NASA/TM-2006-213687

2005 Engineering Annual Report

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

December 2006

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FULL FIELD THERMAL PROTECTION SYSTEM HEALTH MONITORING SYSTEM FOR CREW EXPLORATION VEHICLES

Summary

The thermal protection system (TPS) of a space vehicle is a very critical system, as the tragic Space Shuttle Columbia accident highlighted. Currently there is no system to monitor the health of a TPS. The instrumentation in use on flight vehicles today consists of traditional sensor systems: thermocouples, strain gages, pressure transducers, and a few others. This current technology in sensor systems is all far too heavy to consider for use in a full-field health monitoring system. Fiber optic sensors (specifically fiber Bragg Grating (FBG) sensors) are extremely lightweight, however, and have the capability of multiplexing many sensors onto one fiber, therefore minimizing system weight and complexity.

Objective

The objective of this research is to develop a prototype TPS Health Monitoring System (HMS) for a crew exploration vehicle (CEV) or equivalent flight vehicle with insulated structures (for example, one that uses some form of a TPS). In addition to the sensor and system development effort, an algorithm will be developed to interpret the data and determine TPS health.

Approach

The approach taken in 2005 was to first develop a validated model and test setup before validating the sensor system and the necessary data for algorithm development efforts.

Discussion

The test setup was successfully tested for heating uniformity, controllability, and repeatability. The initial model was completed and was found to match initial test data with the exception of a lag, seen in fig. 2. Inconsistent model and test data can indicate an incomplete understanding of the physics present in the test setup, so the decision was made to investigate both the model and the test setup.

The model investigation began by simplifying the model into a monolith to examine the effects of varying the boundary conditions and adding layers of materials. A material property perturbation study of ± 5 percent was then performed, followed by a study on the effects of material property thermal variation. The study revealed no effect on thermal variation, so, the boundary conditions were refined and a performance envelope study (using 10x conductivity, and one tenth specific heat, etc.) was performed. At this point, the response lag was still not affected, so the focus was turned towards computational aspects.

Two different software packages were used with similar results. Finite Difference and Finite Element solutions (using the same mesh density) were compared. A two-dimensional model was made to determine if the dimensionality of the three-dimensional column of elements was causing a problem. Modifying the model did not resolve the problem, so a mesh refinement study was performed. As the mesh density was increased, the solution diverged to a zero-response. It was at this point that the error was reported to the software, Patran, and the developer determined that there was a program error and the Finite Difference implementation was not functioning properly. The Finite Element scheme was then used in a similar mesh refinement study and the solution converged.

Having thus proven that the computational scheme was functioning, the next step was to determine if there were physics that were not being modeled since the model still lagged behind experiment. The material in the test article is an anisotropic form of fused silica with non-uniform and incomplete material property data. The decision was made to make a test article with well-known material properties in order to isolate whether inaccurately modeled physics were in the test setup itself or if they were within the material of the test article.

In addition to the model investigation, the test setup was closely examined. In an effort to find the source of the lag between the model and test data, many things were varied: test article installation methods, insulation types and placement, emissivity coatings, thermocouple (TC)/FBG placement, test article structure, heat source, data acquisition system, and test article material.

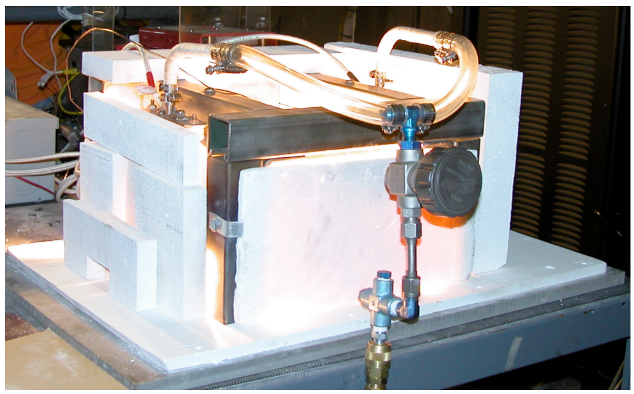

Photo courtesy of Chris Kostyk

Figure 1. Test setup during quartz lamp firing.

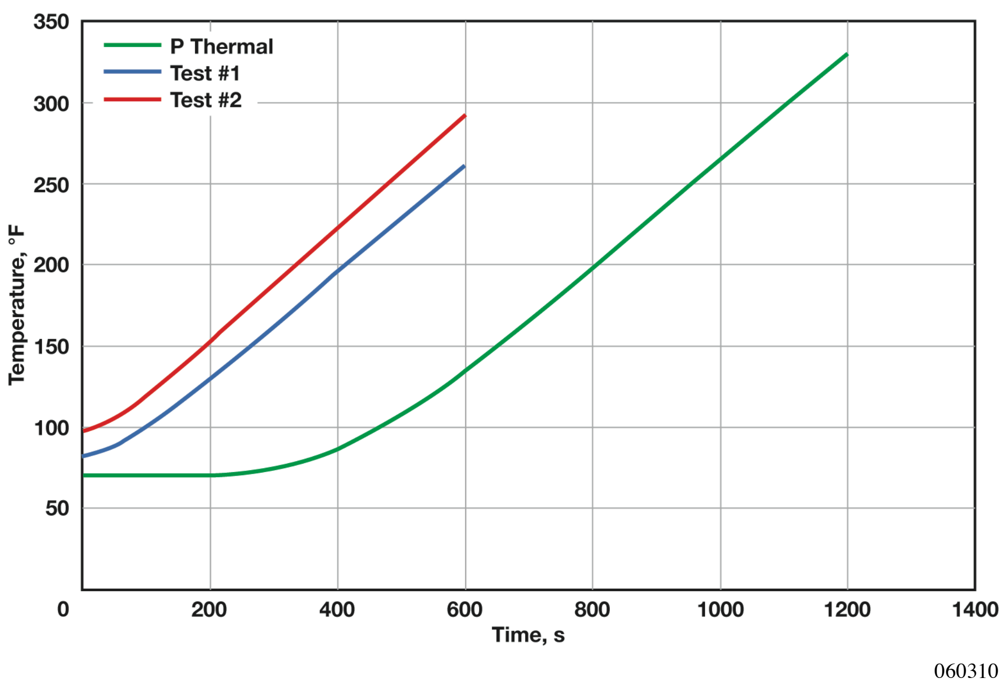

Figure 2. Test data compared with model results – same slope but significant response lag difference.

Status

The test article with well-known material properties (Titanium monolith) and the corresponding model are in the process of being completed. A test article to investigate the veracity of the embedded FBG measurement (against collocated TCs) is also in the fabrication process.

Contact

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C-20A PRECISION AUTOPILOT DEVELOPMENT

Summary

A precision autopilot capability is being developed for the NASA C-20A (Gulfstream Aerospace, Savannah, Georgia) airplane as a part of the agency's Unmanned Aerial Vehicle Synthetic Aperture Radar (UAVSAR) program. The NASA UAVSAR program is developing a Synthetic Aperture Radar that fits within a pod that will be mounted underneath the forward fuselage of the C-20A airplane. The precision autopilot interfaces with the C-20A through the Instrument Landing System (ILS). This approach makes use of the accuracy and safeguards inherent in the autopilot to fly the precision trajectory. This precision autopilot capability is currently in development and will enter flight testing during fall of 2006.

Objective

The precision autopilot will enable repeat pass flights within a 10-m tube for interferometric applications of the Synthetic Aperture Radar (SAR) being developed for the UAVSAR program. Flight lines are expected to be up to 200 km in length. The precision autopilot 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 precision autopilot algorithms are hosted on an onboard precision autopilot computer which provides continuous ILS correction signals directly to the onboard navigation receiver, bypassing the C-20A ILS antennas. Two ILS tester units have been modified to receive the commands from the precision autopilot computer and interface with the onboard navigation receivers. The correction signals allow the C-20A autopilot to execute a simulated ILS approach that meets the requirements for SAR operations. The precision autopilot generates a real-time position solution using information from the airplane and a near real-time differential GPS unit located in the UAVSAR pod.

The precision autopilot 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 is developing a C-20A engineering simulation for development of the precision autopilot. A Monte Carlo capability has been developed in conjunction with the C-20A simulation to examine the precision autopilot performance in the presence of vehicle and atmospheric uncertainties. The Monte Carlo analysis consists of randomly perturbing simulation parameters within specified bounds. A total of 44 simulation parameters are perturbed as a part of the analysis, including aerodynamics, mass properties, system timing, and winds. Figure 1 shows the 10-m tube precision autopilot tracking performance during a 500-run Monte Carlo analysis. The data shown is for approximately the first 4 min after the C-20A has entered the 10-m tube. The precision autopilot meets the performance requirements in the simulation environment.

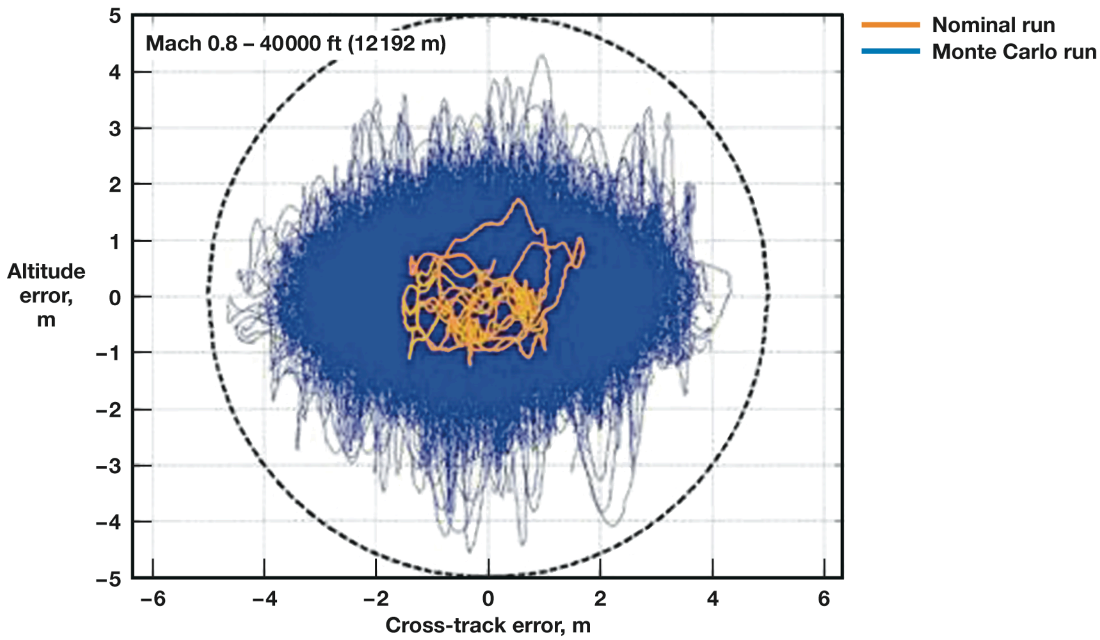

Figure 1. The C-20A precision autopilot 10-m tube tracking performance.

Status

The precision autopilot development program completed a critical design review in April 2006, and will transition from design to verification and validation testing during the summer of 2006. Flight testing of the precision autopilot will commence in the fall of 2006 with a demonstration flight of the SAR planned for late 2006.

Reference

Soren Norvang Madsen, Niels Skou, Johan Granholm, Kim Wildt Woelders, and Erik Lintz Christensen, *A System for Airborne SAR Interferometry*, AEU International Journal of Electronics and Communications, 50(1996) No. 2, pp. 106–111.

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CONTROLLING A 757-200 AIRPLANE WITH THROTTLES ONLY

Summary

In mid-2005, the Department of Homeland Security (DHS) sponsored an initiative to extend throttles-only control (TOC) techniques to current commercial fleet aircraft. This initiative involves both United Airlines (UAL) and the NASA Dryden Flight Research Center (DFRC). It concentrates on the development of piloting techniques for alternate operation of an aircraft, should it become disabled in flight. Named Propulsion-Controlled Aircraft Recovery (PCAR), this project utilizes UAL assets, such as pilots, simulators and aircraft, and NASA engineers to conduct research on the use of TOC as an alternative method of aircraft control. Control using throttles only involves utilizing the thrust generated by the engines to control the aircraft in all three axes of flight, and can be used to supplement or replace degraded flight controls.

Objective

The objective of the project is to generate a set of pilot guidelines for the operation of a specific aircraft without hydraulics that a) have been validated in both flight and simulation by relevant personnel, and b) mesh well with existing commercial operations, maintenance, and training at a minimum cost. The anticipated outcome of the PCAR project is a set of pilot guidelines for TOC operation of a 757-200 (The Boeing Company, Renton, Washington) airplane without conventional controls, to DHS.

Approach

The method of TOC is nothing new to the aviation world. Control with throttles has been used numerous times in the past after in-flight failures have left an aircraft with unusable control cables or a loss of hydraulic fluid and pressure. Perhaps the most recognized TOC event is UAL flight 232, which crash-landed in Sioux City, Iowa in July 1989, after an uncontained in-flight failure of the number two engine. The beauty of TOC is that it depends on piloting technique only, and no modifications, software, or additional maintenance for the airframes are necessary. This makes TOC very cost-effective, which is good news for the industry.

Research data from both simulation and flight has shown that TOC is an effective means of control on the 757-200 airplane. The airplane is very responsive to TOC, and the response is mostly intuitive. As part of TOC research, NASA is working with pilots from UAL to help assess the effectiveness of TOC procedure development. The evaluation pilots include both UAL test pilots and UAL line pilots. These line pilots are qualified on both the 757 and 767 airplanes, and have varied backgrounds and flight experience. In addition to the 70+ hours the PCAR group has spent in the simulator, some time has been spent validating the TOC procedures during flight. Overall, the pilots reported that the aircraft responded slightly better to TOC in flight than in the simulation. More results are expected as the PCAR team continues to analyze the flight data.

Status

As of June 2006, the PCAR team has 52 hours of 757-200 simulation experience, about 3 research hours in flight, and 2 pilot simulation evaluations. It is expected that an additional 5 hours of flight time, 40 more hours of simulation time and additional pilot evaluations will be completed by October 2006.

Contact

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A BRIEF SUMMARY OF THE 2005 PATHFINDER+ AEROELASTIC/TURBULENCE FLIGHT TESTS

Summary

In October 2004, rainfall flooded Rogers Dry Lake in Edwards, California and rendered all runways unusable, delaying flight testing of the Pathfinder+ solar-powered Unmanned Aerial Vehicle. In August 2005, the lakebed with associated runways and the AeroVironment/NASA team was ready to support flight tests. The first flight occurred on August 31, 2005, under clear skies and light winds. The aircraft did encounter some wind shear turbulence at approximately 1200 ft. The more intense turbulence, caused by thermal activity, was encountered at lower altitudes near the end of its two and a half hour flight. The second flight test occurred on September 14, 2005, under mostly clear skies and some light surface breezes. The second flight encountered a higher frequency of wind shear, or mechanical turbulence as compared to the first flight. During this flight, wind speeds were highest below the temperature inversion, which resulted in more mechanical turbulence at a much lower altitude than normal. The duration of the second flight was one hour and eight minutes. This second flight was the final flight of the flight test series and the final flight of the Pathfinder+ aircraft. The Pathfinder+ is scheduled for retirement and to be put on display at the Smithsonian Air and Space Museum in 2006.

EC05-0186-06

Figure 1. Pathfinder+ just after takeoff.

EC05-0201-08

Figure 2. Pathfinder+ and "Turbo booms" in-flight.

Objective

The primary objective of this series of flights was to develop aeroelastic models for the future design of a Pathfinder/Helios-class vehicle. Developing transfer functions between atmospheric perturbations and the physical response of the aircraft to those perturbations is part of the modeling effort.

Approach

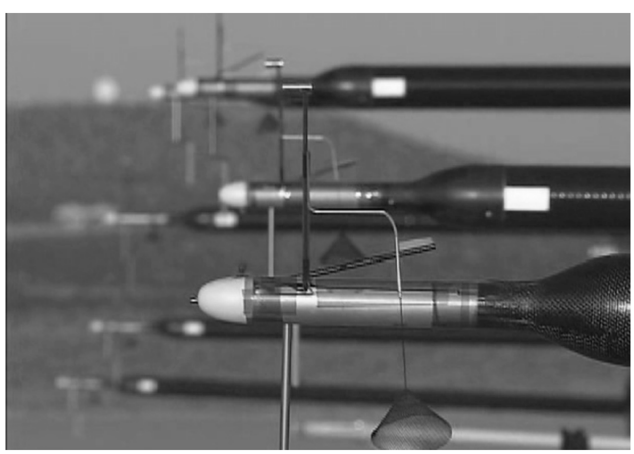

Photo courtesy of Lori Losey

Figure 3. Seven ATMS "Turbo boom" sensors system (alpha, beta, trailing cone, temperature, and Keil probes).

Photo courtesy of Lori Losey

Figure 4. Pathfinder+ with normal dihedral.

Photo courtesy of Lori Losey

Figure 5. Pathfinder+ wing deflection due to turbulence.

The plan was to fly early in the morning, just after sunrise, to avoid thermal activity and search for wind shear turbulence. This would allow the pilot to fly into and out of turbulence at a given altitude or geographic location with some fashion of repeatability and predictability. When it was determined that Pathfinder+ was flying in sufficient turbulence, the aircraft was commanded to fly straight, level, and at a selected airspeed (26–28 ft/s) with no inputs from the pilot. The sensors on the Atmospheric Turbulence Measurement System (ATMS) booms, or "turbo booms," would measure wind gusts ahead of the aircraft position. In addition to the ATMS, strain gages and accelerometers mounted on selected locations on the booms, spar, and trailing edges would measure the structural responses as the aircraft flew into the turbulence (figs. 4 and 5). The data was recorded onboard and a subset of the data was downlinked in real time to flight engineers located inside the Ground Control System (GCS). After the flight tests, dynamics engineers from AeroVironment will review all the data from all the flights and deliver the data to the NASA Dryden Flight Research Center. Initial reviews of the data from the two flight tests seem to indicate that most of the data from the ATMS and strain gage system was valid.

Status

Following the flight tests, engineers at AeroVironment are parsing through the data for delivery to Dryden. This expected to continue through 2005 and into 2006. The Pathfinder+ will be prepped for its delivery to the Smithsonian Air and Space Museum for display. This will involve such tasks as deactivating the solar array and some cosmetic work. The flight test was considered a success and a NASA Level I milestone was satisfied.

Contact

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SIXTY-THOUSAND-POUND CAPACITY STARR SOFT SUPPORT (60K3S)

Summary

A new 60 000-lb capacity Starr Soft Support (60K3S) has been in the making for 3 years and is finally in use. This innovative design can be used for weight and balance measurements, complete inertia tensor measurements, ground vibration tests (GVT), control surface free-play tests and Structural Mode Interaction (SMI) tests using a single basic setup. The 60K3S allows aircraft to be tested in a free-free environment to simulate in-flight boundary conditions. The 60K3S also eliminates the need for engineers to spend weeks trying to model boundary conditions of aircraft sitting on the ground with the nonlinearities of strut and tire stiffness.

Objective

The 60K3S was designed to eliminate the need for a critical lift on one-of-a-kind aircraft while accommodating aircraft weighing up to 60 000 lb, holding a maximum of 20 000 lb at each jacking point (for example, on the F-15, F-18, and G-III as seen in figs. 1–3). This system can be transported from building to building using a forklift and can be placed at the jacking points of the aircraft using a pallet jack. To use the system, a controller cart powers all three motors simultaneously, which extend the electric actuators within a few inches of the aircraft jacking points. Final lateral adjustments of the 60K3S are made using the pallet jacks. After the pallet jacks are removed, the electric actuators continue to extend and raise the vehicle above the ground until the landing gear are clear to retract. The electric actuators are stopped, the gear are retracted, and the actuators lower the aircraft to an agreed-upon working height. At this time, the isolators can be inflated.

Photo courtesy of Tom Tschida

Figure 1. Gulfstream-III ground vibration test.

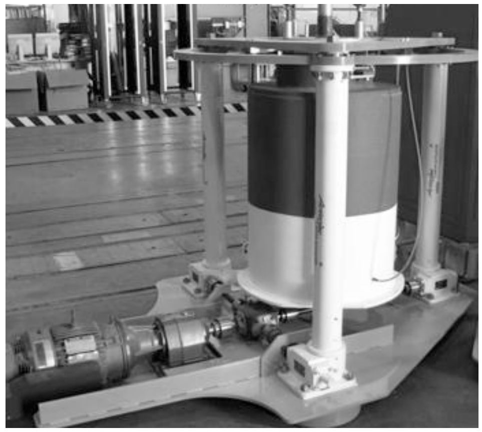

Photo courtesy of Starr Ginn

Figure 2. One of three assemblies of the 60K3S.

Approach

Before the 60K3S soft supports could be used for GVTs, they had to pass proof and load tests.

60K3S Features

Each soft support of the 60K3S weighs approximately 900 lb and is easily maneuverable using a pallet jack. On top of the base plate is an explosion-proof motor (3 HP and 1730 RPM), which allows it to be operated near fueled aircraft in an enclosed hangar. In addition, if the motor is powered down because of an electrical failure, the brake, which has a torque capacity of 15 ft-lb, will be applied and the equipment linked to the motor will be held in place. Connected to the motor is a 10:1 gear reduction to satisfy the requirement of raising the electric actuators at a rate of 3.6 in/min. A custom- made miter box with a rated torque of 1094 in-lb at 900 RPM, allows one input and three outputs. The electric actuators use acme/machine screws, which raise and lower loads such that if power is lost, the actuators will not back-drive. The front actuators of each of the three assemblies have a rotary limit switch, which restricts the actuators from over-extending or back-driving. The isolator is an inflation device, which allows the aircraft to float on air (specifically nitrogen), and simulates a free-free environment.

Proof Test

A very successful proof test was performed in December 2005 for the 60K3S configuration using two diaphragms in each isolator and the short electric actuators (25-in stroke versus the tall electric actuators with a 40-in stroke). Five tests concluded with no anomalies, but a redesign of the pneumatic controller. The first test was a vertical, static-proof test (no inflation) to 150 percent of the rated load (36 000 lb or 108 000 lb total). The design of the loading structure took some very creative engineering since it had to be rated for 324 000 lb so it could load all three soft supports at the same time. The second test was used to determine the side load capacity (no inflation) and followed the aircraft jack specification of being fully extended with 100-percent rated vertical load (24 000 lb) and 150-percent rated side load which was determined to be 4000 lb. The third test was to extend and retract the electric actuators at 125-percent rated vertical load (30 000 lb). The fourth test was to inflate the isolator with less than 150 psi and 100-percent rated diaphragm load of 20 000 lb. During this test we found that the pneumatic controller was limited to 120 psi for the previous isolator configuration of one diaphragm. A new pneumatic controller was quickly designed and fabricated by the Loads Lab Crew and the test was repeated successfully.

Status

The configuration that was proof tested used the short actuators with two diaphragms. Each soft support of the 60K3S is currently in use and has the following working loads: 24 000 lb static with no inflation, 2666 lb total side load vector with no inflation, 24 000 lb for electric actuation, 20 000 lb and less than 150 psi for isolator inflation.

Photo courtesy of Tom Tschida

Figure 3. The F-15B ground vibration test.

Contact

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LOADS CLEARANCE ON A MODIFIED, PREPRODUCTION F-15 AIRPLANE WITH INTELLIGENT FLIGHT CONTROLS

Summary

The F-15 Intelligent Fight Control Systems (IFCS) Gen-2 project objective includes flight test of a dynamic inversion controller augmented by a direct adaptive neural network to demonstrate performance improvements in the presence of simulated failure/damage. The Gen-2 objectives as implemented on the NASA NF-15B (McDonnell Douglas Corporation, St. Louis, Missouri) airplane created challenges for structural loading limitations. A loads clearance approach including a Structural Loads Model Validation (SLMV) flight phase was developed to ensure flight safety.

Objective

The objective of loads clearance on the F-15 IFCS program was to ensure flight safety by understanding the loads the control system imposed on the airplane. To better understand the loads, it became a necessity to first validate a Loads Model through ground test and flight test. The primary objective of the flight test was to obtain enough data to assess the validity of the loads model.

Approach

The SLMV test phase was added to the IFCS program when it was determined that there was insufficient documentation to support the existing structural loads model correctness. Instrumentation was added and ground-calibrated to validate the loads model. Flight conditions were designed to make a grid covering the IFCS envelope. At each flight condition, the following maneuvers were flown: steady heading sideslips, pitch, roll and yaw doublets, full stick rolls, 4g wind-up turns, 4g loaded rolls and push-over, pull-ups. Once validated, the loads model could be used as a loads preflight prediction tool and a real-time control room monitor for the IFCS Gen-2 flights.

A total of six flights were dedicated to loads model validation. Fifteen subsonic flight conditions were flown. The loads model output was monitored and compared to the instrumentation in real time in the control room. To validate the model, each loads model output was assessed with respect to tracking the instrumentation. Attention was paid specifically to how well the trends were matching, whether the loads model was over-, or under-predicting the measured load and if so, what the offset was. A comparison of loads model output to instrumentation was made on the canards, wings, ailerons, rudders, vertical tails, and horizontal tails.

Most comparisons of the loads model to the instrumentation showed reasonable agreement with the exception of the rudder and aileron. The rudder comparison was not a direct comparison since the loads model computed hinge moment while the instrumentation measured bending. Since the rudder loads were predicted to be less than 50 percent design limit load (DLL), the rudder loads were considered to be less critical than other load measurements. Comparing flight data showed the largest difference was a 30 percent DLL. This large discrepancy was in the aileron hinge moment, showing the model under-predicting the instrument output. The trends, for the aileron comparison however, matched well. Table 1 shows the comparisons that were made.

Table 1. Loads model comparisons.

Surface	Equation Type		
	Bending	Shear	Torque
Left/Right Aileron	X		
Left/Right Wing	X	X	X
Left/Right Canard	X	X	X
Left/Right Vertical Tail	X	X	X
Right Horizontal Tail	X	X	X

With the knowledge gained from the SLMV flight test, it was concluded that the loads model was validated. The IFCS program decided not to fly any maneuvers that would exceed 70 percent DLL based on preflight simulation loads model output. This 70 percent DLL limit for the model output covered the worst-case aileron hinge moment under-prediction, thus assuring that 100 percent DLL from instrumentation would not be exceeded. The loads model was used as a preflight prediction tool and a control room monitor for the IFCS Gen-2 flights.

Postflight data comparisons are used to continue validation of the loads model while using the Gen-2 controller and introducing the simulated failure/damage. The loads model output from flight is compared to both the flight measured load and the loads model output from simulation. From this three-way comparison, margin can be added to the simulation loads model to better predict flight loads. Flight data trends show that the real-time loads model produces 15 percent DLL larger canard loads, 10 percent DLL larger wing and horizontal tail loads, and 25 percent DLL larger aileron loads than in the simulation. Measured loads confirm the flight loads model results. The maximum 70 percent DLL planning requirement for simulation predictions gave sufficient margin such that 100 percent DLL was not exceeded in flight.

After analyzing all of the input parameters that contribute to the loads model for both simulation and in real time, it was determined that the simulation was predicting a smaller angle of attack than what actually occurs in flight. This difference in angle of attack produces smaller loads in the simulation loads model than in the real-time loads model.

Status

The loads clearance method used for flight-testing the IFCS Gen-2 project on the NASA NF-15B airplane proved successful. The first phase of flight-testing the IFCS Gen-2 system was completed without major incident, successfully uncovering strengths and weaknesses of the Gen-2 control approach in flight.

Contact

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BUILD-UP APPROACH TO DETERMINE THE CONNECTION STIFFNESS FOR THE F-15B/QUIETSPIKE™ INTERFACE

Summary

The correlation effort for the QuietSpike™ (Gulfstream Aerospace, Savannah, Georgia) (QS) boom Finite Element Model (FEM) was devised to follow the QS build-up ground vibration testing (GVT) approach. A non-traditional testing and model correlation method had to be implemented because of flight-test article availability and intense project schedule. A mock-up version of the QuietSpike™ boom was designed and fabricated with similar modal characteristics as those of the QS flight article. An FEM of this mock QS was then generated and updated with the appropriate GVT data. This facilitated the mating of the actual QS FEM, once correlated to GVT, to the F-15B FEM.

Objective

The objective of this work was to determine the connection stiffness on the airplane radar bulkhead where the QS and the F-15B mate, such that when the FEM of the actual QS was validated, it could be readily mated to the airplane FEM. This approach was established to enable the project to progress in the model development, testing, and flutter predictions without the actual QS flight test hardware.

Approach

When the F-15B (McDonnell Douglas, St. Louis, Missouri) testbed airplane receives a new flight research experiment, one of the greatest unknowns is the connection stiffness between the experiment and the airplane. The traditional approach to determining the connection stiffness entails the mating of the article to the airplane and performing a GVT. The GVT data is then used to update the connection stiffness in the FEM of the test article mated to the airplane. However, for the QS project, because of the availability of the actual flight test article, it was determined that flight preparation work could still be done if a different approach was implemented.

A series of four GVTs coupled with several model updates were done to acquire the test article connection stiffness before the QS arrival at the NASA Dryden Flight Research Center. A mock QS boom was designed and fabricated to be similar to the QS flight hardware in mass, center of gravity, and inertias, such that it could be used in GVTs. A detailed FEM of this mock QS boom was created from computer aided design (CAD) drawings. A second, simplified beam model of the mock QS was created and then updated to match the detailed model. Figure 1 shows the various models of the mock QS that were developed to accomplish this task.

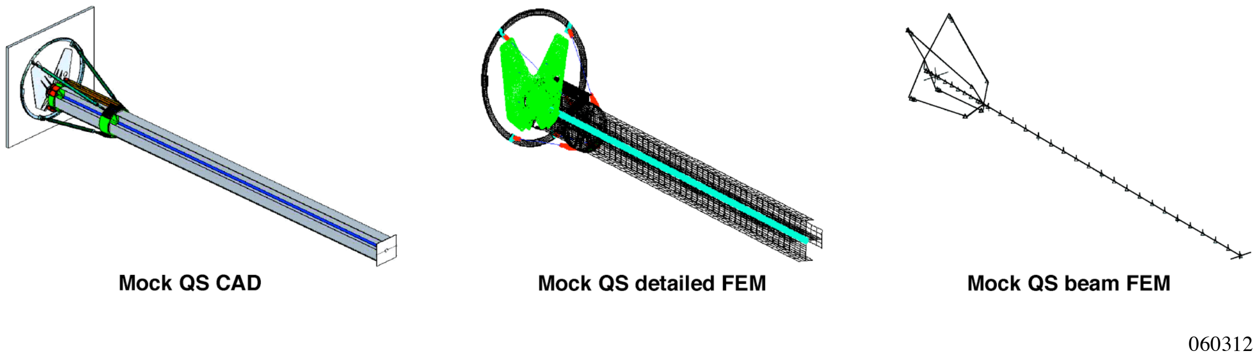

Figure 1. Mock QS model development.

The mock QS beam model was updated to the mock QS detailed model using an in-house Mode Matching Code. This Mode Matching Code modifies the current model to create a new model that has similar mode shapes and frequencies as the target model. In this case, the target model is the detailed model and the modification is done to the beam model. The first three modes of the mock QS detailed model or the target modes are shown in fig. 2.

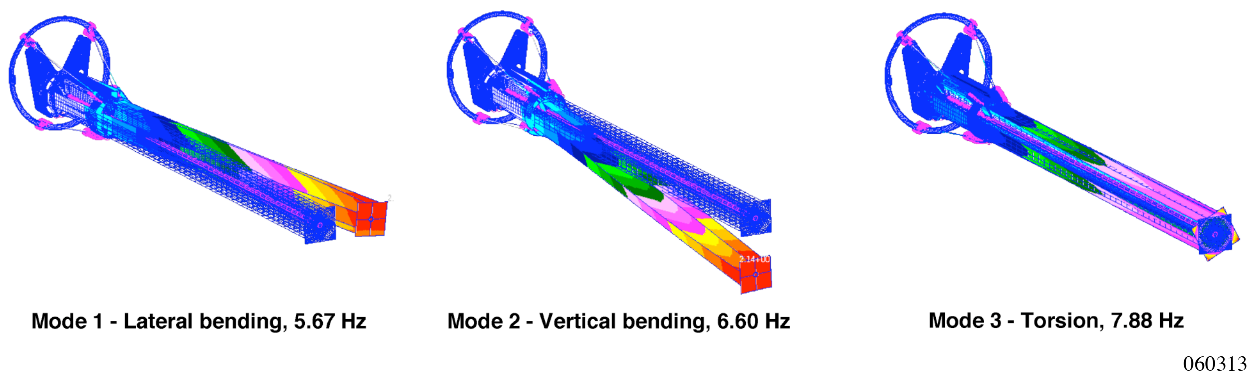

Figure 2. Mock QS target mode shapes and frequencies.

The Mode Matching Code outputs frequencies and modal assurance criteria (MAC) values that allow for the comparison of the updated model to the target model. A MAC value quantifies how well modes match between the target and the updated models. The frequency and MAC values for the mock QS beam model are listed in table 1.

Table 1. Comparison of updated mock QS beam and detailed FEMs.

Mode	Equivalent Beam FEM (updated model)		Detailed FEM (target model)
	MAC (%)	Frequency (Hz)	Frequency (Hz)
1	98.29	5.674	5.675
2	98.14	6.607	6.602
3	78.64	7.867	7.876

The next step consisted in validating the mock QS beam model with the actual mock QS structure. This was accomplished in the first GVT, which consisted of attaching the mock QS boom to a strongback and exciting it in the lateral and vertical directions. This GVT data was used to update the structural properties of the mock QS beam model. The second GVT consisted of attaching the mock QS boom to the aircraft using the exact same pick-up points and fasteners as the flight hardware. The mock QS boom was again excited in both the lateral and vertical directions. This GVT provided the data necessary to correlate the analytical connection stiffness between the F-15B airplane and mock QS FEM to the actual connection stiffness. At this point, the F-15B model was ready to have the actual QS FEM mated to its radar bulkhead once the QS FEM was validated with GVT data.

Status

The QS modeling effort has been finalized and the flutter analyses have been completed. The F-15B/QS project has been through flight readiness review and is currently preparing for its first flight.

Contact

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QUIETSPIKE™ BUILD-UP GROUND VIBRATION TESTING APPROACH

Summary

The F-15B (McDonnell Douglas, St. Louis, Missouri) airplane, tail number 836 was selected to fly Gulfstream Aerospace Corporation's (GAC) (Savannah, Georgia) QuietSpike™ (QS) project; however, this experiment is very unique and unlike any of the previous testbed experiments. It involves the addition of a relatively long quiet spike boom attached to the radar bulkhead of the airplane. This QS experiment is a stepping stone to airframe structural morphing technologies designed to mitigate the sonic boom strength of business jets over land. Prior to flying the QuietSpike™ boom on the F-15B airplane, several ground vibration tests (GVT) were required in order to understand the QS modal characteristics and coupling effects with the airplane. Because of the project's intense schedule, a "traditional" GVT of the mated F-15B QuietSpike™ ready-for-flight configuration would not have left sufficient time available for the finite element model update and flutter analyses before flight test.

Objective

The objective of the QS build-up ground vibration testing approach was to ultimately obtain confidence in an F-15B QuietSpike™ finite element model (FEM) to be used for the flutter analysis. In order to obtain this F-15B QS FEM with reliable foundation stiffness between the QS and F-15B radar bulkhead, several GVT configurations were performed. Each of the four GVTs performed had a specific objective. The overall intent was to provide adequate data to replicate a "traditional" F-15B QS GVT with actual ready-for-flight hardware. The NASA Dryden Flight Research Center was in charge of the 1st, 2nd, and 4th GVTs and the 3rd GVT was conducted by ATA Engineering at GAC in Savannah, Georgia. In order for this build-up GVT approach to be feasible, it was absolutely critical that each GVT configuration matched as closely as possible the connection interface between the QS and airplane radar bulkhead.

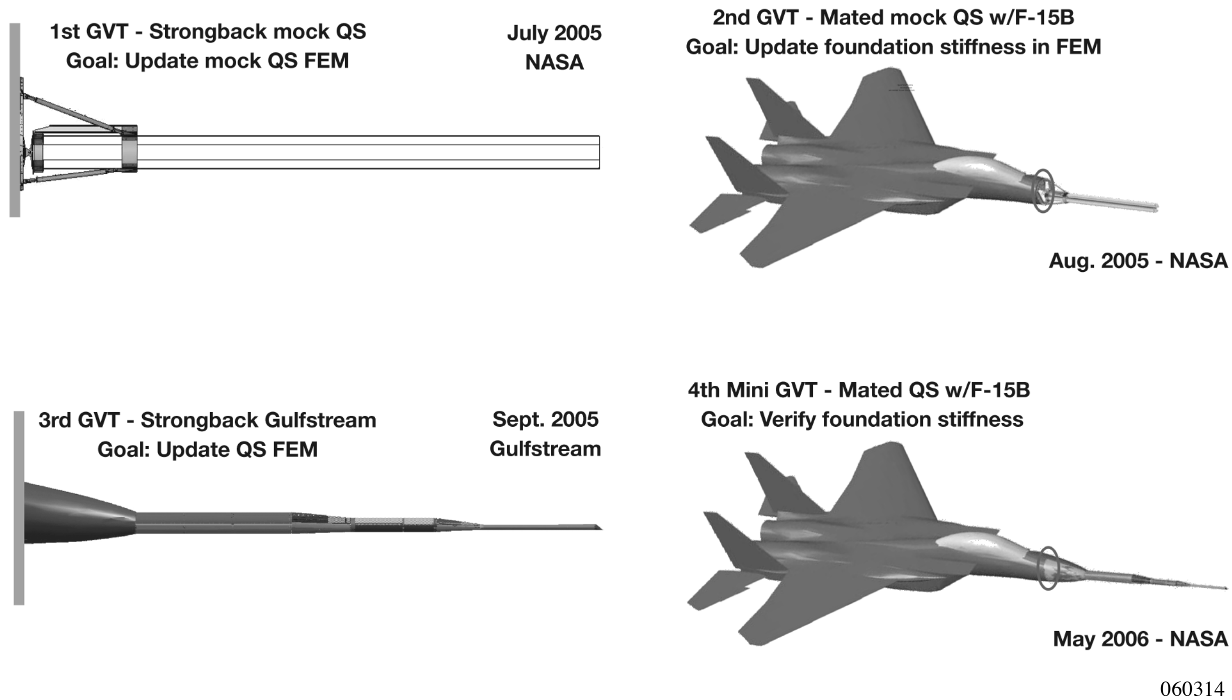

Figure 1. QuietSpike™ build-up ground vibration testing approach.

Approach

To conduct the 1st and 2nd GVTs, a mock-up quiet spike boom was designed and fabricated with similar weight, center of gravity, and moment of inertia characteristics to that of the extended configuration of the QuietSpike™ flight hardware. The mock QS was also designed to interface with the same supporting structure hardware that the QS uses to mount to the airplane radar bulkhead. As shown on fig. 1, the 1st GVT performed was the Strongback mock QS GVT with the goal of updating the analytical mock QS FEM based on GVT results. This 1st GVT and FEM update assumed a completely rigid connection between the mock QS and the Strongback. The 2nd GVT involved the mated mock QS with the F-15B airplane. This 2nd test characterized the scaling factor for the connection foundation stiffness between the mock QS and the F-15B radar bulkhead. The 3rd GVT was very similar to the 1st GVT in the respect of using a Strongback and assuming a rigid connection, but it was performed on the actual QS flight hardware at GAC. The Strongback QS GVT data was used to update the extended QS FEM from which the retracted QS FEM was analytically generated. Once the extended QS FEM was updated from the 3rd GVT data, the foundation stiffness scaling factor (springs) established from the 2nd GVT were included in the QS FEM and attached to the F-15B FEM. This combined F-15B QS FEM was used as a baseline model for parametric variations in QS/F-15B foundation stiffness and in the QS joint stiffness for flutter sensitivity analyses. The 4th *mini*-GVT performed was on the mated QS with the F-15B airplane; the final flight configuration. The objectives were to measure the primary frequencies of the extended QS on the F-15B airplane for validation of the foundation stiffness scaling factor and also to measure the primary frequencies of the retracted QS to verify that the retracted QS FEM configuration was correctly modeled. Data from the 4th GVT showed excellent F-15B QS test-to-model frequency correlation (within 3.5 percent, see table 1), so no model updating was necessary as was the original intent of the whole QS GVT build-up approach. The final GVT data was also used to determine which flutter analysis case was appropriate to select for the sensitivity study.

Table 1. The F-15B QS FEM and 4th GVT comparison.

Mode Shape Description	F-15B QS FEM (free-free) Freq. (Hz)	4th GVT (Soft Support) Freq. (Hz)	FEM vs. 4th GVT % Error
	QuietSpike™ Extended		
QS Lateral 1st Bending, slight Vertical Tail 1st Bending-Antisymm.	5.87	6.03	-2.73
QS Vertical 1st Bending, slight Horz. Stab. 1st Bending-Symm.	6.7	6.83	-1.94
	QuietSpike™ Intermediate Position		
QS Lateral 1st Bending, Fuselage Lateral Bending, Vertical Tail 1st Bending-Antisymm.	N.A.	6.23	– – –
QS Vertical 1st Bending, Fuselage Vertical Bending, Horz. Stab. 1st Bending-Symm.	N.A.	7.75	– – –
	QuietSpike™ Retracted		
QS Lateral 1st Bending, Fuselage Lateral Bending, Vertical Tail 1st Bending-Antisymm.	7.00	6.77	3.29
QS Vertical 1st Bending, Fuselage Vertical Bending, Horz. Stab. 1st Bending-Symm., Vertical Tail 1st Bending-Symm.	8.26	7.97	3.51

Status

All ground vibration testing and finite element model updates have been completed along with the final flutter analysis. The QuietSpike™ experiment is expected to start flying in the summer of 2006.

Contact

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FIBER OPTIC SENSOR ATTACHMENT DEVELOPMENT AND PERFORMANCE EVALUATIONS

Summary

Research conducted in the Flight Loads Laboratory (FLL) at the NASA Dryden Flight Research Center (DFRC) has subjected fiber optic (FO) sensors to hostile environments for in-flight applications and hot-structures ground testing (on hypersonic or reentry vehicles). Sensor attachment of both Fiber Bragg Gratings (FBG) and silica-based Extrinsic Fabry-Perot Interferometers (EFPI) have been accomplished on metallic and composite substrates. The FO sensors have been successfully demonstrated:

- at room and elevated temperatures (to 1850°F)
- with combined applied thermal and mechanical loads
- and on both small laboratory coupons and large-scale structures for ground testing

Further development has been initiated to enhance the upper temperature limit for measuring strains. Current ceramic composite materials are to be used for structural load-bearing components and have applications in temperatures as high as 3000°F. A means for taking measurements on these materials at their maximum operating temperature is desired to validate models and ultimately minimize the size and mass of vehicle components.

Objective

The objective of this project is to develop attachment techniques and evaluate FO strain and temperature sensor performance for structural health monitoring aerospace applications. The main tasks during sensor evaluation include: 1) characterization of apparent strain (ξ_{app}) of gold-coated silica EFPI's for post-test corrections of indicated strain values, 2) surface attachment and correction of thermal output of FBG to 500°F, and 3) develop attachment techniques for sapphire strain sensors for applications that exceed the maximum operating temperature of the current silica EFPI sensor.

Approach

Thermal-sprayed sensor attachment procedures were developed and tested for both carbon-carbon (C/C) and carbon-silicon carbide (C-SiC) substrates. Testing of silica EFPI sensors to 1850°F was performed to evaluate attachment integrity and sensor performance. Dilatometer tests compared substrate expansion versus sensor output and generated ξ_{app} correction curves for X-37 control surface tests. The EFPI's were attached to the large-scale C/C X-37 qualification flaperon unit and provided strain measurements during hot-structures ground testing to temperatures of 1900°F.

Thermal testing of FBGs was completed to 400°F. The FBGs were tested both bonded and unbonded to an epoxy graphite coupon. It was found that FBG thermal output had excellent correlation with calculated theoretical microstrain per temperature ratios.

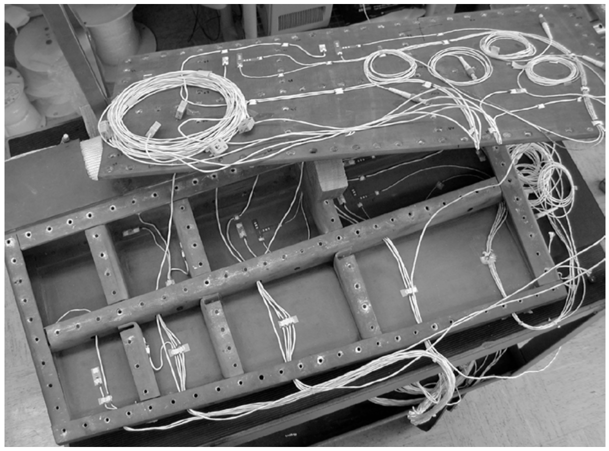

Photo courtesy of Anthony Piazza

Figure 1. Leeward side instrumentation on X-37 flaperon qual unit (access panel removed).

Photo courtesy of Anthony Piazza

Figure 2. The FBG instrumented epoxy graphite coupon prior to thermal tests.

Status

Further work with gold-coated silica EFPI sensors is required to address sensor-to-sensor scatter on C/C composites and to refine thermal corrections for indicated strain values. Work on the development of a sapphire strain sensor using similar interferometry technologies is continuing under a NASA Phase II Small Business Innovation Research (SBIR) with Lambda, Inc. As these sensors are still in production, their technical readiness level is 2.

Thermal-spray attachment research has been accomplished through an unsolicited proposal with Drexel University's Center for the Plasma Processing of Materials, directed under Dr. Richard Knight. Very good results have been achieved on the C/C SiC conversion layer to at least 2400°F. Future work includes attachment studies conducted on C-SiC substrates (with oxidation prohibiting coating) and functionally graded materials in support of ruddervator hot-structures testing under the NASA AMRD Hypersonics program.

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PROGRESS WITH RECONFIGURABLE INSTRUMENTATION SIGNAL CONDITIONING DEVELOPMENT

Summary

New signal conditioning circuitry for increasing flight research productivity has been designed, fabricated, and is now undergoing laboratory testing. The new design is intended to have the same analog functionality and research-quality performance of existing designs, but with remotely adjustable gain, offset, and filter settings.

Objective

The objective of the project is to design a printed circuit board that does not require physical access or replacement of components when fine-tuning of circuit performance is required.

Approach

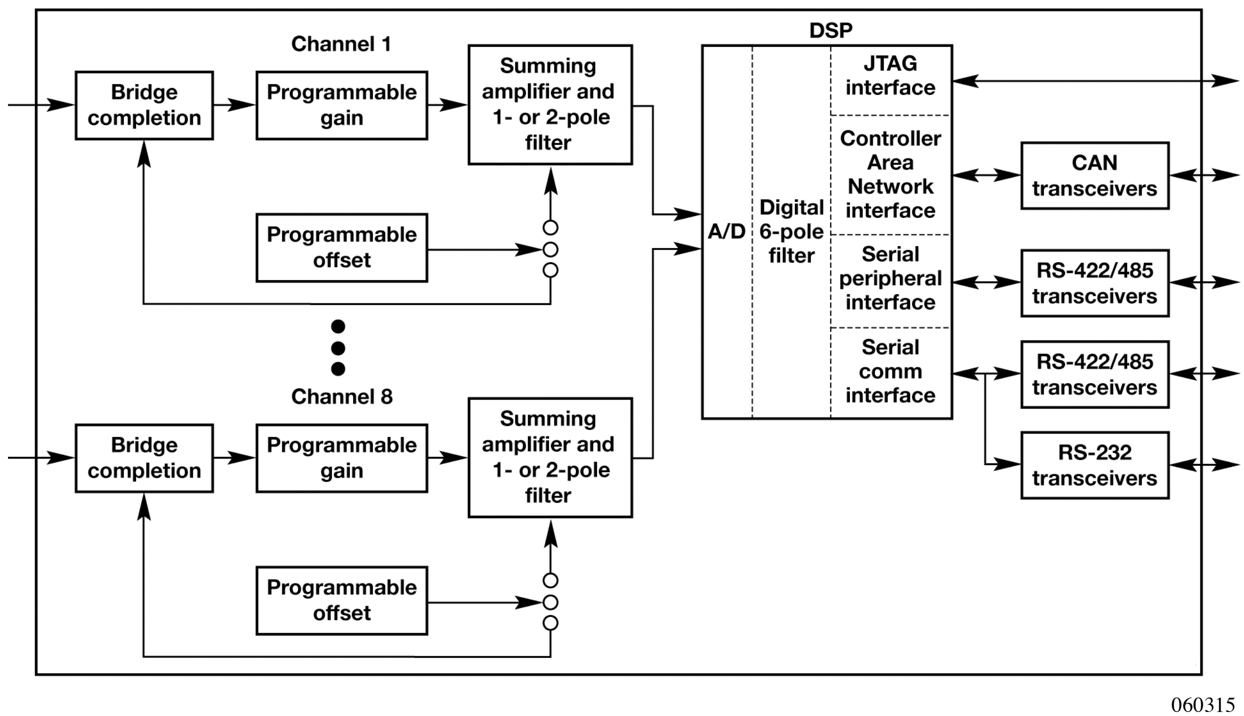

Figure 1. Reconfigurable Instrumentation Signal Conditioning simplified block diagram.

Status

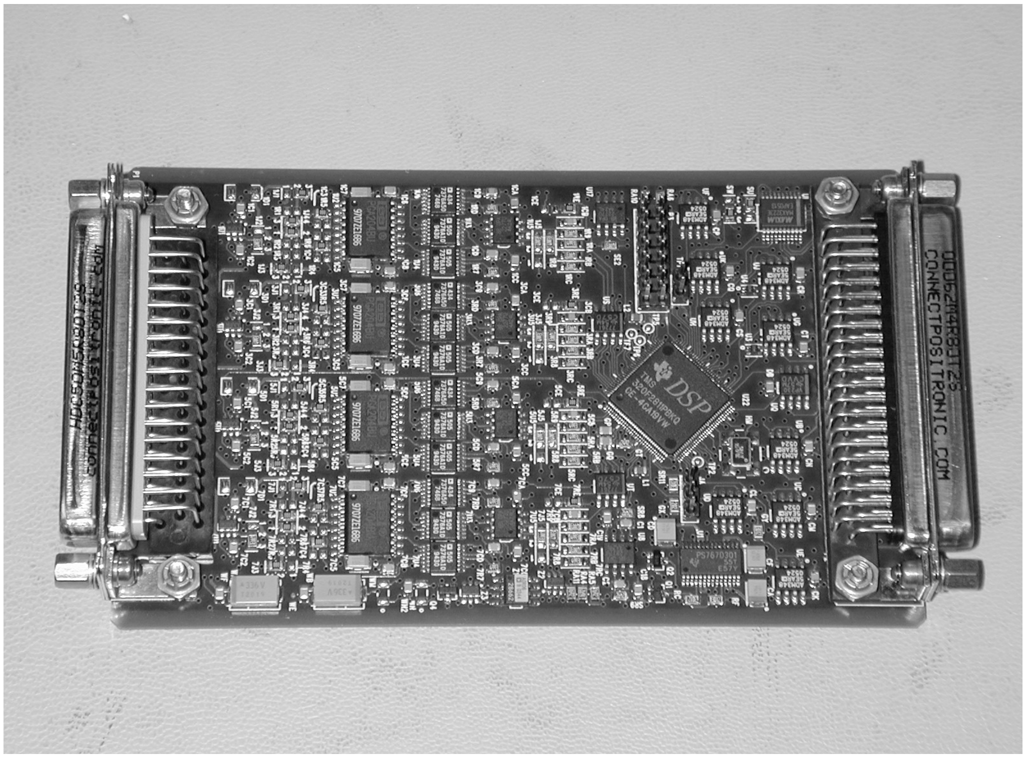

Photo courtesy of Phil Hamory

Figure 2. Photograph of Reconfigurable Instrumentation Signal Conditioning printed circuit board.

The printed circuit board has been designed and fabricated and is now undergoing laboratory testing. Capabilities that have been proven so far include adjusting the gain and offset, providing a serial data stream to the serial digital interface module of one of the existing Pulse Code Modulation (PCM) systems in use at the NASA Dryden Flight Research Center, generating an independent PCM stream, and remotely reprogramming the board’s flash memory. Noise level, environmental, and in-flight testing are to follow.

Reference

1. Hamory, Philip J., "Reconfigurable Instrumentation Signal Conditioning," in 2004 Research Engineering Annual Report, pg. 53, NASA/TM-2006-213677, 2006.

Contacts

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NONLINEAR BLACK-BOX MODELING OF AEROELASTIC SYSTEMS USING A STRUCTURE DETECTION APPROACH

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 provide a simpler controller design. Structure computation as a tool for "black-box" modeling is not well known to the flight-test community, but may be of critical importance in the development of robust, parsimonious models. Moreover, this approach may lead to efficient strategies for rapid envelope expansion that may save significant development time and reduce costs. Structure detection methods applicable to NARMAX (Nonlinear AutoRegressive Moving Average eXogenous) modeling are applied to aeroelastic dynamics, whose properties are demonstrated via continuous-time simulations. Simulation results from a nonlinear dynamic model of aircraft structural free-play demonstrate that methods developed for NARMAX structure computation provide a high degree of accuracy for selection of the exact model structure from an over-parameterized model description. Applicability of these methods to the F/A-18 Active Aeroelastic Wing using flight-test data is shown by identifying a parsimonious system description with a high percent fit for cross-validated data.

Objective

The behavior of many nonlinear dynamic systems can be represented as an expansion of their present output value in terms of present and past values of the input signal, past values of the output signal, and past values of the innovation, represented as a discrete-time polynomial. A system modeled in this form is popularly known as a NARMAX and is linear-in-the-parameters. When fully expanded, this system representation yields a large number of possible terms which may be required to represent the dynamic process. In practice, many of these candidate terms are insignificant and, therefore, can be removed. Consequently, the structure detection problem is that of selecting a subset of candidate terms that best predicts the output while maintaining an efficient system description. There are two fundamental approaches to the structure detection problem: (1) exhaustive search, whereby every possible subset of the full model is considered, or (2) parameter variance, whereby the covariance matrix based on input-output data and estimated residuals is used to assess parameter relevance. Both have problems. Exhaustive search requires a large number of computations whereas parameter variance estimates are often inaccurate when the number of candidate terms is large.

System identification, or black-box modeling, is a critical step in aircraft development, analysis, and validation for flight worthiness. 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 NARMAX structure detection techniques as a tool for model development is that it can provide a parsimonious system description which can describe complex aeroelastic behavior over a large operating range. Consequently, this provides models that can be more robust and, therefore, reduces development time.

Approach

Investigations of the applicability of NARMAX structure detection methods were performed to (1) a simulated model of aircraft freeplay and (2) the F/A-18 Active Aeroelastic Wing (AAW) flight test data. Results show that NARMAX structure detection techniques provide a high degree of accuracy for selection of the exact model structure using simulated data. The figure shows the predicted output for a cross-validation data set for the identified structure. The upper panel displays the full 26 s time history of the accelerometer response recorded on the right wing outer-fold aft. The lower panel displays a 10 s segment of the predicted output superimposed on top of the measured output which accounts for over 94 percent of the measured output variance. The result demonstrates that the computed model structure is capable of reproducing the measured output with a high degree of accuracy.

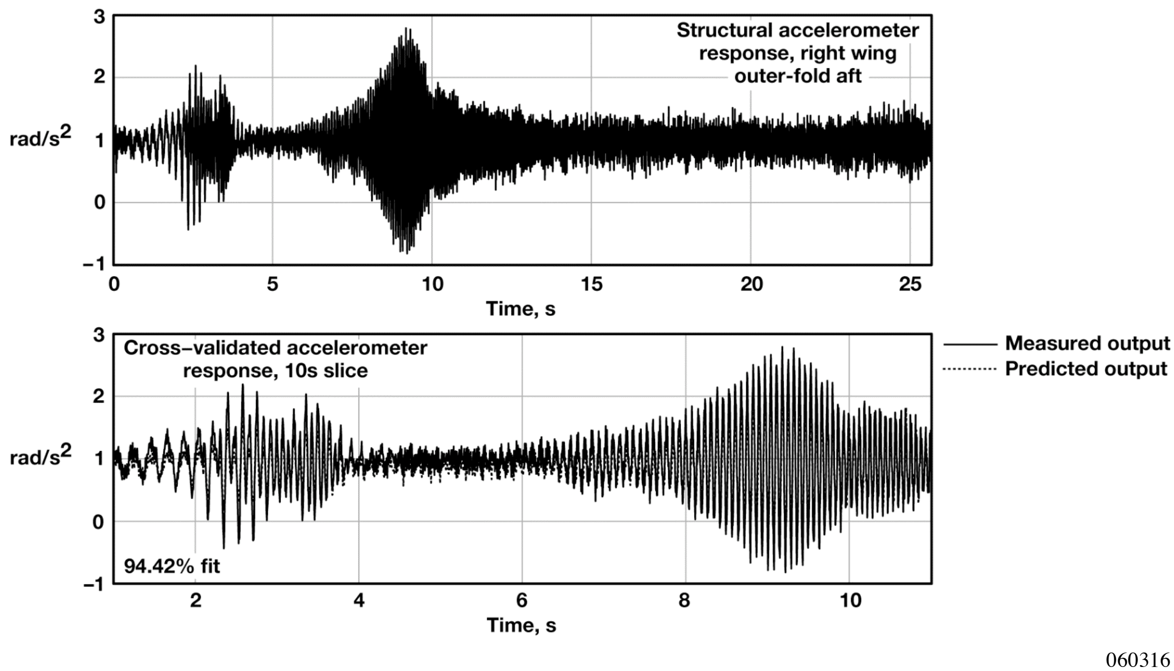

Figure 1. Entire and limited time history of structural accelerometer response.

Status

In aerospace systems analysis, the main objective is not only to estimate system parameters, but to gain insight into the structure of the underlying system. Structure computation is of significant relevance and importance to modeling and designing aircraft and aerospace vehicles. This analysis provides a parsimonious system description with a high percent fit for cross-validated data. Black-box modeling research contributes to the understanding of the use of structure detection for modeling and identification of aerospace systems. Results may have practical significance in the analysis of aircraft dynamics during envelope expansion leading to more efficient control strategies. This structure detection approach could also allow greater insight into the functionality of various systems dynamics by providing a quantitative model that is easily interpretable.

Contact

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HIGH-SPEED FIBER BRAGG GRATING INTERROGATION SYSTEM DEVELOPMENT

Summary

A high-speed ground-based interrogation system was developed at the NASA Dryden Flight Research Center for the study and characterization of fiber Bragg gratings. The system utilizes a Compact peripheral component interconnect (CompactPCI) tunable laser along with fiber Bragg gratings to measure surface strain. A Dryden-developed wavelength-to-strain algorithm was implemented to maximize computational efficiency resulting in higher sample rates. To validate this system, laboratory tests were performed with fiber optic sensors and collocated conventional strain gages. Test results show that the high speed system represents a viable laboratory interrogation tool for fiber Bragg gratings.

Objective

The objective of this research is to develop a high-speed, 10 samples per second (sps), fiber Bragg grating interrogation system applicable for use in a laboratory environment. One drawback to the fiber Bragg grating technology has always been sample rate limitations. This has been the result of processing large amounts of data to yield one sample iteration. In the early stages of development, four years ago (2001), the sample rate was one sample every 10 s. In 2004, Dryden achieved 1 sps (currently in industry, the sample rate is 4 sps). The system has also been designed to measure strain over a 20 ft segment of sensor fiber. Information from the interrogation system is recorded locally and a subset of that information sent to a monitoring notebook computer via an Ethernet connection.

Approach

A redesign of an existing fiber optics interrogation system was performed considering an increase in performance by an order of magnitude. The previous design was proven in the Flight Load Laboratory using a notebook personal computer (PC) along with Dryden's second generation Fiber Optics Instrumentation Development (FOID) system, as seen in fig. 1. It is a single fiber system with a sample rate of 1 sps.

Photo courtesy of Anthony Piazza

Figure 1. Previously developed interrogation.

The subcomponents that make up the FOID system are as follows: a C-band tunable laser, an optical network, an optical-to-electrical (O/E) amplifier/converter, a high-speed (1.25 MHz) analog-to-digital (A/D) converter, and an embedded processor with large mass storage capacity.

The FOID system collected, archived, processed, and sent data via a 10/100 Ethernet connection to a notebook PC that displayed the results. Great emphasis was placed on increasing processor power, optimizing the interrogation algorithm, and increasing the A/D sample rate.

The redesign, called FOID Ultra Fast (UF) (fig. 2), included migrating to a CompactPCI extensions for instrumentation (PXI) bus structure where off-the-shelf compact, high-performance components such as faster processors and A/D cards could be incorporated. The new design features an Intel 2.0 GHz, Core Duo T2500 processor from National Instruments (Austin, Texas) with a gigabit Ethernet interface for fast data transfers to a host PC. A 2 channel, 50 MHz per channel, A/D card is used to host 2 sensor fibers, with each fiber capable of sensing up to 20 ft, with a strain measurement every 0.5 in, for a total of 960 measurements. A modified, per Dryden's specifications, CompactPCI PXI C-band tunable laser was also integrated. The Dryden Bragg grating interrogation algorithm, which featured a "hopping Fast Fourier Transform (FFT)," was optimized for performance and features a "hobbling FFT." With the use of the original Dryden algorithm, the FOID system achieves a sample rate of 18 sps, while the optimized system yields the same speed performance but increases system resolution by a factor of 4.

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Figure 2. The FOID UF system with notebook CPU.

Status

Multiple systems are being assembled for off-site use at other facilities, including NASA, universities, and corporate industry. A flight version of this system is being developed for possible opportunities on board Dryden research aircraft. Also, a multi-channel (32 fiber) system is also being considered for very large scaled testing as well as higher sampling systems.

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HILBERT–HUANG TRANSFORM-BASED STABILITY SPECTRAL ANALYSIS

Summary

The Hilbert–Huang transform (HHT) has been applied to analyze flight flutter data. The analysis shows the yielding of the test wing after the onset of flutter, but just before breaking off at the wingtip. Based on HHT, research using a new stability spectrum shows both positive (stable) and negative (unstable) damping. Flutter occurs in a different frequency range than that determined by modal analysis and identification. Both HHT- and the Teager Energy Operator-based nonlinearity indicator show that structural dynamics are nonlinear throughout the flight-test flutter maneuver.

Objective

Past analysis of Aeroelastic Test Wing (ATW) data indicates that flutter occurred for the bending mode at 18 Hz and for the torsion mode at 24 Hz, with a decreasing bending mode damping and an increasing torsion mode damping. Video footage taken during the test clearly shows much larger amplitude and lower frequency bending motions, which are totally unaccounted for in standard analysis of the accelerometer data. Although the tests were very successful, the data analysis had always been a problem because of the transient and nonlinear properties of the wing-fracturing event. No traditional method fit the requirement to analyze these data effectively, as one needs a method that will take the nonstationary and nonlinear processes into full consideration. The Hilbert–Huang transform (HHT) is designed specifically for the analysis of nonstationary and nonlinear processes. The data analysis part of the flutter test is revisited using the HHT. This approach opens a new way to study a non-destructive test and structural health monitoring data. Also researched is a newly-developed Stability Spectral Analysis based on HHT.

Approach

Shortcomings of present state-of-the-art methods in computing the damping spectrum are accounted for in developing the stability spectrum. The stability spectrum is built by using the damping computed on the cubic, spline-fitted envelope of the HHT intrinsic mode functions (IMFs). The damping loss factor is a function of instantaneous amplitude, $\alpha(\omega,t)$, in terms of time, t , and frequency, ω . The damping spectrum has previously been defined as

$$\eta^2(\omega,t) = \frac{\left[\dfrac{-2}{a(\omega,t)}\dfrac{d\,a(\omega,t)}{dt}\right]^2}{\omega^2 + \left[\dfrac{1}{a(\omega,t)}\dfrac{d\,a(\omega,t)}{dt}\right]^2}$$

In the current research, damping loss factor is now being tracked for both positive (indicating stable condition) and negative (indicating unstable condition) values, thereby defining a stability index, rather than the above expression in which the difference between the positive and the negative signs is obliterated by the squaring operation. To define the stability spectrum, the following improvements are made over the previous, state-of-the-art damping spectral analysis: a change of the envelope determined through Hilbert transform to cubic spline, definition of both positive damping (as indication for stable dynamics) and negative damping (as indication for unstable dynamics) as the stability index, and computation of the instantaneous frequency using a new normalized HHT. Also, the addition of a nonlinearity indicator through the Teager Energy Operator will flag the time period when the dynamics become nonlinear.

The stability spectrum of the ATW flutter data indicates that the most unstable modes cover 2 to 5 Hz vibrations in addition to the 18 Hz vibration. The displacement of the ATW boom at the tip of the airfoil was computed by double integration of the acceleration data in the figure (fig. 1). Here, one can see some low-frequency displacement changes as in the visual assessment of the flutter. The high-frequency vibration is still coexisting with the much larger amplitude low-frequency vibration. The final bending just before fracture of the wing reaches almost 12 cm. The Hilbert spectrum for the displacement is given in fig. 2. The vibration at 18 Hz is clearly shown in the Hilbert spectrum of the displacement. In addition to these high-frequency oscillations, there are low-frequency components as revealed in the stability spectrum.

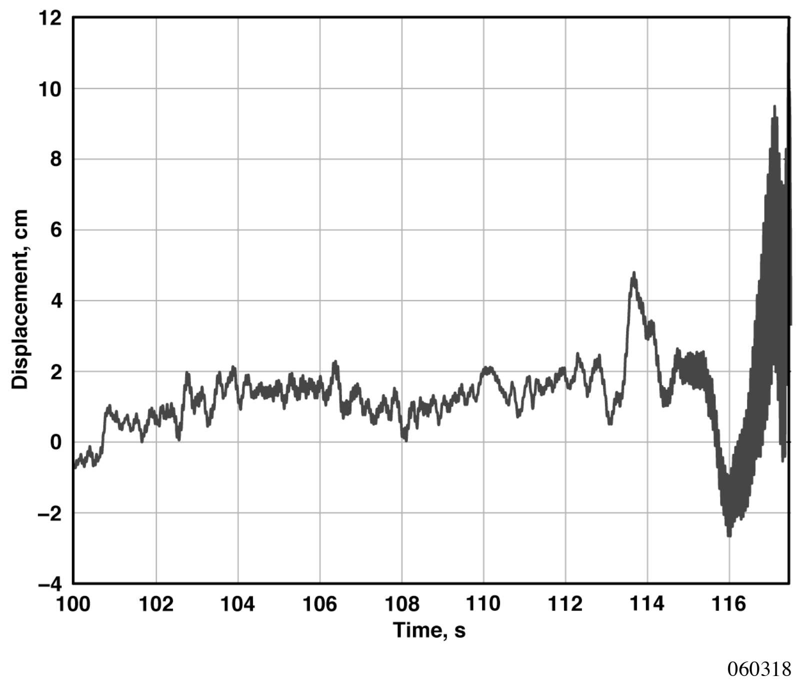

Figure 1. Displacement of the ATW boom.

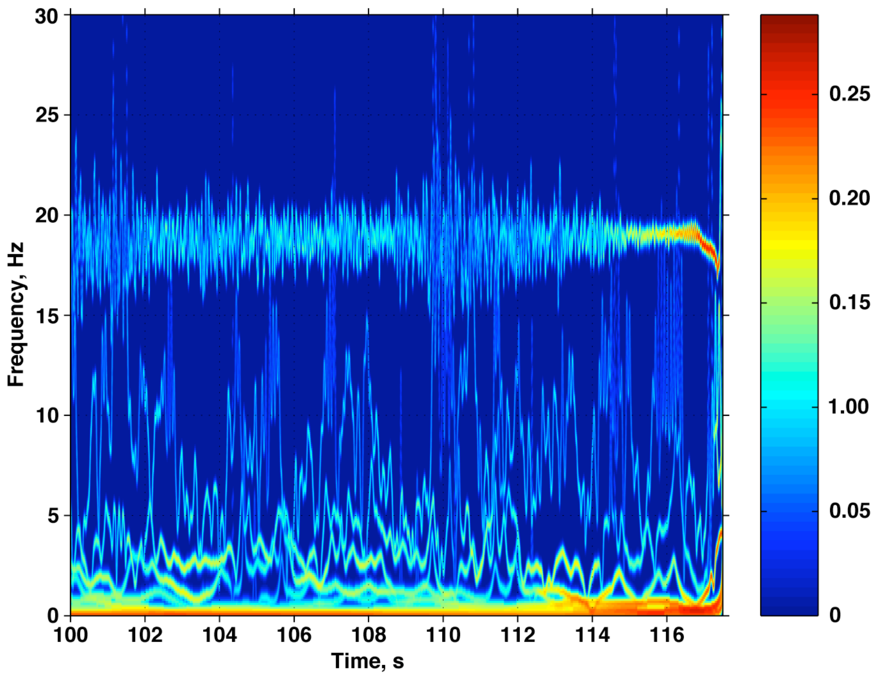

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Figure 2. The Hilbert spectrum for the displacement of the ATW boom.

Status

The HHT can provide crucial and detailed information for studying linear and nonlinear structural dynamics. The stability spectrum provides marginal damping factor as functions of time and frequency, therefore it is a crucial diagnosis tool to detect the health of the system. The ATW flight test data has demonstrated that HHT can be extremely useful in analyzing vibration data and also as part of a nondestructive health monitoring system. In any future structural health monitoring system, this stability index can be a vital structural health indicator.

Contact

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MULTI-INPUT MULTI-OUTPUT FLIGHT DATA ANALYSIS WITH HHT

Summary

This research investigates the utility of the Hilbert–Huang Transform (HHT) for the analysis of aeroelastic multi-input multi-output (MIMO) flight data. The recently-developed Hilbert–Huang algorithm addresses the limitations of the classical Hilbert transform through a process known as empirical mode decomposition. Using this approach, the data is filtered into a series of intrinsic mode functions, each of which admits a well-behaved Hilbert transform. In this manner, the Hilbert–Huang algorithm affords time-frequency analysis of a large class of signals. The purpose of this research is to demonstrate the potential applications of the Hilbert–Huang algorithm for the analysis of multi-loop aeroelastic systems. Applications for correlations between system input and output, and among output sensors, characterize the time-varying amplitude and frequency correlations present in the various components of multiple data channels. Online stability analyses and modal identification are new applications for the algorithm, particularly in the area of aeroelastic and aeroservoelastic systems analysis.

Objective

An objective of signal-adaptive basis function derivations using the Hilbert–Huang algorithm is to yield intrinsic mode functions (IMF) that give instantaneous frequencies as functions of time that permit identification of imbedded structures. Instantaneous frequency is a quantity critical for understanding nonstationary and nonlinear processes. An empirical mode decomposition (EMD) process responds to the dilemma surrounding the applicability of instantaneous frequency from the Hilbert transform of a general multicomponent signal. The EMD decomposes a multicomponent signal into its associated monocomponents, called IMFs, while not obscuring or obliterating the physical essentials of the signal. It allows the traditional definition of instantaneous frequency to be complete by being applicable to both mono- and multicomponent signals. The adaptive and nonarbitrary decomposition using EMD produces an orthogonal set of intrinsic components, each retaining the true physical characteristics of the original signal.

There is a multiresolution quality in the EMD process that deals with intermittency by allowing multiple time-scales within an IMF, but not allowing a similar time-scale simultaneously with other IMFs. System identification in the IMF sub-component environment is a practical endeavor in the domain of multiresolution system identification.

Approach

Correlations are made between Hilbert-transformed IMFs of various signals given the associated complex analytic signals $Z_x(t)$ and $Z_y(t)$ from original signals $x(t)$ and $y(t)$

$$Z_x(t) = x(t) + ix_H(t); \qquad Z_y(t) = y(t) + iy_H(t)$$

where $x_H(t)$ and $y_H(t)$ are Hilbert transforms of signals $x(t)$ and $y(t)$, respectively, by considering the cross-analytic signal defined by

$$Z_{xy}(t) = Z_x(t)Z_y(t).$$

A measure of the local correlation between components, in terms of simultaneous changes in instantaneous amplitude or frequency (phase) between analytic signals, is the Hilbert Local

Correlation Coefficient (HLCC). From this, we get instantaneous transfer function (ITF), its instantaneous magnitude (IM), and its instantaneous phase (IP).

$$ITF(t) = Z_{xy}(t) / Z_{xx}(t)\,; \qquad IM(t) = \left|ITF(t)\right|; \qquad IP(t) = \cos^{-1}\left[HLCC(t)\right]$$

The concept of using the orthogonal analytic IMFs from the inputs and outputs is used to establish a multi-loop connotation of input IMFs to output IMFs to generate correlation and stability properties. The input IMFs are interpreted as an orthogonal decomposition of the input(s), and the same for output IMFs. This can be generalized to MIMO signal analysis where each signal is represented by its Hilbert-transformed EMD. From the transformed IMFs, $\left\{Z_x(t); Z_y(t)\right\}$ defines the corresponding set of two-dimensional Hilbert empirigrams, $\left\{H_x(v_x,t); H_y(v_y,t)\right\}$, and the Hilbert cross-empirigram $H_{xy}(v,t) = H_x(v_x,t)H_y(v_y,t)$, for each IMF number, v.

A singular value analysis of the operator between Hilbert-transformed IMFs represents relative contributions from the principal cross-correlation analytic IMFs as a result of correlation of input analytic IMFs to output analytic IMFs. The maximum singular value of this input-output operator corresponds to the structured singular value with a full-complex uncertainty block structure. For time-domain MIMO signal analysis, it is appropriate to combine complex uncertainty blocks for each input-output into a multi-block structure, where each complex sub-block corresponds to an input-output analytic IMF complex uncertainty structure. This is analogous to frequency-domain robust stability analysis, except that now at each point in time, there is instantaneous frequency information and there results a time-dependent robust stability analysis. From the Aerostructures Test Wing (ATW), IMFs of a wingtip accelerometer are shown in fig. 1, and corresponding structured singular value plots between input analytic IMFs and all three wingtip accelerometer output analytic IMFs are shown in fig. 2.

Figure 1. The EMD of wingtip accelerometer near flutter (original at top, residual at bottom).

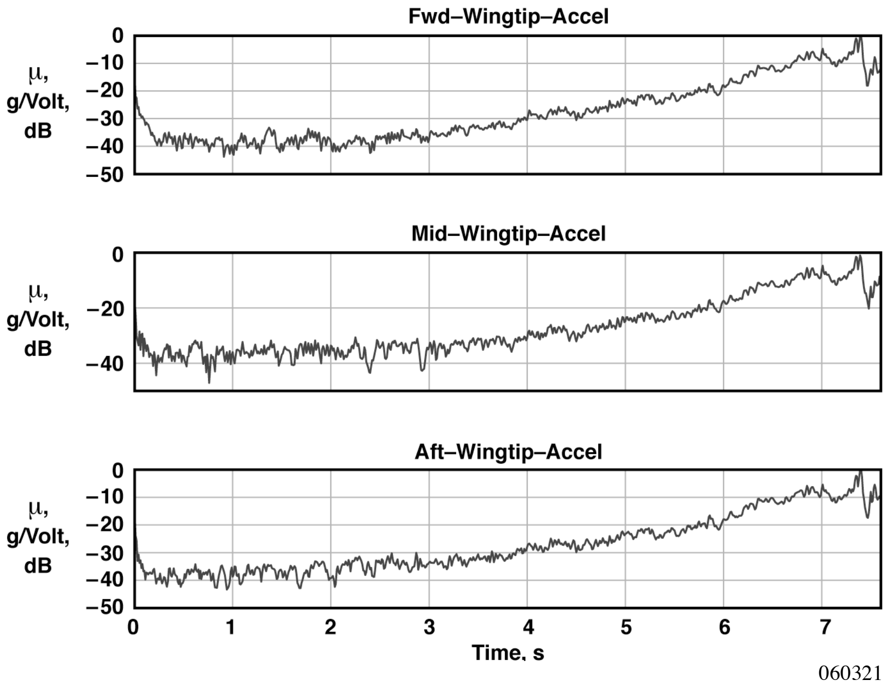

Figure 2. Structured singular value between input-output IMFs of wingtip accelerometers.

Status

Investigation of correlations between input-output and between sensors in terms of instantaneous system identification using system input-output signal analysis characterizes the time-varying amplitude and frequency components of multiple data channels in terms of the IMFs of the HHT. These procedures are significant departures from Fourier and other time-frequency or time-scale wavelet approaches.

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AEROELASTIC FLIGHT DATA ANALYSIS WITH THE HILBERT–HUANG ALGORITHM

Summary

This research investigates the utility of the Hilbert–Huang transform (HHT) for the analysis of aeroelastic flight data. It is well known that the classical Hilbert transform can be used for time-frequency analysis of functions or signals. Unfortunately, the Hilbert transform can only be effectively applied to an extremely small class of signals, namely those characterized by a single frequency component at any instant in time. The recently-developed Hilbert–Huang algorithm addresses the limitations of the classical Hilbert transform through a process known as empirical mode decomposition (EMD). Using this approach, the data is filtered into a series of intrinsic mode functions (IMFs), each of which admits a well-behaved Hilbert transform. In this manner, the Hilbert–Huang algorithm affords time-frequency analysis of a large class of signals. The algorithm has been applied in the analysis of scientific data, structural system identification, mechanical system fault detection, and even image processing. This research demonstrates the applications of the Hilbert–Huang algorithm for the analysis of aeroelastic systems, with improvements such as localized–on-line processing. Applications for correlations between system input and output, and among output sensors, are used to characterize the time-varying amplitude and frequency correlations present in the various components of multiple data channels. Online stability analyses and modal identification are other objectives. Example demonstrations include using aeroelastic test data from the F-18 Active Aeroelastic Wing (AAW) airplane, an Aerostructures Test Wing, and pitch–plunge simulation.

Objective

With the HHT, the IMFs yield instantaneous frequencies as functions of time that give sharp identification of imbedded structures. The main conceptual innovation in this approach is the introduction of the instantaneous frequencies for complicated data sets, which eliminates the need for spurious harmonics to represent nonlinear and nonstationary signals. This research looks at the effect of enhancements like local–on-line versions of the algorithm. To date, HHT analysis has only been performed on individual signals without regard to correlation with other data channels or system inputs-to-outputs. Applications for correlations between system signals are used to characterize the time-varying amplitude and frequency modulations present in the various components of multiple data channels, including input and distributed sensors. In these respects, this research elucidates the way EMD behaves in the analysis of general aeroelastic and aeroservoelastic flight test data.

Approach Example

Local–On-line Decompositions

An aileron command multisine input used on the F-18 AAW is shown in the figure using the standard EMD [fig. 1(a)] and local–on-line version [fig. 1(b)]. The bandpass nature of IMFs is reflected in the three standard IMF mean frequencies, {23.9, 16.3, 7.7} Hz for each of the IMFs {#1, #2, #3}, respectively, and online corresponding IMF mean frequencies {23.6, 13.0, 7.2} Hz. Immediately noticeable is the more efficient extraction of the signal components by the local–on-line algorithm, most evident by the sparse second and third IMFs (imf2 and imf3) being more sparse than the corresponding standard IMFs, and less residual. Besides the obvious advantage of an online algorithm for decomposing data, it also clearly surpasses the standard algorithm in terms of computational burden, especially with long original data records. An added bonus is that it generally has better orthogonality properties among the IMFs.

Status

System input–output signal analysis characterizes the time-varying amplitude and frequency components of multiple data channels, including input-to-output and distributed sensors, in terms of the IMFs of the HHT. These are significant departures from Fourier and other time-frequency or time-scale wavelet approaches. Online stability analyses and modal identification are also possible. System identification in the IMF subcomponent environment is a practical endeavor in the context of multiresolution system identification. Exploiting local properties for signal analysis applies to spatial data as well as temporal data with frequency and scale variations for general space-time-frequency-scale signal processing.

Modern intelligent control and integrated aerostructures require control feedback signal processing cognizant of system stability and health. Time-varying linear or nonlinear modal characteristics derived from flight data are within the realm of the HHT. Further research will investigate these issues and HHT connections between localized instantaneous dynamics, health diagnostics, and global system stability and performance for monitoring and prediction.

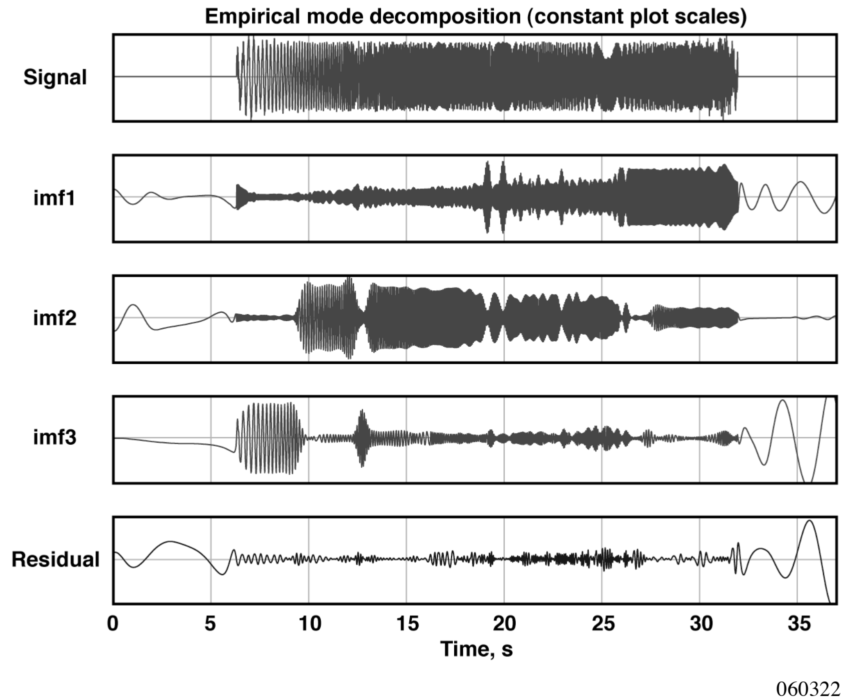

Figure 1(a) Standard empirical mode decomposition.

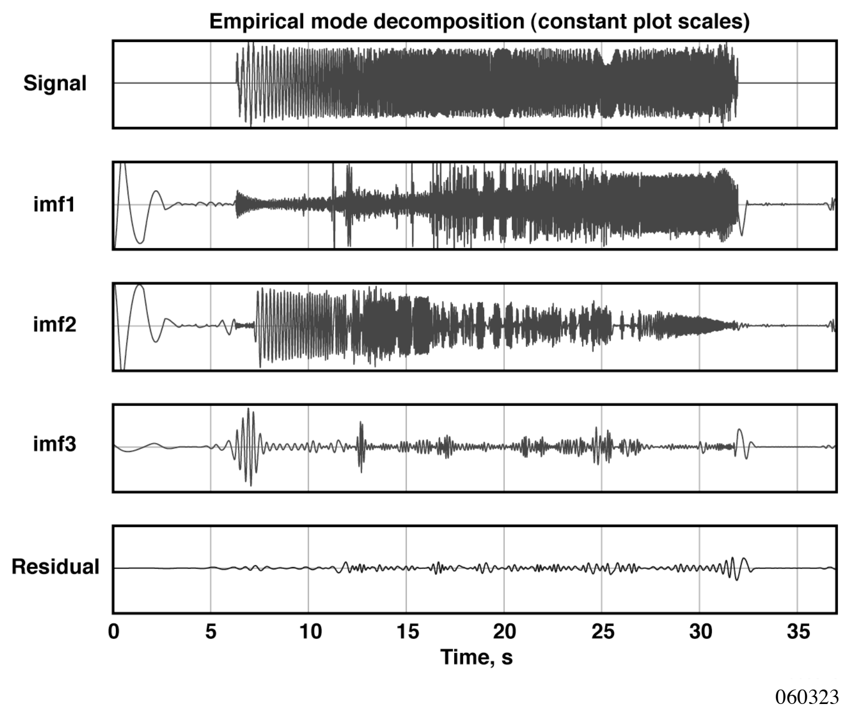

Figure 1(b) Local–on-line empirical mode decomposition.

Figure 1. Standard (a) and local–on-line (b) empirical mode decomposition of an F/A-18 active aeroelastic wing multisine aileron command input.

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A LEAST ABSOLUTE SHRINKAGE AND SELECTION OPERATOR (LASSO) FOR NONLINEAR SYSTEM IDENTIFICATION

Summary

Identification of parametric nonlinear models involves estimating unknown parameters and detecting the underlying structure. Structure computation is concerned with selecting a subset of parameters to give a parsimonious description of the system that may afford greater insight into the functionality of the system or a simpler controller design. In this research, a least absolute shrinkage and selection operator (LASSO) technique is investigated for computing efficient model descriptions of nonlinear systems. The LASSO minimizes the residual sum of squares by the addition of a 1-norm penalty term on the parameter vector of the traditional 2-norm minimization problem. Use of the LASSO for structure detection is a natural extension of this constrained minimization approach to pseudolinear regression problems, which produces some model parameters that are exactly zero and, therefore, yields a parsimonious system description. The performance of this LASSO structure detection method was evaluated by using it to estimate the structure of a nonlinear polynomial model. The applicability of the method to more complex systems was shown by identifying a parsimonious system description of the F/A-18 (McDonnell Douglas Corporation, St. Louis Missouri and Northrop Corporation, Newbury Park, California) Active Aeroelastic Wing (AAW) from flight test data.

Objective

Discrete-time nonlinear polynomials are often useful to describe the input-output behavior of complex systems encountered in many control engineering, aerospace engineering, and biological science applications. These polynomial mappings describe the dynamic relationship of a system by expanding the present output value in terms of present and past values of the input signal and past values of the output signal. These models are popularly known as polynomial Nonlinear AutoRegressive, Moving Average eXogenous (NARMAX) models. Many systems are described by these polynomial models having only a few terms. Even if the system order is known through some a priori knowledge, a full expansion of this model representation yields a large number of candidate terms that may be required to represent the system dynamics. Often many of these candidate terms are insignificant and, therefore, can be removed. Hence, the structure detection problem is that of selecting a subset of candidate terms that best predicts the output while maintaining an efficient system description.

The relevance of structure computation is, for example, controller design and study of aerospace vehicle dynamics. For control, a parsimonious system description is essential for many control strategies. In modeling, the objective is often to gain insight into the function of the underlying system. Recently, a bootstrap method has been proposed to solve the structure detection problem for over-parameterized models. Although it has been demonstrated that the bootstrap is a useful tool for structure detection, there is a limitation with this technique. As a result of the large number of candidate terms for a given model order, limited data records available for any practical identification problem, and the data length required to guarantee convergence, it may not be feasible to analyze highly complex systems with the bootstrap technique.

Approach

Nonlinear aeroelastic dynamics of aircraft are highly complex processes likely involving a large number of candidate terms which may not be accurately characterized by current approaches.

The application of a novel method for NARMAX model identification via a least absolute shrinkage and selection operator (LASSO) is being researched. This approach permits identification of NARMAX models in situations in which current methods cannot be applied. In an over-parameterized polynomial NARMAX model, LASSO yields good results for structure detection in the presence of additive output noise. Application of structure computation to aeroelastic modeling is performed using flight test data from the F/A-18 AAW airplane. It has been shown that LASSO yields a parsimonious model structure whilst maintaining a high percent fit to cross-validation data.

The figure shows the predicted output for a cross-validation data set for the identified structure. The upper panel displays the full 26 s time history of the accelerometer response recorded on the left wing. The lower panel displays a 4.5 s slice of the predicted output superimposed on top of the measured output. The predicted output accounts for over 98 percent of the variance of the measured outputs. The result demonstrates that the computed model structure is capable of reproducing the measured output with high accuracy.

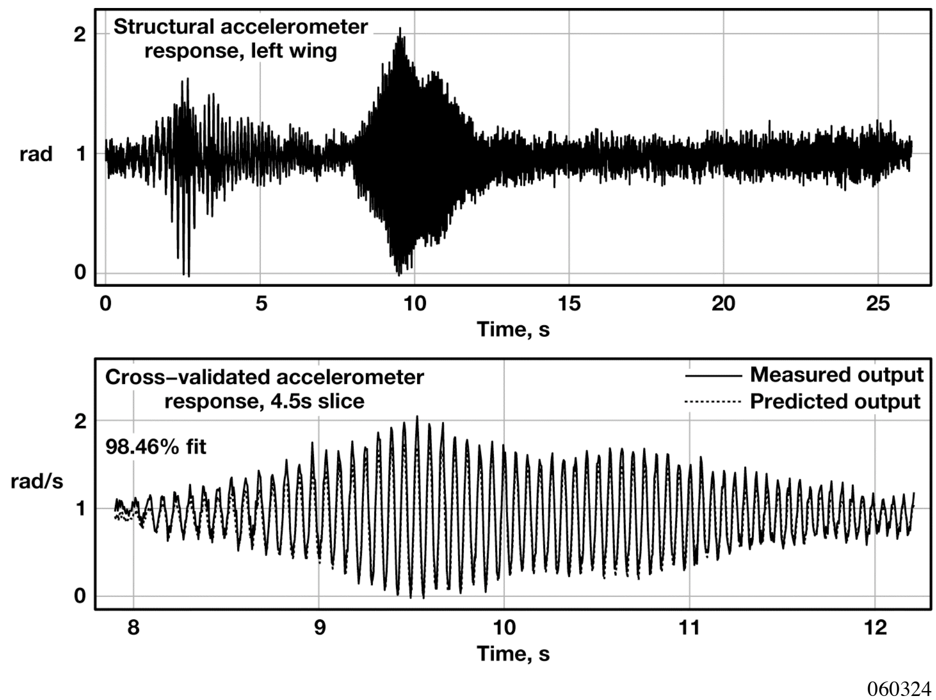

Figure 1. Accelerometer responses.

Status

The LASSO technique is a novel approach for detecting the structure of over-parameterized nonlinear models in situations where other methods may be inadequate, and is clearly amenable to the study of a wide range of nonlinear aerospace systems.

The optimization criteria in the LASSO setup is motivated by the well-known fact that a 1-norm penalty appended to a quadratic objective tends to yield a sparse solution. It is, however, only a heuristic for addressing the underlying problem: achieving few nonzero parameters. An alternative way to address this is to use combinatorial optimization.

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TECH BRIEFS AND PATENTS

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2005 Tech Briefs Published:

Les Gong

Real-Time Simulation of Aeroheating the Hyper-X Airplane

DRC-098-076

2005 Patent Issued:

Patent Issued- **May 17 2005**: DRC-099-002

Titled: "A Method For Reducing The Drag Of Blunt-Based Vehicles By Adaptively Increasing Forebody Roughness"

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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.

15. SUBJECT TERMS

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