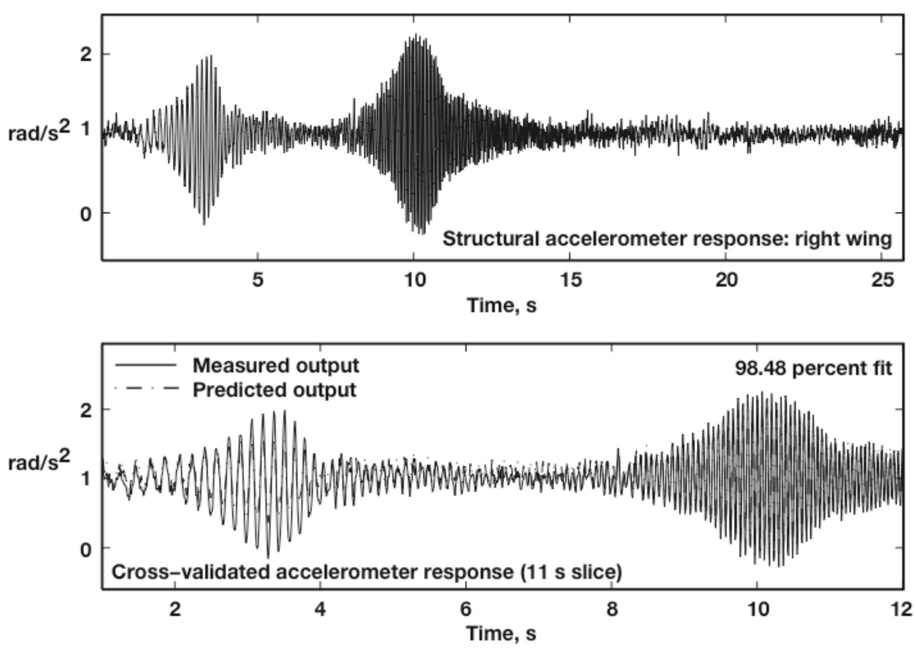
(a) Mach 0.85, Alt. 4,572 m (15,000ft).



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(b) Mach 0.95, Alt. 4,572 m (15,000ft).

Figure 1. Cross-validation data set output.

The upper panel displays the full 26 s time history of the accelerometer response recorded on the right wing. The lower panel displays a 10 (Mach 0.85) and 11 (Mach 0.95) second slice of the predicted output superimposed on the measured output. For Mach 0.85 [fig. 1(a)] the predicted output accounts for over 95 percent of the measured outputs variance whilst for Mach 0.95 [fig. 1(b)] the predicted output accounts for over 98 percent of the measured outputs variance. The results demonstrate that the computed model structures are capable of reproducing the measured output with high accuracy.

Status

The LASSO is a novel approach for detecting the structure of overparameterised nonlinear models. These results may have practical significance in the analysis of aircraft dynamics during envelope expansion and could lead to more efficient control strategies. In addition, this technique could allow greater insight into the functionality of various systems dynamics, by providing a quantitative model that is easily interpretable.

Contacts

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F-15 INTELLIGENT FLIGHT CONTROLS

Summary

Eighteen flights were flown in early 2006 providing evaluation of a direct adaptive neural-network-based flight control concept. A highly modified NF-15B (McDonnell Douglas Corporation, St. Louis, Missouri, now The Boeing Company, Chicago, Illinois) airplane, tail number 837 was used as the demonstration vehicle. It was shown that in some cases improved handling qualities were observed as a result of the adaptive system.

Objective

The use of neural networks and similar adaptive technologies in the design of highly fault- and damage-tolerant flight control systems shows promise in making future aircraft far more survivable than current technology allows. During the flight evaluations in 2006, the neural network was engaged and allowed to learn in real time to dynamically alter the aircraft handling qualities characteristics in the presence of simulated failure conditions. The objective was to demonstrate an improvement of one Cooper-Harper rating level when the adaptation system is engaged.

Approach

A simplified Sigma-Pi neural network was implemented in a direct adaptive control architecture. When significant tracking errors are encountered, the neural network adjusts to counteract them. Failures were simulated by freezing a stabilator control surface and also by changing an angle-of-attack feedback to artificially destabilize the vehicle. Formation flight and air-to-air tracking tasks were flown and Cooper-Harper handling qualities ratings were assigned.

Results

For the simulated destabilization failure, handling qualities ratings indicated a modest improvement in performance. The ratings improved from 4 to 3, moving from the level 2 region to the level 1 region. As seen in figure 1, the severity of the failure was less than predicted by simulation so the improvements attributed to the neural network were less dramatic than desired. A software change will provide a larger failure that will hopefully provide a better demonstration of the neural network capability.

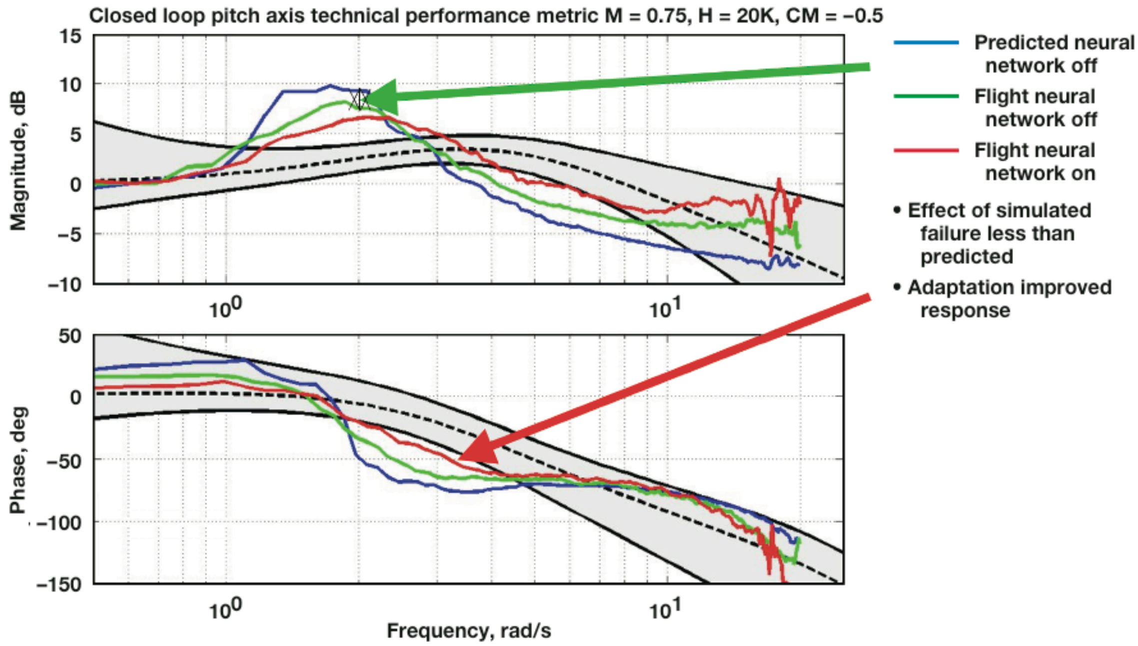

Figure 1. Simulated destabilization failure and angle-of-attack feedback change.

For the frozen stabilator, the responsiveness in the pitch axis was definitely improved by the neural network, however, a tendency for roll pilot-induced oscillation (PIO) was introduced. Further investigation showed that dead zones within the system prevented the adaptation from adjusting for the PIO. Design improvements to the neural network are being made to address this limitation.

Status

The F-15 Intelligent Flight Control System (IFCS) project will continue to support research in adaptive controls under the Integrated Resilient Aircraft Controls (IRAC) project under the Aeronautics Research Mission Directorate (ARMD). The work will include further refinement to the existing neural network algorithm as well as development of handling qualities metrics for asymmetric vehicles.

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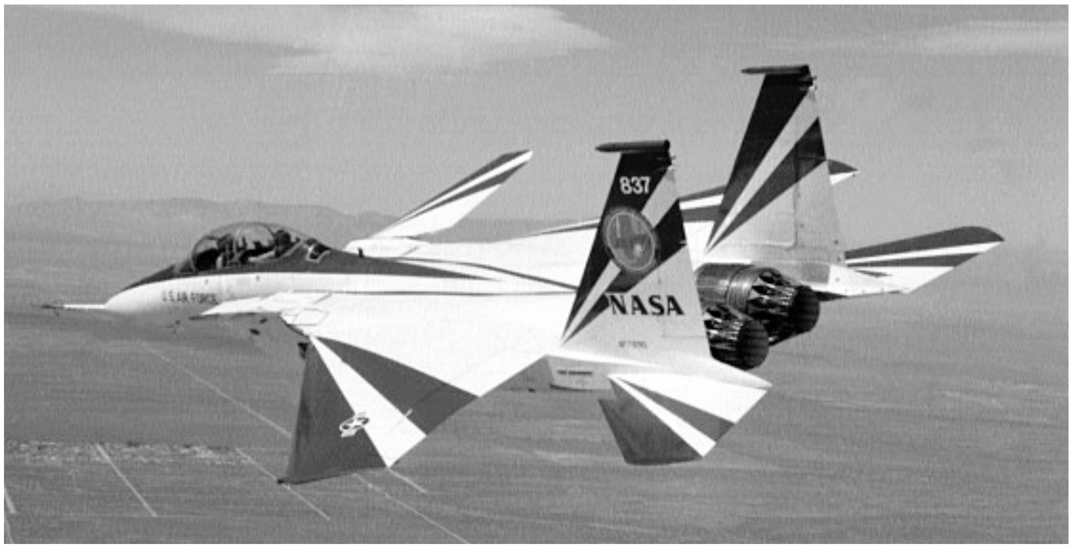
F-15 INTELLIGENT FLIGHT CONTROL SYSTEM NEURAL NETWORK FLIGHT SYSTEMS SUMMARY

Summary

The NASA F-15 (McDonnell Douglas Corporation, St. Louis, Missouri, now The Boeing Company, Chicago, Illinois) airplane was used as a testbed to flight-test a neural network controller algorithm called "Sigma Pi." The airplane is shown in figure 1.



EC96-43485-3

Figure 1. The NASA F-15, tail number 837 intelligent flight control system airplane.

Two types of failures were included to demonstrate the effectiveness of the neural network controller.

A simulated change in aircraft stability was achieved by changing the gain on the angle-of-attack feedback to the symmetric canard. With fixed canards, the aircraft is both statically and dynamically unstable. Angle-of-attack feedback to the canards is required for longitudinal stability.

A change in control effectiveness was simulated by biasing and freezing one of the stabilator control surfaces.

The flight test was performed at Mach 0.75 at an altitude of 20,000 ft with maneuvering at 3 g tracking and formation tasks. Results from flight tests show that the canard multiplier failures were less severe than predicted by the nonlinear simulation. The adaptive system seemed to provide some improvement with these failures; however, the change was less dramatic than was predicted.

The stabilator failures provided a good example of an asymmetric vehicle. The neural networks provided some relief from the coupled behavior, however, with the neural networks engaged, the system tended to be much more PIO-prone in the pitch axis. Pilot control stick motions revealed that pilot compensation was adequate to deal with most of the cross coupling when accomplishing a pitch task. For task accomplishment requiring motions in the lateral axis, however, pilot compensation was less successful.

Objective

The objective of the experiment was to determine if a neural network and similar adaptive technologies in the design of highly fault- and damage-tolerant flight control systems shows promise in making future aircraft far more survivable than current technology allows. Positive results would hopefully stimulate new technologies in this area.

Approach

The approach is to degrade the flying qualities of the F-15 airplane by inducing a frozen stab offset failure or destabilizing the airplane by changing the gain of the angle-of-attack signal to the canard schedule. The neural network signal is used to compensate for the errors in the dynamic inversion model and restore the handling qualities.

Status

The flight tests showed that the current failure magnitudes did not degrade the handling qualities as significantly as desired; consequently, the improvement by the neural network was minimal. The next flight phase will include greater failures to affect the flying qualities so that the neural network compensation will be more significant. New flight computer software to increase the angle-of-attack multiplier to the canard has completed testing at Boeing (Chicago, Illinois) and is ready to be installed into the F-15 airplane, tail number 837 for the next series of flight tests. In addition, two new neural network designs are being developed and should be ready for flight test later this year.

Contact

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F-15B QUIET SPIKE™ STRUCTURAL MODE INTERACTION GROUND TEST AND AEROSERVOELASTIC FLIGHT TEST

Summary

In preparation for the F-15B airplane (McDonnell Douglas Corporation, St. Louis, Missouri, now The Boeing Company, Chicago, Illinois) Quiet Spike™ (Gulfstream Aerospace, Savannah, Georgia) (QS) flight test, baseline F-15B airplane and modified F-15B QS Structural Mode Interaction (SMI) ground tests were performed to compare with military specification requirements for safety-of-flight. Ground-test results were unsatisfactory, so there was some concern about aeroservoelastic (ASE) stability margins in flight. The baseline F-15B airplane was cleared through extensive previous flight-testing and a dedicated ASE flight (testing low dynamic pressure, the worst case). Unfortunately, the modified F-15B-QS configuration had significantly lower margins from both the ground SMI tests, and from an ASE analysis with coupled aerostructural control dynamics, especially with the spike-boom retracted at higher angle of attack (AOA) with less fuel weight. This resulted in an extensive ASE flight clearance based on the SMI and ASE results.

Objectives

Structural Mode Interaction ground tests

- Define the dynamics of the airframe with actuators and dynamic coupling between the airframe and flight control system by changing control feedback gains.
- Determine effects of increasing gains in control system sensor feedbacks up to 8dB (factor of 2.5).
- Acquire open- and closed-loop frequency response data for ASE model updating.

Success Criteria: Stability in closed-loop aircraft responses up to at least 6dB (factor of 2.0), and quantify angle-of-attack and gain-margin relationship.

Aeroservoelastic flight test

Establish an ASE stability flight envelope for all F-15B-QS configurations and flight-test research conditions.

Approach

Structural Mode Interaction ground tests

Ground tests were performed with the gear down and deflated tires, nominally with the control augmentation system turned on (CAS-on) in all axes. A gear-up configuration could be simulated with proper angle of attack and airspeed conditions. An adjustable gain box was used to modify the control law loop gain during closed-loop testing, provide the interface to obtain the desired open-loop frequency responses, and to provide an emergency shutdown capability by opening all the flight control feedback loops. The QS boom was both retracted and extended (and half-extended) with the airplane in both high- and low-fuel configurations. Here is a summary of the results (Baseline = F-15B airplane).

No lateral-directional anomalies - 8dB satisfied for ALL configurations

Gear-down longitudinal - at least 8dB margin (spike extended or retracted)

Gear-up longitudinal - stable-to-LCO (10-13hz) gain factor ranges (LCO = limit cycle oscillation)

Table 1. SMI ground test results summary.

	Retracted	**½-Extended**	**Extended**
Heavy: 16-deg AOA (Baseline = 3.5dB)	0–0.8dB LCO at $\times$ 1.1	3.5–5dB	3.5–6dB (9hz)
Heavy: 7-deg AOA (Baseline = 6dB)	0.8–1.6dB LCO at $\times$ 1.2	5–6dB	6–8dB (9hz)
Light: 16-deg AOA	0dB Unstable	0.8–1.6dB	3.5–6dB
Light: 7-deg AOA	0–0.8dB LCO at $\times$ 1.1	3–3.5dB	6–8dB

The main culprit in the poor retracted-boom configuration results was an Nz-feedback to the stabilator amplification with CAS-on at higher AOA (gain) that caused the LCO responses.

Aeroservoelastic flight test

Since the SMI test results show that the extended-spike configuration is more like the baseline F-15B airplane without the spike, this configuration was cleared for the entire research flight envelope before gear-up retracted-boom clearance was attempted with CAS-on. There was a fairly constant boom response across Mach and Qbar with no noticeable ASE structural response in surfaces or feedbacks. The extended-boom configuration matched the baseline F-15B airplane very well.

The retracted-boom configuration was then cleared for selected subsonic and supersonic flight conditions. The subsonic envelope to Mach 0.8 was cleared at altitudes of 15,000, 30,000, 25,000, and 8000 ft, and tower-flyby, followed by supersonic clearance up to Mach 1.4 at an altitude of 40,000 ft.

Status

The higher dynamic pressure test points seemed to exhibit less ASE response in the stabilators even though the Nz response was more pronounced. This may be attributed to higher damping on the stabilators than the analysis indicated. Other conservative factors in the analysis include using zero structural damping and a crude model-updating procedure based on limited SMI data. These results demonstrate that the SMI test results, although used as a strong indication of possible stability issues in flight, are not definitive. The updated analysis showed a possible ASE problem in flight, but with no phase-related stability information for the multi-loop feedback configuration, this is also somewhat conservative. Linear and nonlinear system identification procedures are being investigated for deeper insight and understanding of the ASE dynamics.

Contact

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QUIET SPIKE™ FLIGHT-TESTING

Summary

The Gulfstream Aerospace Corporation (GAC) (Savannah, Georgia) partnered with the NASA Dryden Flight Research Center (DFRC) to execute the Quiet Spike™ (QS) project. Following a survey of potential test platforms, the NASA F-15B (McDonnell Douglas Corporation, St. Louis, Missouri, now The Boeing Company, Chicago, Illinois) airplane, tail number 836, was selected as the target test vehicle primarily because of its unique ability to carry a large-scale test apparatus to relevant supersonic flight speeds. The airplane radome was removed and a long, composite boom (spike) with a stationary extension and two moveable extensions were attached to the radar bulkhead so that the boom could be extended and retracted in flight (fig. 1).

EC06-0054-148

Figure 1. The NASA F-15B, tail number 836 with the attached Quiet Spike™.

Objective

The objective of this project was to prove that, while in flight, changing the shape of the front of an airplane prior to supersonic acceleration could reduce peak sonic boom amplitude. The project test system was also expected to partition the otherwise strong bow shock into a series of reduced-strength, noncoalescing shocklets.

Approach

The concept was to extend the airplane front-end prior to supersonic acceleration. This morphing would effectively lengthen the vehicle, reducing peak sonic boom amplitude, but is also expected to partition the otherwise strong bow shock into a series of reduced-strength, noncoalescing shocklets.

This combination of boom shaping techniques is predicted to transform the classic, high-impulse N-wave pattern typically generated by an aircraft traveling at supersonic speed into a signature more closely resembling a sinusoidal wave with a greatly reduced perceived loudness. ‘Quiet Spike™, is the GAC nomenclature for its recently patented front-end vehicle morphing arrangement.

The ability of QS to effectively shape a vehicle far-field sonic boom signature is highly dependent on the area distribution characteristics of the aircraft. The full aero-acoustic benefits of front-end morphing at far-field are only possible when the QS article and vehicle configuration are designed in consideration of each other. Adding QS technology to the airframe of an existing, non-boom-optimized supersonic vehicle is unlikely to result in an improved far-field signature because of the generally over-powering influence of wing- and inlet-generated shocks.

Therefore, it is generally recognized within NASA and the industry that a clean-sheet vehicle design is required to demonstrate the theoretically predicted far-field aero-acoustic benefits of QS-type morphing and other boom-mitigating concepts. The NASA Aeronautics Research Mission Directorate (ARMD) Supersonics Division has placed increased priority on near-term development and flight-testing of such a vehicle. To help achieve this objective, GAC believes that static and dynamic aerostructural proof-of-concept testing is a prudent step to take before a clean-sheet effort to reduce risk associated with a follow-on test program.

Aircraft 24 VDC power was supplied to operate motors used to extend and retract the QS. Aircraft 115 VAC power and 400 Hz power was converted and conditioned as required for use in the QS instrumentation system.

Mission critical (MC) research instrumentation included those required for measuring air data and aircraft state, QS static structural response, dynamic structural response, QS internal environment, and extend/retract system state. The DFRC installed instrumentation on the F-15B airplane (for example, strain gages and accelerometers) to ensure the structural safety of the aircraft.

Status

The airplane went through an envelope expansion flight program because effects of the spike being extended and retracted were unknown. Extension and retraction flight tests were conducted in order to measure QS operational loads, verify binding-free articulation, and measure the dynamic response of the spike at various flight-test conditions. The QS was extended early in the program to verify operation, then remained retracted and was flown on certain expansion envelope flights before flying the extended envelope points. The aircraft flew subsonic and supersonic flights out to Mach 1.8. A probing flight was conducted to measure near-field acoustics using another F-15 airplane.

The spike in the retracted position, measured at the front of the radome, was 14 ft and extended it measured 24 ft. The current for extending or retracting activity was in the range of 4.5–5.8 Amperes. The spike extension and retraction in flight took 21–22 seconds.

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STABILITY, CONTROL, AND HANDLING QUALITIES ANALYSIS OF THE F-15B QUIET SPIKE^{TM} AIRPLANE

Summary

The primary objective of the Quiet Spike™ (Gulfstream Aerospace, Savannah, Georgia) (QS) flight research program, figure 1, was the aerodynamic and structural proof-of-concept of a telescoping, half-scale, sonic-boom suppressing nose boom on a F-15B (McDonnell Douglas Corporation, St. Louis, Missouri, now The Boeing Company, Chicago, Illinois) airplane. The program goal was to collect flight data for these disciplines to 1.8 Mach. In the area of stability and controls, the primary objective was to assess the effect of the spike on the stability, controllability, and handling qualities of the airplane. The primary goal of this test philosophy was maintaining safety of flight. Flight-validated simulator predictions were used for envelope clearance to sequentially higher dynamic pressure, Mach number, angle of attack or sideslip.

EC-06-0184-10

Figure 1. F-15B Quiet Spike^{TM} with boom extended.

Objective

The primary concern was to assess aircraft stability and handling qualities throughout the subsonic, transonic, and supersonic flight regimes. There were three main issues: the uncertainty of the spike-influenced aerodynamics on the F-15B airplane flight dynamics; the F-15B airplane flight dynamics implications because of spike-induced air flow in the vicinity of air data and angle-of-attack sensors; and unfavorable effects caused by the spike during failure modes with a reduced flight envelope.

Approach

A series of aerodynamics stress cases were defined and analyzed in simulation for several different configurations and flight conditions. The stress cases varied aerodynamic uncertainties in worst-case directions in an attempt to excite a dynamic response that would reveal the maximum tolerable model uncertainties. Stability, handling qualities, and maneuver limit metrics were applied to the simulation data to evaluate the stress cases. Critical or potentially undesirable dynamics were identified for piloted simulation evaluations.

As a result of the stress analysis and pilot-in-the-loop simulation, regions of acceptable aerodynamic variations for key aerodynamic parameters were defined. Not only do these regions provide a measure of the robustness of the F-15B/QS configuration, they provided a means for flight-test clearance.

After each flight, three different parameter identification (PID) methods were used to identify and compare aerodynamic parameters against the predetermined acceptable aerodynamic uncertainty variations. As long as PID estimates and the trends that were projected to new flight-test clearance points stayed within the region of acceptable variation, those test points were cleared for testing.

Results

The preflight analysis results indicated that the QS would not cause any significant destabilization of the aircraft or degradation in controllability. This analysis was generally validated in flight. Most of the PID parameters estimated from flight data stayed within the regions of acceptable aerodynamic variations. Some deviations outside the bounds did occur for the damping derivatives *Cmq* and *Cnr* at some transonic and supersonic conditions. At these conditions, piloted evaluations of the simulation updated with the flight-estimated PID indicated undesirable, but controllable dynamics. The PID estimated static stability derivatives, *Cma* and *Cnb*, showed reasonable variations with respect to their boundaries. A rapid reduction in *Cnb* occurred in the 1.2–1.3 Mach region, but flattened out at higher Mach numbers as shown in figure 2. This method of clearance provided an efficient means of clearing the envelope to the program's goal of 1.8 Mach.

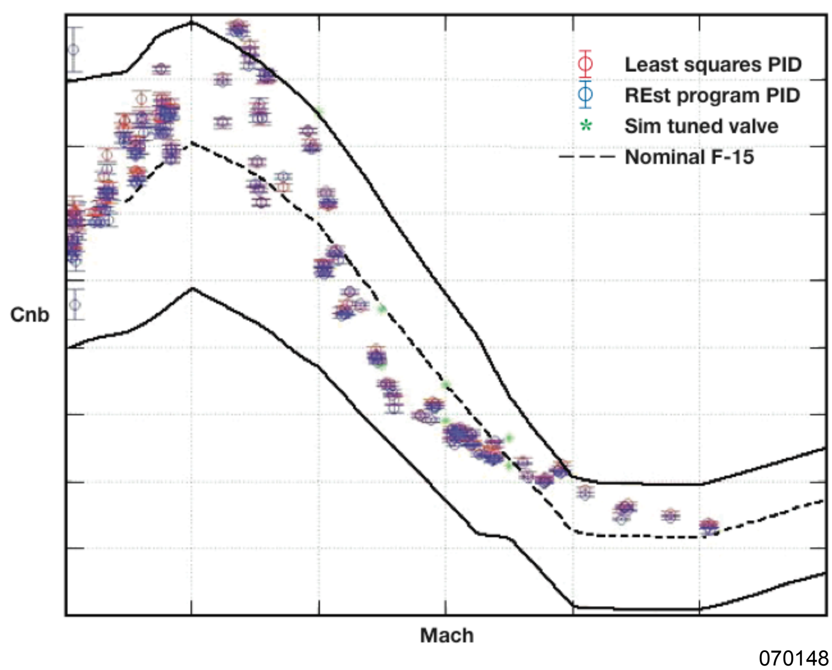

Figure 2. *Cnb* estimates for the transonic/supersonic regime compared to acceptable boundaries.

Status

Flight-testing was successfully completed to Mach 1.8.

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AUTONOMOUS AIRBORNE REFUELING DEMONSTRATION PHASE 1 TESTING

Summary

An autonomous system for hose-and-drogue air-to-air refueling was developed and flight-tested between May 2005 and August 2006. Ten flights were flown, culminating with the successful autonomous refueling engagement on the final flight on August 30, 2006.

Objective

The goal of the project was to develop and flight-test an autonomous airborne refueling demonstration (AARD) engagement using the Navy style hose-and-drogue air-to-air refueling method.

Approach

The prime contractor for this Defense Advanced Research Projects Agency (Arlington, Virginia) (DARPA)-sponsored program was Sierra Nevada Corporation (Sparks, Nevada) (SNC). The NASA Dryden Flight Research Center (DFRC) acted as the responsible test organization and provided the receiver test vehicle. The tanker airplane was contracted through Omega Aerial Refueling Services, Inc. (Alexandria, Virginia) and the optical tracking system, seen in figure 1 was contracted through OCTEC Ltd. (Bracknell, Berkshire, England).

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Figure 1. The OCTEC camera tracking system.

The two aircraft involved in the project were the Omega B-707 (The Boeing Company, Chicago, Illinois) tanker airplane and the NASA Dryden F/A-18B (McDonnell Douglas Corporation, St. Louis, Missouri, and Northrop Corporation, Newbury Park, California) systems research airplane. The tanker airplane used a standard refueling system and was unmodified except for a GPS antenna, a data-link antenna, and an instrumentation pallet mounted on the floor of the cabin. The sole purpose of this pallet was to measure and transmit tanker GPS/INS data to the receiver. On the receiver side, the F/A-18 airplane had a Research Flight Control System (RFCS), a custom AARD controller, a camera and image processor, and a pilot vehicle interface (PVI). The RFCS replicated the standard F/A-18 control laws and accepted analog inputs for the pilot control stick and throttle. The camera system and processor would track and report the tanker drogue position and velocity. The AARD controller executed the guidance and outer loop flight control laws, using GPS/INS data from both the receiver and tanker, along with the camera tracking data. Dual redundant analog voltages for pilot stick and throttle were output from the AARD controller and read by the RFCS through custom AMUX cards. The PVI was developed by DFRC to provide mode switching commands to the AARD controller, feedback to the aircrew, and management of the data telemetry systems.

Ten flights were flown, starting on June 15, 2006, and ending on August 30, 2006. Flight activities started using a Sabreliner (Sabreliner Corporation, St. Louis, Missouri) airplane from the National Test Pilot School (Mojave, California) as a surrogate tanker to perform the relative navigation testing of the system along with PID maneuvers. Once drogue tracking was necessary, flights were conducted with the Omega tanker airplane. Several flights were spent troubleshooting various systems and dealing with systems failures on the tanker or receiver. On the final flight, all systems were functional and several plug attempts were made. Of six attempts, the third and sixth were successful in plugging the drogue. All miss declarations resulted in controlled, predictable, and safe retreats from the drogue. Figure 2 shows the first capture attempt, resulting in a miss, and figure 3 shows a successful engagement.

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Figure 2. The extent of the miss on the first capture attempt.

Figure 3. The second successful autonomous refueling engagement.

Status

Phase 2 of the AARD program started in October 2006, with the purpose of developing a rendezvous system, improving controller performance, improving the optical tracker, and investigating capture of the drogue in a turn.

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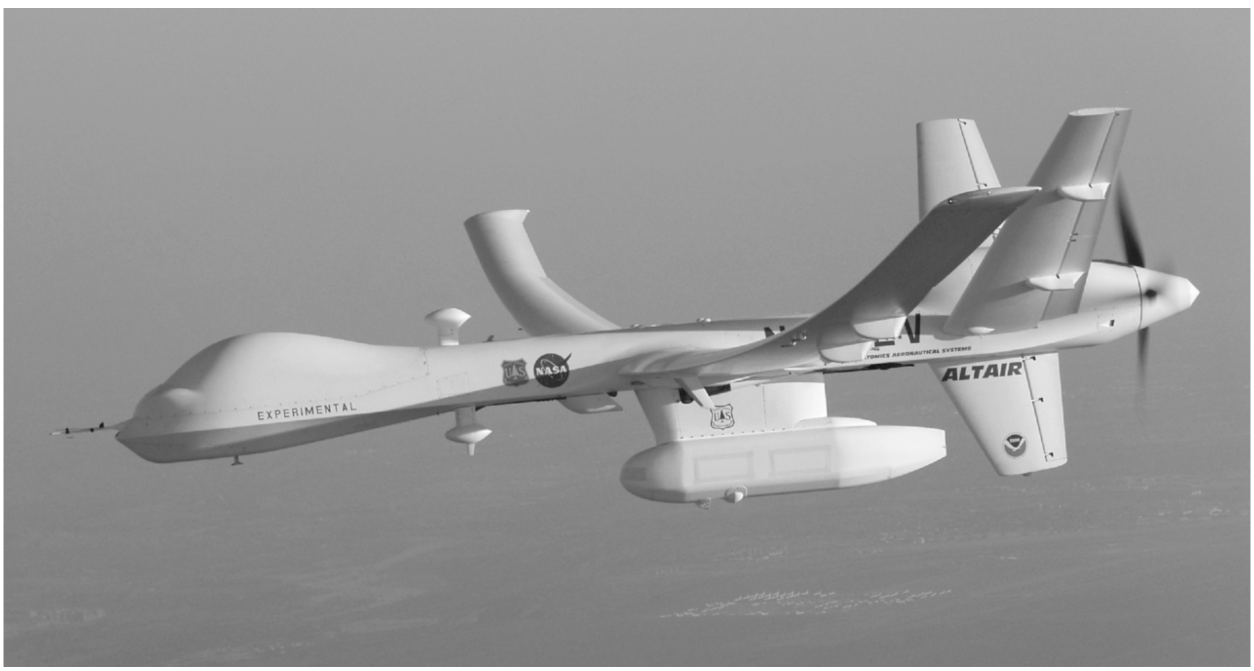
ALTAIR WESTERN STATES FIRE MISSION

Summary

The Altair Western States Fire Mission teamed NASA Dryden Flight Research Center (DFRC) (Edwards, California), NASA Ames Research Center (ARC) (Moffett Field, California) and General Atomics Aeronautical Systems Inc. (GA-ASI) (San Diego, California) with the USDA Forest Service to demonstrate the capability to use an Unmanned Air Vehicle (UAV) as a wildfire remote sensing platform. The experiment demonstrated the combined use of a NASA ARC-designed thermal multispectral scanner integrated on a large payload capacity UAV, a satellite image data telemetry system, near-real-time image georectification, and rapid Internet data dissemination to fire center and disaster managers. The flight demonstrations were conducted in September and October of 2006, as seen in figure 1. Two days after the research effort had been completed, the Governor of California requested emergency support for remote sensing over the Esperanza fire south of Beaumont, California. The entire team responded to the request to execute an extremely effective operational fire mission. This required completely reinstalling the fire pod and rewiring the Altair aircraft, planning the mission and requesting and receiving an emergency Certificate of Authorization (COA) from the Federal Aviation Administration (FAA) to operate in the National Airspace System (NAS.)



ED06-0208-1

Figure 1. Altair UAV with centerline-mounted fire pod flying over Edwards AFB, October 2006.

Objective

The primary objective was to achieve science flight(s) of 20–24 hours in duration, a period of which was required to be over an active fire with the fire sensors operating. The project also intended to demonstrate long-range remote operations within the NAS. The Altair Western States Fire Mission payload consisted of a NASA ARC multispectral scanner with channels in the visible, shortwave infrared (SWIR), and thermal infrared (TIR) spectrum, integrated into the fire sensor pod and mounted on the Altair aircraft.

Approach

The DFRC procured the sensor pod, also referenced as the "fire pod," from GA-ASI. Integration and testing of the multispectral scanner into the fire pod and integration of the fire pod onto the Altair unmanned aircraft was jointly performed by DFRC, ARC, and GA-ASI. Then, DFRC and GA-ASI planned, coordinated and conducted flight operations from the GA-ASI Gray Butte, California ground station. The ARC scientists collected images from the multispectral scanner in the Ground Control Station (GCS) and retransmitted these images to the Collaborative Decision Environment (CDE) at ARC via the internet. The images were processed with georectification software to overlay them precisely on a Google™ Earth (Google, Mountain View, California) global map as depicted in figure 2. These images could then be viewed anywhere in the world with a simple Internet connection to Google™ Earth and the appropriate Keyhole Markup Language (KML) file to link to the CDE. This allowed the on-site fire commanders at remote command centers to see near real-time images of the given fire. Additional atmospheric sensors were mounted in the Altair payload bay in an effort to collect atmospheric research data. These included the ARC atmospheric sensor "Argus," the National Oceanographic Atmospheric Administration (NOAA) gas chromatograph ozone (GC/OZ) photometer sensor, and the DFRC Research Environment for Vehicle-Embedded Analysis on Linux (REVEAL) box.

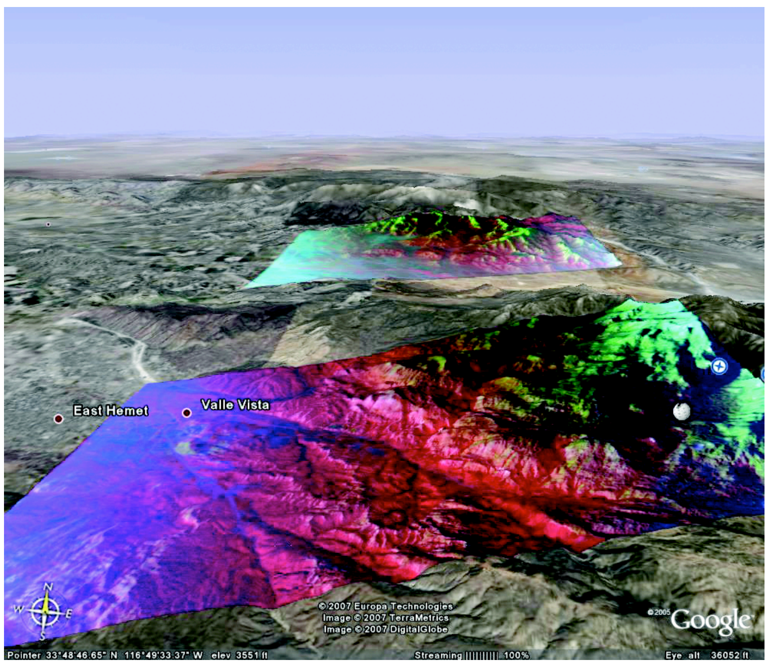


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Figure 2. Mult-spectral scanner images from Esperanza Fire, October 28, 2006. Overlaid on Google™ Earth Map.

This is the first time the Argus tunable diode laser spectrometer from the NASA Ames Research Center had flown on a UAV. Argus is relatively small (43 cm x 30 cm x 30 cm) and lightweight (21 kg) and flew in fully autonomous mode. It was configured to measure carbon monoxide (CO) during the Altair fire mission campaign. Argus not only gained valuable scientific data but also learned much about the health of the instrument on long duration high-altitude flights, which proved invaluable to the atmospheric scientists, Max Lowenstein and Jimena Lopez. The Argus team was able to generate CO vertical profiles from a maximum altitude of approximately 35,000 ft. This helps in understanding the dynamics of the atmosphere over a large vertical range. Additionally, and more importantly, Argus was able to provide satellite validation information for the NASA Aura satellite on four different occasions.

Conclusion

The Western States Fire Mission team successfully completed several major objectives including an endurance record for Altair by flying within the restricted range R-2508 for approximately 24 hours. The team initially flew Altair in the NAS under FAA authorization over Yosemite National Park and successfully collected fire images. While the project was not able to acquire the requested COA to fly long distances in the NAS, the project established a working relationship with the FAA and paved the way for future UAV missions in the NAS. The real success came at the end of the project when the project deployed Altair with the fire pod with only 24-hour notice at the request of Governor Schwarzenegger to the Esperanza fire. This demonstrated the capability to support an operational mission in the NAS with an unmanned air vehicle.

Status

NASA Dryden has procured a Predator B (General Atomics Aeronautical Systems, Inc., San Diego, California) aircraft called “Ikhana.” The Ikhana platform will perform a combination of aeronautical research and airborne Earth science research. The fire pod will be integrated with a new mount onto the in-board wing station on the Ikhana aircraft. The Western States Fire Mission team will attempt to complete fire mission objectives that include long distance deployment and remote operation to support fire missions in Montana, Idaho, and Washington State.

Contacts

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OPTIMAL USE OF VERTICAL GUSTS

Summary

Equations for determining vertical wind velocity and vertical wind shear were developed for the purpose of atmospheric energy harvesting. The vertical wind state was calculated from surface positions, body axis accelerations and rates, Euler angles, and GPS position. The estimated velocity was used to determine optimal pitching maneuvers for one of three objectives during straight-line flight in the presence of headwind or tailwind. The selected objectives are: minimal energy consumption, minimal energy consumption while maintaining arrival time, and maximum cross-country speed. The equations used for cross-country pitching maneuvers were taken from sailplane cross-country racing theory. This approach has been tested in simulation with good results.

Objective

The objective of this study was to develop algorithms needed for an aircraft to extend its performance during point-to-point flight through vertical gusts.

Approach

Vertical wind velocity was determined by finding the difference between the inertial velocity and the air-relative velocity of the aircraft. Inertial velocity was found by blending the accelerometer measurements and the GPS position measurements with a complementary filter. Air-relative velocity was first determined by estimating angle of attack and sideslip angle. The normal and lateral force equations were solved for angle of attack and sideslip angle to give a reasonable estimate of the wind angles. Air-relative velocity was found by transforming the measured true airspeed into the inertial reference frame. Vertical wind shear was determined by differentiating the vertical wind equation and neglecting small terms. The comparison of true vertical wind to estimated vertical wind during simulated flight is shown in figure 1.

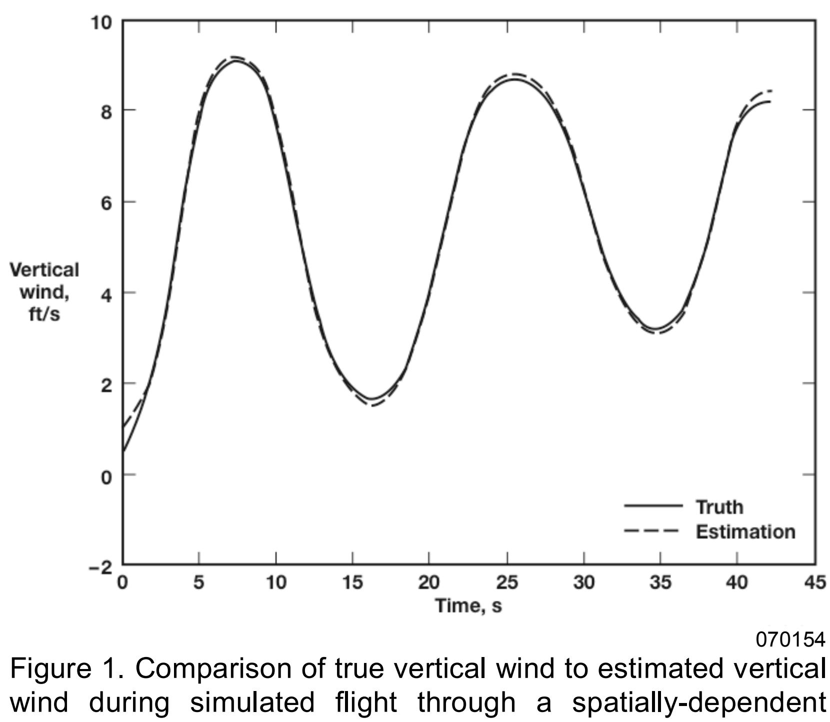

Figure 1. Comparison of true vertical wind to estimated vertical wind during simulated flight through a spatially-dependent vertical wind field.

Vertical wind was used to determine the optimal speed for a glider flying in a moving air mass to maximize one of three objectives. The selected objectives are minimal energy consumption, minimal energy consumption while maintaining arrival time, and maximum cross-country speed. The speed-to-fly theory for glider pilots was used to determine the optimal speed for minimal energy usage or maximum cross-country speed in the presence of vertical air currents and headwind or tailwind. The speed to fly for minimal energy consumption while maintaining arrival time was determined by comparing the height gain and speed loss of a particular airspeed with the height required to catch up to an imaginary aircraft that maintained the original cruise speed. Figure 2 shows preliminary results of the speed-to-fly algorithms as implemented on a Piccolo (Cloud Cap Technology, Inc., Hood River, Oregon) flight computer and running in a hardware-in-the-loop simulation. The NASA Dryden Flight Research Center thermal model was used for these tests. The unmodified aircraft response is shown in black, the minimal energy mode is shown in red, and the maximum cross-country speed mode is shown in green. The maximum cross-country speed mode assumes gliding flight and thus did not produce a speed gain over the baseline flight path. Future versions of these algorithms will account for engine effects.

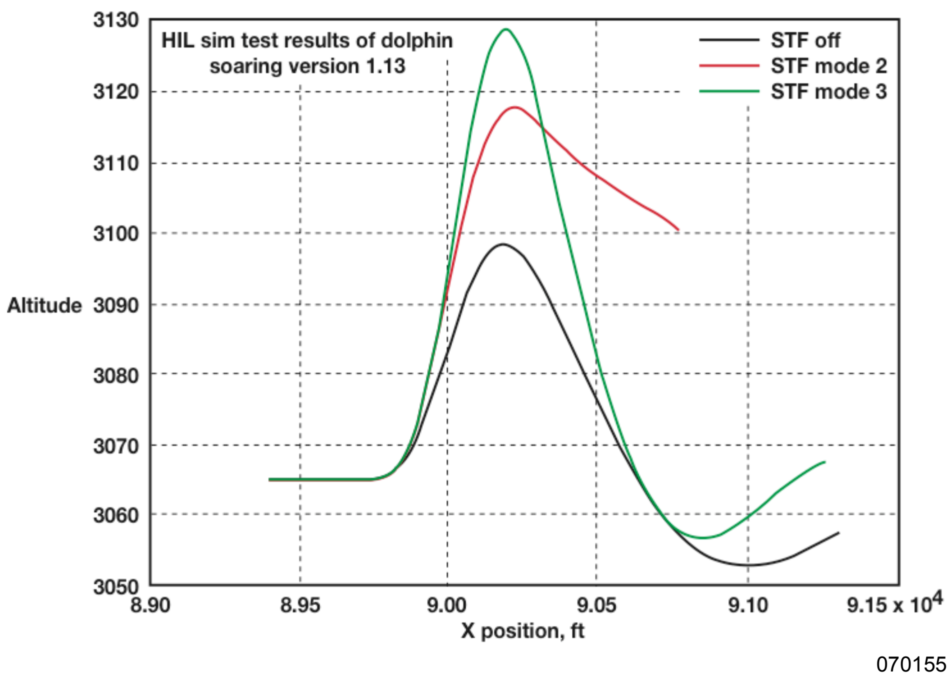

Figure 2. Preliminary speed-to-fly algorithm results.

Status

This work was concluded in September of 2006.

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AERONAUTICS RESEARCH MISSION DIRECTORATE HYPERSONICS MATERIAL AND STRUCTURES: CARBON-SILICON CARBIDE RUDDERVATOR SUBCOMPONENT TEST AND ANALYSIS TASK

Summary

In fiscal year 2006, planning was initiated for the NASA Aeronautics Research Mission Directorate (ARMD) Fundamental Aeronautics Program. As part of the ARMD planning efforts, NASA personnel proposed using the Carbon-Silicon Carbide (C/SiC) Ruddervator Subcomponent Test Article (RSTA) as a test structure to support research objectives within the Hypersonics Materials & Structures (M&S) program. By the end of fiscal year 2006, the RSTA test effort was established and incorporated into the Hypersonic M&S program.

Objective

The C/SiC RSTA is a hot-structure control surface that was designed, fabricated, but never tested under the X-37 long-duration orbital vehicle technology development program, seen in figure 1. The RSTA was designed by Materials Research & Design, Inc. (MR&D) of Wayne, Pennsylvania, and manufactured by GE Power System Composites (GE PSC) of Newark, Delaware. The RSTA is a truncated version of the full-scale X-37 control surface but it incorporates all of the major full-scale features, including the metallic spindle, five major C/SiC quasi isotropic lay-up components fastened together with mostly C/SiC fasteners, and face-sheets which serve as access panels for assembly of the RSTA.

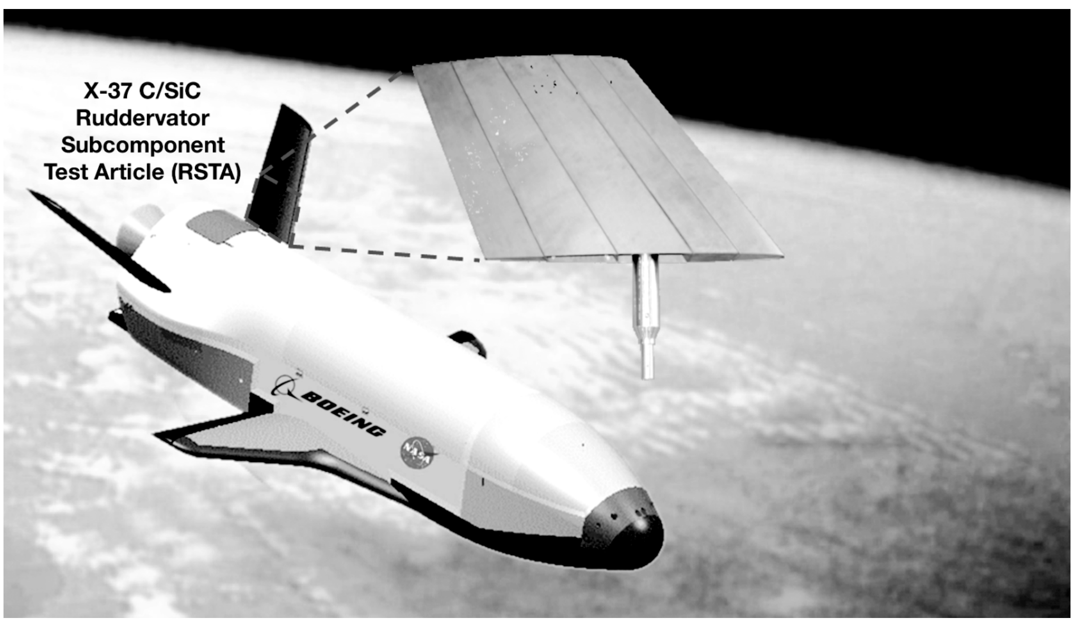

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Figure 1. Location of the RSTA on the X-37.

The research objectives that were proposed to the Hypersonics M&S program were as follows:

- To test and evaluate the thermal, structural, and dynamic performance of the C/SiC RSTA through the application of relevant hypersonic thermal, structural, acoustic, and vibration loads.

- Establish an extensive database for current and future structural analysis developments and evaluation.
- Perform pre- and post-test thermal-structural analysis to support test operations and evaluation of NASA subsequent advanced analysis methods.

Approach

A four-phase test program has been developed to subject the C/SiC RSTA to relevant thermal, structural, acoustic, and vibration loads expected for hypersonic reentry and transatmospheric flight trajectories. Details of the four test phases and test location are provided as follows:

- Test Phase 1: Acoustic and vibration testing to X-37 ascent conditions. (Testing to be conducted at NASA Langley.)
- Test Phase 2: Thermal and combined thermal/structural loading to X-37 reentry conditions and transatmospheric load conditions. (Testing to be conducted at NASA Dryden.)
- Test Phase 3: Room-temperature mechanical proof loading to X-37 design conditions. (Testing to be conducted at NASA Dryden.)
- Test Phase 4: Vibration and thermal/acoustic loading to transatmospheric load conditions. (Testing to be conducted at NASA Langley.)

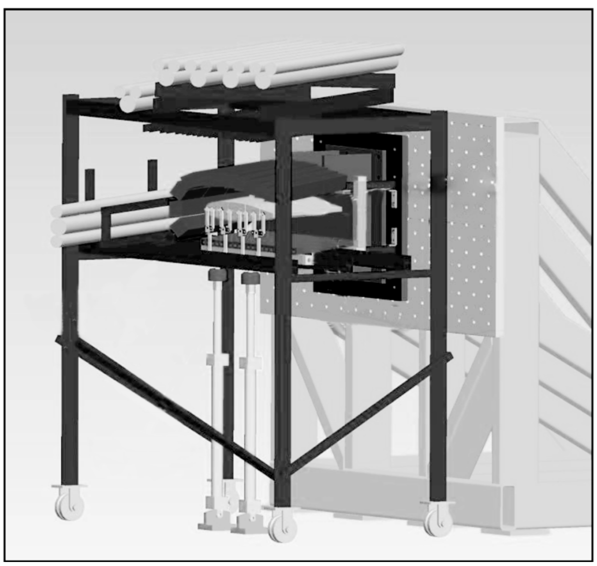
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Figure 2. RSTA Phase 2 test setup.

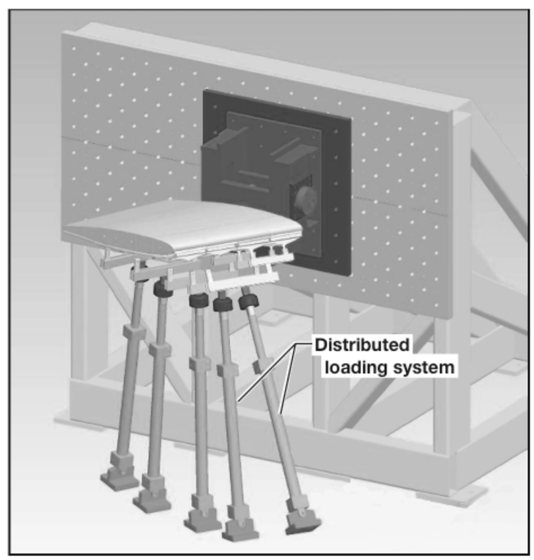

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Figure 3. RSTA Phase 3 test setup.

Status

The RSTA test program was incorporated into the ARMD Hypersonics M&S implementation plan and was initiated in October 2006.

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DESIGN OF THE PHOENIX MISSLE HYPERSONIC TESTBED (PMHT)

Summary

Following the success of the Hyper-X project, conclusion of the X-43 Program, and the retirement of the NASA B-52 (The Boeing Company, Chicago, Illinois) airplane, tail number 008, researchers at the NASA Dryden Flight Research Center began to look for another research test vehicle, or testbed, for the hypersonic flight regime. A new hypersonic testbed would have to be low cost to survive in the financial environment of shrinking aeronautics funding within NASA. A low-cost approach to develop a hypersonic testbed was found in a surplus of U.S. Navy Phoenix (Raytheon Company, Waltham, Massachusetts) air-to-air missiles in combination with the NASA F-15B (McDonnell Douglas Corporation, St. Louis, Missouri) airplane, tail number 836. Currently, the only other option for hypersonic flight-testing is through the use of sounding rockets that are unguided and limited in their trajectory variability. Also, sounding rockets cannot be used to verify control system algorithms because they lack a guidance system. As shown in figure 1, the F-15B airplane will be used to carry the missiles to supersonic conditions for launch. The missiles would then be launched, carrying experimental research payloads to their test conditions at speeds in excess of Mach 5. This effort is being performed with the assistance of the Naval Air Warfare Center Weapons Division (NAWC-WD) at China Lake, California.

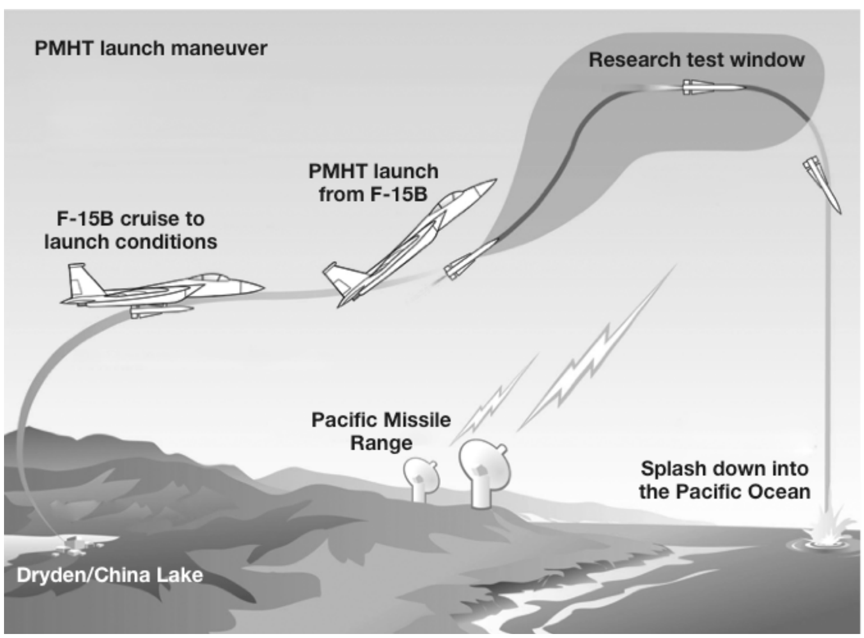

Figure 1. Operational model for the PMHT.

The Phoenix missiles used for hypersonic research will have their explosive warheads removed and their tracking and guidance systems replaced with a smaller, more lightweight guidance system. The missiles will also be heavily instrumented to obtain and transmit test data from experiments in such areas as thermal protection materials, scramjet propulsion, guidance and control, boundary layer transition, and aerodynamics at hypersonic speeds.

Objectives

The primary research need is a low-cost, hypersonic, research flight-test capability to increase the amount of hypersonic flight data that will help bridge the large developmental gap between ground testing, analysis, and major flight demonstrator X-planes.

The goals of the Phoenix missile hypersonic testbed (PHMT) effort are to develop an air-launched missile booster research testbed to accurately deliver research payloads through programmable guidance to hypersonic test conditions at low cost and with a high flight rate.

Before launching research payloads with the PMHT, the development of the testbed is the primary research objective. The ground and flight research objectives are:

1. A payload capacity of 5.5 ft^3.
2. To exceed (with different trajectories): Mach 5 with at least 500 psf dynamic pressure or dynamic pressure of 2000 psf with at least Mach 3
3. A unit test cost under $500,000
4. A minimum of 2 test flights per year
5. To utilize surplus air-launched missiles and NASA aircraft

Approach

The development of the PMHT will follow the NASA systems engineering process. Following the feasibility study and the mission concept design review, system requirements will be compiled and the preliminary design will commence. Following the preliminary design review, the detailed critical design will commence, ending with the critical design review.

Concurrent with the design process, a series of F-15B airplane flights will occur with a missile carried captive; the data from which will be used to validate the use of F-15 simulation in the design process and to evaluate aircraft handling qualities. These flights are critical to imbue the design process with actual flight test data.

Figure 2 shows the concept of the modified missile. The missile warhead and tracker will be removed to provide payload space. Every attempt will be made to keep hardware from the tactical round to reduce cost and redesign; however, some of the missile systems will be replaced with smaller systems, including the guidance computer, to provide additional payload volume and added functionality. Also, some systems need to be added such as a flight termination system (FTS) and telemetry. The details of the systems configuration will be determined during the preliminary design phase. A launch control computer will be designed to interface with the aircraft and the new missile systems.

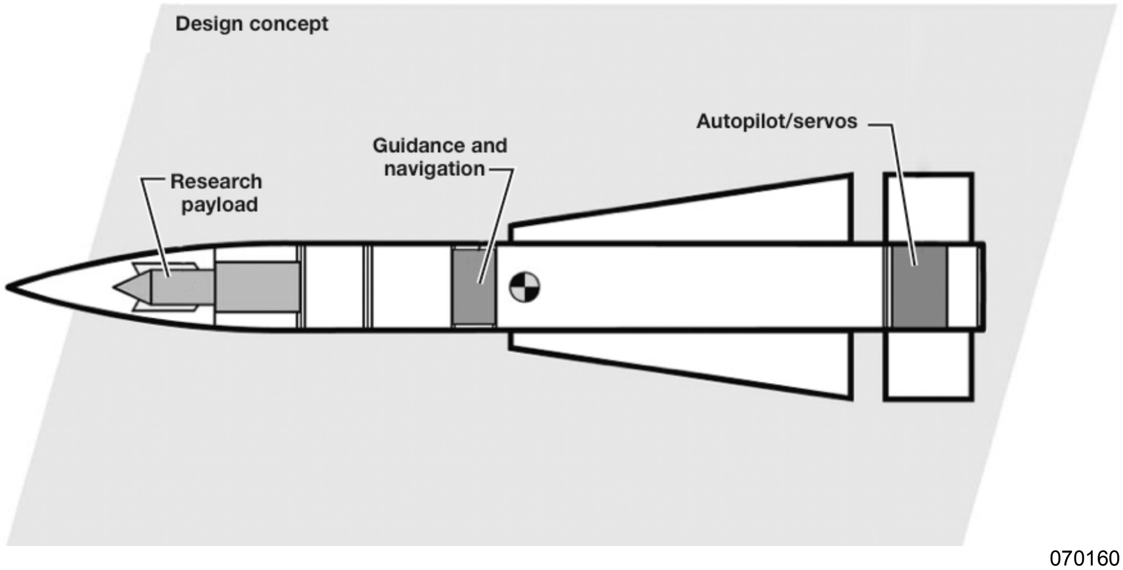

Figure 2. Modified missile design concept.

Concurrent with and following the design phase, integration and verification and validation will begin. Ground test hardware will be constructed to test the systems design. Ground tests will include hardware-in-the-loop tests, aircraft-in-the-loop tests, ground vibration tests, electromagnetic interference tests, and others as necessary. Captive flight tests will be performed to check out the systems in the air, and a drop test will be performed before a live launch.

Status

At present, NASA is performing design, analysis, and testing leading up to a critical design review at the end of 2007. As part of the preliminary studies, several captive-carry flights will be flown by the F-15B airplane with an inactive missile containing no propellant carried on the aircraft centerline pylon to determine the performance of the aircraft when carrying the missile. The project has not been funded to conduct actual launches to obtain hypersonic flight-test data, and a decision on funding such research is not expected before 2008.

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14. ABSTRACT
Selected research and technology activities at Dryden Flight Research Center are summarized. These activities exemplify the Center's varied and productive research efforts.

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