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2000–2001 Research Engineering Annual Report

Compiled by
J. Larry Crawford and Everlyn Cruciani
NASA Dryden Flight Research Center
Edwards, California

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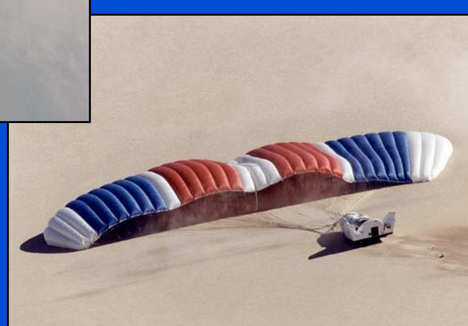
Edwards, California

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Preface

The NASA Dryden Flight Research Center's Research Engineering Directorate is a diverse and broad-based organization composed of the many disciplinary skills required to successfully conduct flight research. The directorate is comprised of six branches representing the principal disciplines of: Aerodynamics, Controls and Dynamics, Flight Systems, Flight Instrumentation, Propulsion and Performance, and Aerostructures. The Directorate organization is illustrated on the chart following this page.

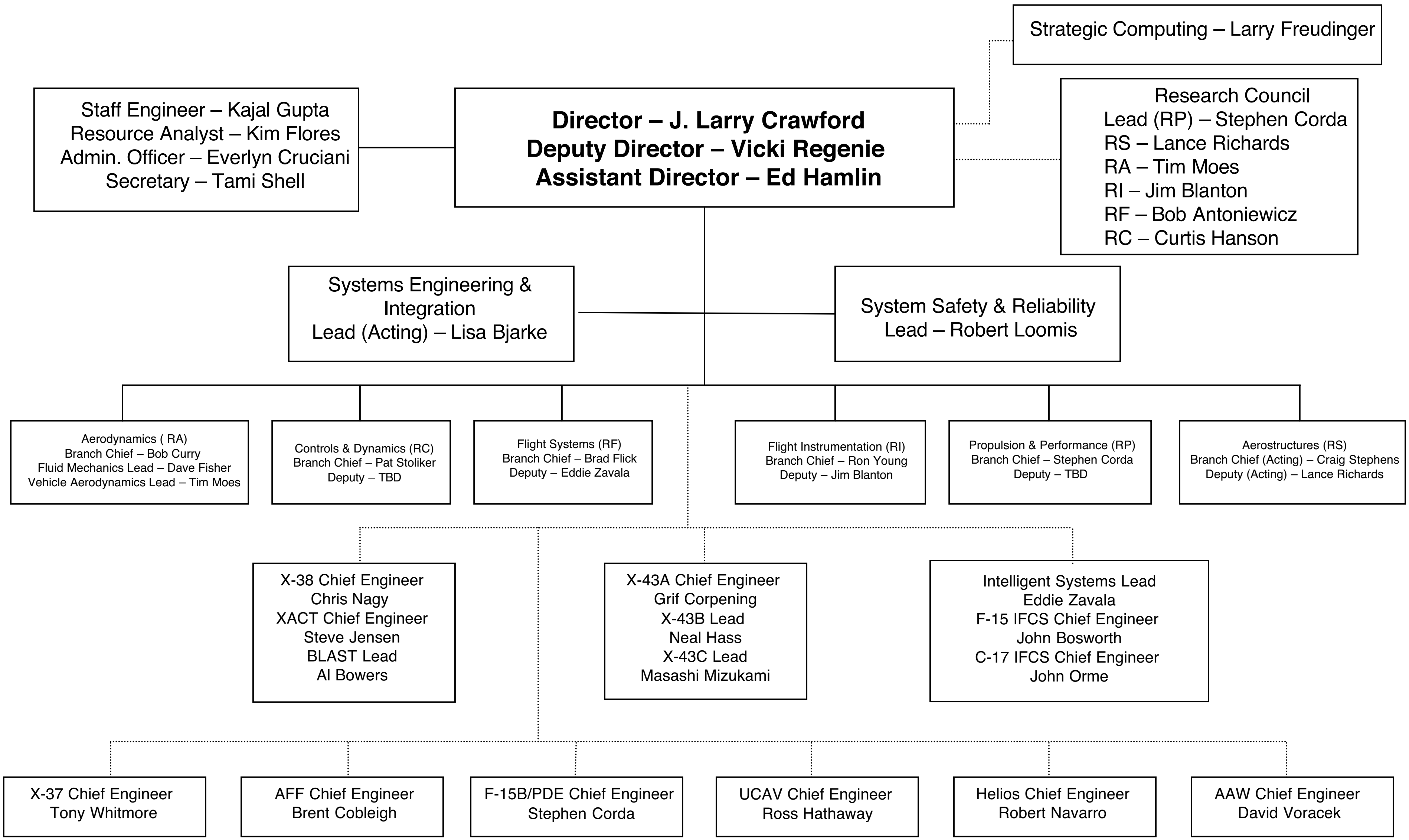
We are very proud of the many significant accomplishments of our technical staff during the calendar years 2000–2001. These milestones include both those accomplished in support of research programs as well as basic research performed within the Directorate, supported by our competitively-funded Flight Test Techniques and Disciplinary Flight Research programs.

This Annual Report encompasses the full range of research accomplishments, from the project level, flight test techniques, to disciplinary flight research programs. It includes one-page summaries of each program; more details are available from the principal investigators as noted on the summaries. There are also included a list of the many technical publications completed in the last year, from in-house, university, and contract researchers under the auspices of the Directorate.

Calendar year 2002 promises to be an even more productive year, with a mix of new and continuing research programs. I look forward to reporting on these efforts next year.

J. Larry Crawford
Director of Research Engineering
Dryden Flight Research Center

Research Engineering Directorate (Code R)



J. Larry Crawford, Director
Research Engineering Directorate

5-18-01

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Director	J. Larry Crawford
Deputy Director	Vicki Regenie
Assistant Director	Edmund Hamlin
Administrative Officer	Everlyn Cruciani

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RA – Aerodynamics	Robert Curry
RC – Controls and Dynamics	Patrick Stoliker
RF – Flight Systems	Brad Flick
RI – Flight Instrumentation	Ronald Young
RP – Propulsion and Performance	Stephen Corda
RS – Aerostructures	Mike Kehoe

2001 Research Engineering Directorate Staff

Director	J. Larry Crawford
Deputy Director	Vicki Regenie
Assistant Director	Larry Montoya
Administrative Officer	Everlyn Cruciani

<u>Branch Codes and Chiefs</u>

RA – Aerodynamics	Al Bowers
RC – Controls and Dynamics	Patrick Stoliker
RF – Flight Systems	Bob Antoniewicz
RI – Flight Instrumentation	Ronald Young
RP – Propulsion and Performance	David Lux
RS – Aerostructures	Steve Thornton

Low-Speed/High Reynolds Number Ground Research Vehicle

Summary
A ground research vehicle (GRV) has been built for use as a low-speed, high-Reynolds-number testbed (Figure 1). The current use of the GRV is to study base drag, which is a major component of drag for objects with truncated base areas, such as reusable launch vehicles.

Objective
Hoerner and Saltzman have demonstrated a direct relationship between viscous forebody drag and base drag (Figure 2). This curve indicates that total drag can be minimized by varying skin friction. The GRV is currently being used to investigate this concept by obtaining extensive drag data for various vehicle configurations.

Approach
The GRV was fabricated by modifying an existing GMC van. The vehicle was lengthened and a metal frame was constructed around it. Flat panels were attached to this frame to create the outer surface of the vehicle.

The primary instrumentation system on the GRV consists of 125 pressure ports located on the vehicle's surface. The ports are connected to differential pressure transducers, which are linked to a laptop computer via ethernet. The data are recorded using LabView and saved for post-processing, where they will be used to compute drag values.

The GRV is also instrumented with two GPS devices. An Ashtech GPS unit is used to record speed and position information, which can be used post-drive to compute total drag. A handheld GPS is used in conjunction with the LabView software to provide real-time data. It also serves as the GRV's speedometer.

The primary test "maneuvers" are coastdowns and constant-speed tests, and are performed on an Edwards AFB runway.

Status
Several ground tests have been completed on the baseline, highest-drag configuration. The instrumentation system continues to be refined.

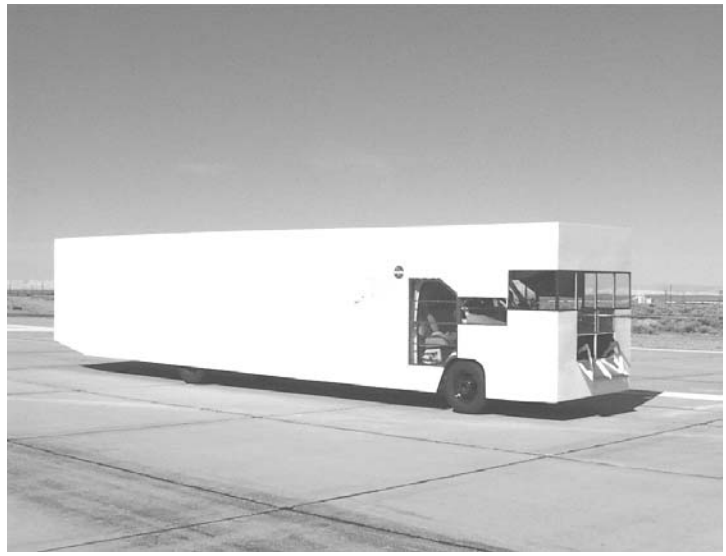

Figure 1: The Ground Research

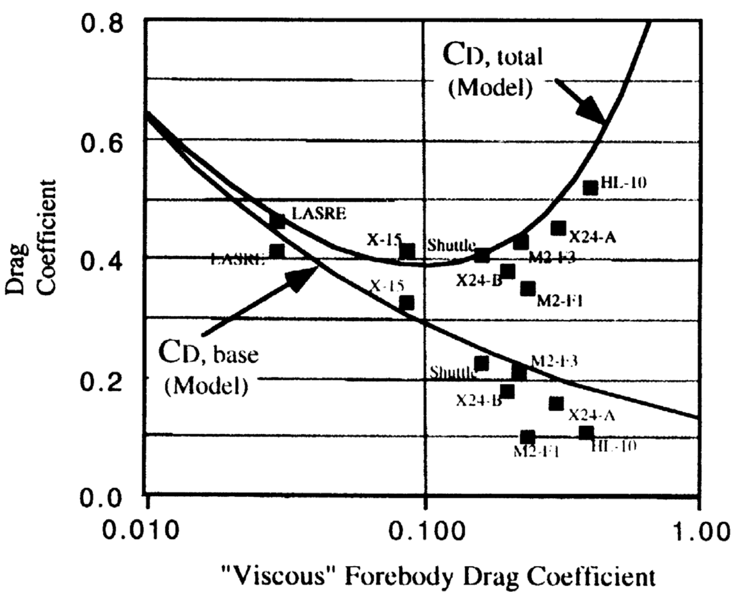

Figure 2

Reference
Saltzman, et. al., *Flight Determined Subsonic Lift and Drag Characteristics of Seven Blunt-Based Lifting-Body and Wing-Body Reentry Vehicle Configurations*, AIAA Paper #99-0383, 1999.

Contacts
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Hot Wire Anemometry System

Summary

A hot wire anemometry system has been developed to measure turbulence in aircraft boundary layers. The system consists of hot wires and a positioning system, an anemometry system, and a data acquisition system. The system was flight tested on an F-15B aircraft. The system was mounted on a flight test fixture, which is located below the F-15B aircraft fuselage.

Objectives

The objective was to demonstrate that hot wire anemometry could effectively measure in-flight boundary layer turbulence. The key was in obtaining a high measurement signal-to-noise ratio.

Justification

Boundary layer turbulence is the largest research subject in aerodynamics. There is much to be learned about turbulence. Turbulent flight data is essential to further our understanding of turbulence and to develop better turbulent prediction models for future aircraft design. To date, NASA has not developed an instrument to measure boundary layer turbulence in-flight. This project serves to fill that void.

Approach

A constant voltage anemometry system (CVA) was developed specifically to measure in-flight boundary layer turbulence. The CVA system is much less susceptible to noise from aircraft power systems and radio frequency interference.

Results

Figure 1 presents a sample of the hot wire anemometry data obtained from the flight tests. The figure shows the power spectral density of the turbulence data taken at 1, 2, 3, and 4 cm from the flight test fixture surface. The boundary layer thickness is 3 cm. The figure shows that there is a significant amount of turbulence spectra in the boundary layer at 1 and 2 cm from the surface. A minimal amount of spectra exits at 3 cm from the surface, the edge of the turbulent boundary layer. The peak spectra occur at about 1000 Hz. The turbulent energy cascade is clearly shown above 12 kHz by the constant negative 5/4 slope. The data was sampled at 50 kHz and low-pass filtered at 20 kHz.

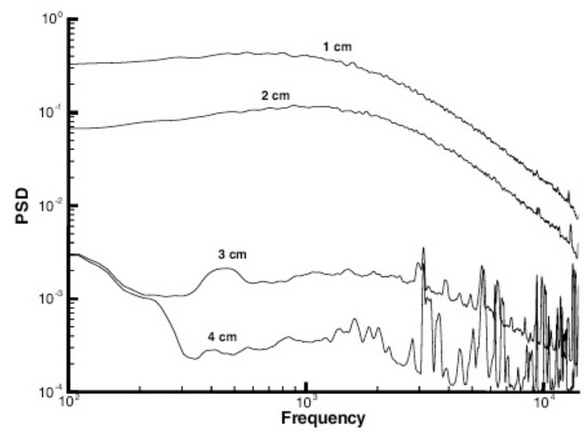

Figure 1. Power spectral density for a 5-micron hot wire at 1, 2, 3, and 4 cm from surface. Flight condition is 0.5 Mach, 15,000 ft altitude, 15.3 million Reynolds number.

Status

The flight tests have been completed. The signal-to-noise ratio was excellent (greater than 20). Hot wire anemometry has been demonstrated to effectively measure in-flight boundary layer turbulence.

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Stratospheric Mountain Wave Flight Research

INTRODUCTION
Stratospheric Mountain Waves (SMW) are the atmospheric phenomena forming the basis for this project. They are defined here as mountain waves that propagate strongly, and with continuity into the middle and upper stratosphere (>70Kft), and are not extinguished, trapped, or reflected at or near the tropopause (fig 1). Because of the many atmospheric waves discussed in the meteorological literature, the specific nomenclature "Stratospheric Mountain Waves" is used here to avoid confusion. The historical experience of high-flying aircraft is generally in the lower region of their domain. Amplification with increasing altitude and the instability caused by over amplification can lead to "overturning" or breaking waves. These breaking waves and the associated turbulence are significant to the future of high altitude air traffic flying in the regions of mountain waves.

The ability of an aircraft to seek and sample the meteorological parameters of a complete mountain wave from its lowest levels would provide significant boost to the understanding of the processes that lead to the origination, growth, and the breaking of these waves. Waves propagate upward with increasing strength, generally maintaining a constant energy, defined as; air-density x vertical-wind-component squared. The combination of circumstances necessary for the formation of stratospheric mountain waves in the Northern Hemisphere is most commonly found around the 60 to 70 degree latitude near the outer region of the polar vortex. In the Southern Hemisphere they extend further toward the equator, because of the larger extent of the polar vortex in that hemisphere. Mountain wave propagation is more often limited by decreasing or reversing wind with altitude.

OBJECTIVES:
Currently a study is underway to verify two principal assertions as to the feasibility of the high-altitude mountain-wave flight project. First is to verify the strength, location, structure, and frequency of occurrence of strong mountain waves. Secondly, to verify that the aerodynamic performance of a sailplane will enable it to climb to 100,000 feet in the waves.

APPROACH:
The meteorological approach is to search and identify existing historical sources of mountain wave data from U.S. and foreign countries and from high altitude research organizations. Additional evidence of SMW confirmation will be to examine these data for large temperature variations together with strong winds within the data from balloon borne ascents.

Mountain-wave numerical modeling will be conducted for selected cases using dedicated balloon ascents and other wave data available. Numerical model results will be added to simulation to estimate the static elastic aerodynamic coefficients of the desired sailplane. The resulting model will be used in batch simulation and piloted simulation to assess the flying qualities. The direct effect of wave structure on sailplane control can be shown. Variations of the parameters will be made to determine the most attractive combination of aerodynamic stability, and augmentation. For this study only generic actuator and sensor dynamics, and control laws will be used.

STATUS:
Plans are currently underway to obtain additional upper-air data from countries known to have SMW observed. In addition, a small research proposal is currently under consideration by NASA to conduct phase 1 flight tests to 40 kft altitude in the local Edwards environment beginning Spring 2001. If successful, additional phase 1 flights to 62,000 feet in New Zealand could be conducted. New Zealand flights would prove this concept of penetrating the tropopause and flying into the stratosphere.

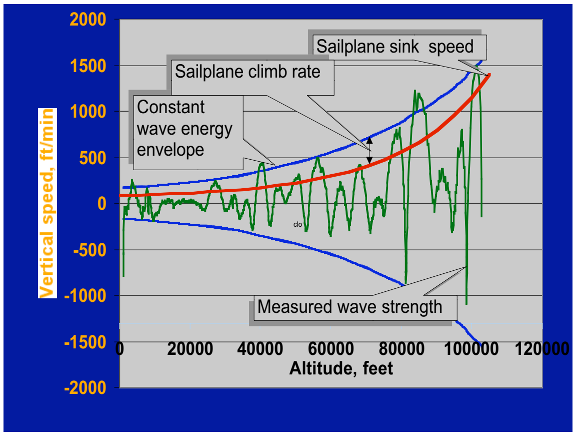

Fig 1. Energy curves for wave amplitude and aircraft sink speed

Contacts
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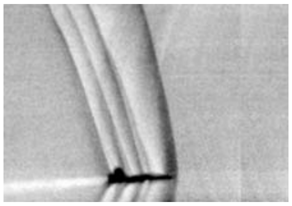
Cockpit Guidance And Camera Control Software For Air-To-Air Schlieren Imaging

Summary

High resolution images of shockwave patterns on full-scale aircraft are to be taken using air-to-air schlieren photography. This technique requires precise positioning of two aircraft in relation to the Sun, which is used for illumination. The two aircraft also have a great differential speed, one being supersonic, and the other is at 250 knots. The time-delay-integration, TDI, camera used to capture the schlieren images requires accurate line-scan rates to avoid blurring. Matlab-based software was developed to determine from GPS sensors the relative aircraft position and velocity as well as Sun aspect, display trajectory guidance information to the pilot with the schlieren camera, control camera operation, and record all trajectory and image data on a small personal computer. The computer is flight-ruggedized to 5.4 Gs with a size of about 5 x 8 x 8 inches.



Ground-to-Air Schlieren of T-38. Air-To-Air Images Will Have About One Inch Resolution

By utilizing commercial off-the-shelf software and hardware for data importation, graphics generation, and camera control, significant cost and time savings was achieved. Most of the algorithms were adapted from existing in-house engineering code. This capability for relative positioning determination and pilot guidance may be used on other NASA Dryden flight projects. The Autonomous Formation Flight program will use this software as a basis for an independent measurement of relative position.

Objective:

Provide pilot guidance and camera control to obtain high resolution schlieren images of full-scale aircraft supersonic flow.

Approach:

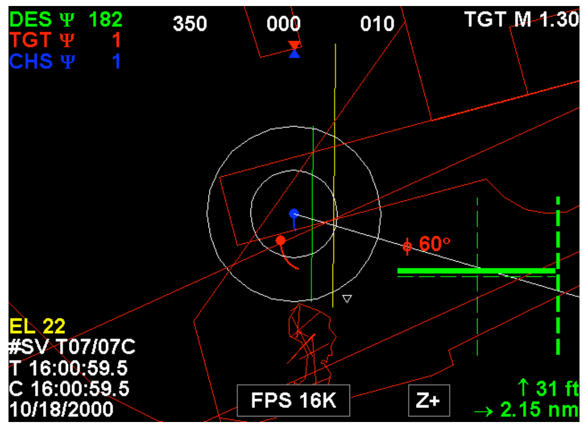
A two-seat F-18 is configured to take the schlieren images. The Schlieren computer is located in the rear cockpit, and the Schlieren camera and optics is housed in a Forward Looking InfraRed (FLIR) pod. Six inch monitors are in both front and rear cockpits. The computer is running Windows NT, Matlab from Mathworks, Extended Real Time Toolbox from Humusoft, and XCAP imaging software and a PIXCI-D frame grabber board from EPIX.

This new schlieren software takes serial GPS data from both aircraft and computes relative aircraft position and velocity as well as angles to the Sun. Computations are made for the point where the two aircraft and the Sun are colinear, and computes the proper linescan rate for the camera. A moving map showing both aircraft, local landmarks, and restricted airspaces guide the F-18 to the proper flight track to take the schlieren image. This display includes a scrolling heading indication for both aircraft as well as the desired heading. Inset guidance needles shows the offset to the proper flight track and indicates the required bank angle to capture the track from the downwind leg. The operator can select different flight path separations and map scaling via a graphical user interface or key press.

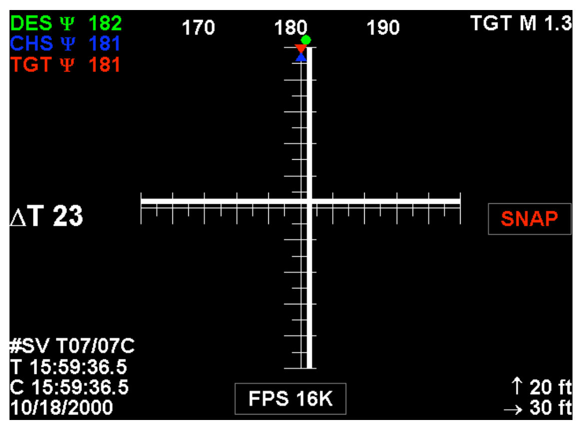
When close to the imaging position, the display changes to large terminal pilot guidance needles, including a time-to-go indication. When in the proper position, the software commands the camera to scan at the proper rate and saves the image to non-volatile memory. The operator can also manually snap an image via a graphical user interface or key press. The schlieren image is displayed to the pilot a few seconds after it is taken. All GPS data from both aircraft are saved for postflight analysis.

Results:

Testing in two moving ground vehicles showed accuracy of about 2 feet, showed latency of about 0.3 seconds, and properly commanded the TDI camera.



Moving Map Pilot Guidance



Terminal Pilot Guidance Needles

Contacts:

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David Richwine, NASA Dryden, P, X-2170

Comparison of Wind Measuring Systems

Introduction:
A preliminary field test of wind accuracy compared radio wind sounding (rawinsonde) balloons, Global Positioning System (GPS) and Radio Automatic Theodolite Sounder (RATS) rawinsondes, to a Range Instrumention Radar (RIR)-716 tracked reflector. Precision radio detection and ranging (radar) tracking of a Jimsphere balloon or reflector has been used as a method for high accuracy wind velocity measurement in support of special aerospace range operations such as the space shuttle program. Recently, programs at the (NASA) Dryden Flight Research Center (DFRC) such as the Future-X (X-40), and other Uninhabited Aerial Vehicles (UAV's) have mandated requirements for high accuracy atmospheric wind data with rapid updates.

Objectives:
Conventional rawinsondes, including the RATS do not have as much accuracy and resolution as provided by precision radar tracking. When radar is used to track the balloon it is often difficult to obtain radar lock, particularly in the presence of ground clutter at lower altitudes when the balloon is released at a mission location some distance from the radar. Demonstration that the GPS rawinsonde system tracking precision is comparable to range radar would allow GPS systems to support range wind measurement requirements, thereby circumventing the ground clutter problem and freeing the range tracking radar to other missions while saving costs that would otherwise be spend on radar costs.

Approach:
Upper-air wind measurement systems comparison tests were conducted by the United States Air Force (USAF) and NASA Dryden Flight Research Center (DFRC) at Edwards Air Force Base (EAFB), Edwards, California. The two radio wind sounding (rawinsonde) packages and the 6-inch spherical radar reflector were placed on the same balloon tether line in order to sample the same atmosphere. Test days were selected at random based on NASA and Air Force range schedules and not on meteorological conditions.

Results:
Figures 1 and 2 show the observed wind speed and direction verses altitude for the November 3, 1999 sondes. All three systems show general agreement in velocity and direction. Missing data and large differences of wind velocity and direction are due to loss in radar tracking of the reflector and light winds. In contrast to the radar, both rawinsonde systems maintained track continuously from near ground level to altitude. Some comparison of the the numbers are as follows: wind velocity RMS of the radar/GPS system was within 1.09 kts (0.56 m/s) while the wind velocity RMS of the radar/RATS was more than quadruple that at 4.98 (2.56 m/s). Similarly, RMS direction differences for the radar/GPS was 7.23 and 13.99 degrees for the radar/RATS. During these test the RATS and GPS data was available almost immediately while it was several hours until the radar data was available for processing.

Conclusion:
These limited tests have shown that the use of GPS rawinsondes has the potential to provide researchers and planners increased wind accuracy and resolution over conventional rawinsonde tracking systems. The present test results indicate that the GPS sonde system provides cost-effective fast response wind data profiles with accuracy comparable to the radar-tracked reflectors.

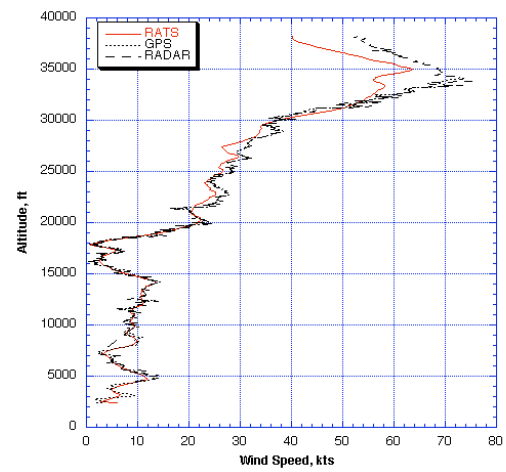

Fig 1 Wind speed comparison

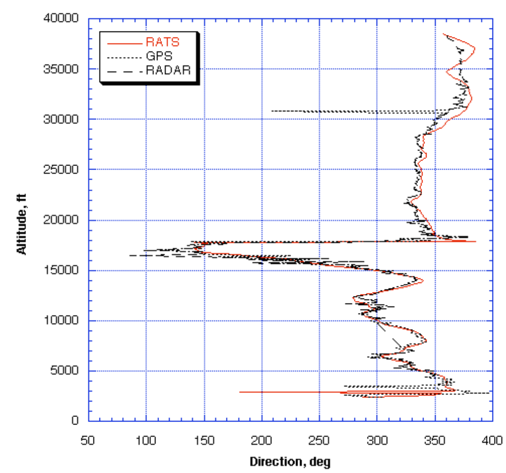

Fig 2 Wind direction comparison

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Linear analysis of the X-33

Program Overview

The X-33 program will demonstrate new technologies required for a Reusable Launch Vehicle (RLV) using a half-scale prototype. The X-33 will be an unmanned vehicle, launched vertically, reaching an altitude of over 200,000 feet at speeds approaching Mach 10. The vehicle will operate autonomously from launch to landing. Some of the technologies to be demonstrated are: metallic thermal protection system, linear aerospike engines, and an integrated vehicle health monitoring system. NASA Dryden is supporting this activity through analysis of the stability and control of the vehicle.

Approach and Results:

The X-33 Integrated Test Facility (ITF) 6 degree-of-freedom nonlinear simulation was used to generate linear models of the X-33 vehicle. Various X-33 specific models which make up the simulation (aerodynamic, engine, actuator, reaction control system, etc) were built from either wind tunnel tests or from computational models which have been validated with component tests. A finite difference method was used to approximate the coefficients in linear equations of motion.

Linear perturbation models of rigid body equations of motion in the form of

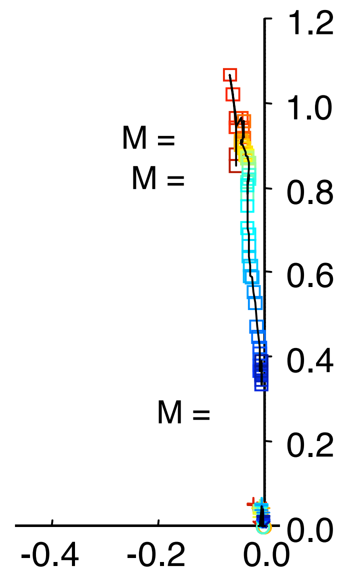
$$\dot{x} = Ax + Bu$$
$$y = Hx + Fu$$

Matrix coefficients were obtained by numerical perturbation of the states and control effectors about each predefined flight condition.

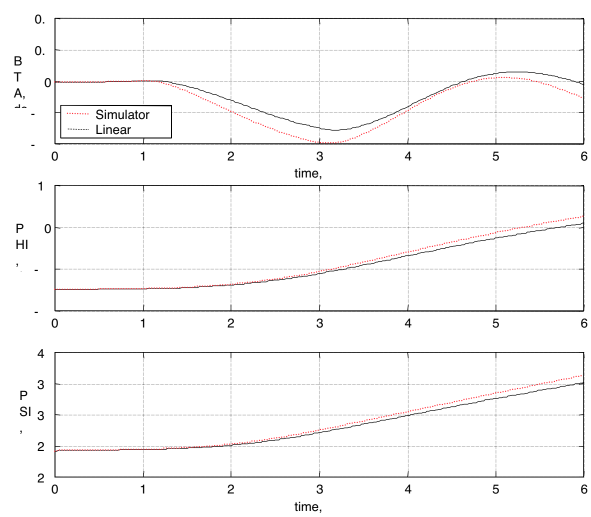
This dynamic model includes the eight control surfaces (body flaps, rudders, inboard and outboard elevons), the reaction control system jets, and the engine thrust vectoring.

The linear model generator in the ITF simulation has the capability to produced linear models of the equations of motion at a user defined flight conditions along the vehicle trajectory. Linear models can be generated along the flight path based on Mach number, altitude, and time. This allows the X-33 team to develop root contour plots such as the one shown. The root contour plots help to identify critical conditions along the trajectory which should be analyzed further.

The linear models generated by the ITF simulation are then used for stability and control analysis. Since the vehicle is not trimmed, the rates of the vehicle are non-zero. For time domain comparison the initial rates of the vehicle were taken into account. The figure shows a comparison of open loop linear model and the nonlinear simulation for a RCS yaw pulse during entry. Preliminary closed loop time domain and frequency domain analysis has been preformed for the TAEM region of flight. Future work includes closed loop time domain and frequency comparisons with the nonlinear simulation for ascent, transition, entry flight phases.



Longitudinal Root Contour for Entry Phase



Open-loop Comparison of Nonlinear Simulation and Linear Model for a RCS Yaw Pulse

Contact: Cathy Bahm, NASA Dryden, RC, x-3123, Bob Clarke, NASA Dryden, RC, x-3799, John Burken, NASA Dryden, RC, x-3307

Monte Carlo Analysis of the X-33 simulation

Summary
The primary goal of the Monte Carlo testing of the simulation is to establish mission success probability. The secondary goal is to determine the effects of various dispersions on the vehicle's dynamics and performance. The analysis can eventually serve as an end-to-end check for validation and verification.

Objective
The objective of performing a Monte Carlo Analysis on the simulation software is to examine effects of uncertainties in the vehicle dynamics, system and external environment on the X-33 flight vehicle.

Justification
- proves robustness of vehicle
- complements (linear) frequency domain analysis of vehicle stability
- provides time-domain traces of engineering parameters
- provides dispersion envelopes

Benefits
- identification of critical dispersions
- identification of control and guidance problem areas
- ability to bound dispersed trajectory for ground control

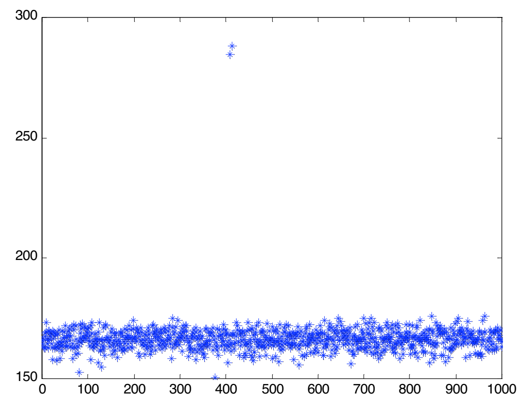
Approach
The main question associated with Monte Carlo analysis is determining the number of trials that need to be performed before the statistics of a variable can be estimated with reasonable accuracy. Once the desired number of runs is determined, files containing all relevant dispersions are generated. Dispersion values are varied in a normal random distribution, and then stored in individual input files. This collection of files is then run sequentially from a main script, which directs the storage of relevant data.

Additional scripts have been written to process the data for analysis. These processing scripts have the ability to plot variables at certain flight conditions, plot values for individual dispersions, and plot values for all dispersions for a particular Monte Carlo run.
From these analysis plots, determinations can then be made regarding the effects of each dispersion on the vehicle, as well as the effects of the group of dispersions as a whole on a particular run.

Selected Results
The Monte Carlo analysis techniques used in this analysis allowed the identification of several control and guidance problems in the software.

The analysis also allowed the identification of critical dispersion parameters. From the analysis it was apparent that drag dispersions were very critical to the vehicle.

In some cases, a single dispersion parameter may not be identifiable as the critical dispersion parameter. In those cases, an examination of general parameters, such as equivalent airspeed at touchdown, can show which combinations of dispersions may be critical. A plot of equivalent airspeed at touchdown shows that 2 of the 1000 cases run had combinations of dispersions which caused the simulation to exceed limits. This is shown in the figure below.



Status
Monte Carlo analysis has been performed on a preliminary build of the X-33 flight simulation software.

Further analysis on future simulation software will be conducted, along with additional Monte Carlo analysis related to the reconfigurable flight control system on the X-33.

Contact:
Peggy Williams, NASA Dryden, RC, X-2508

Application of CONDUIT to the Active Aeroelastic Wing

Summary

The Active Aeroelastic Wing (AAW) program is investigating the issues of controlling an aircraft with an aeroelastic wing. The control laws for the aircraft were developed using a process developed by Boeing, called Integrated Structural Maneuver Design (ISMD). This process proposes to minimize loading of the wing root and wing fold throughout any given maneuver.

An investigation into an alternate design method was conducted, using a design program called CONDUIT (Control Designer's Unified Interface), which was developed by the Army/NASA Rotorcraft Division at the NASA Ames Research Center. It is a control system design tool that uses a multi-objective function optimization to tune selected control system design parameters.

Objective

The objective of this study was to re-design the control system using CONDUIT to maximize the roll performance of the aircraft, while maintaining adequate stability and loads margins. In doing this investigation, CONDUIT could be evaluated as a possible alternate design method for the AAW project.

Justification

One objective of the AAW project is to develop design tools for future AAW aircraft. This study would demonstrate the applicability of CONDUIT as such a tool. It would also serve as an evaluation tool for checking the compliance of the AAW control laws to handling qualities specifications. Additionally, it would provide results against which the Boeing process could be compared.

Approach

The AAW control laws, designed by Boeing, were modeled in Matlab Simulink and incorporated into CONDUIT. A structural loads model, also created by Boeing, was incorporated into CONDUIT in two specifications, to constrain loads on the wing root and wing fold. Additional specifications were chosen to ensure stability and performance. Through these specifications, the objective function of the optimization was set to maximize performance while maintaining stability and loads margins. The scheduled gains of the ISMD control system were set as design parameters for CONDUIT to optimize.

Results

The final results from CONDUIT for a typical subsonic case are shown below. Each plot is a graphical representation of an optimization constraint or handling qualities specification, where the white, gray, and dark gray regions represent handling qualities of Level 1, Level 2, and Level 3 respectively. In the course of this optimization, the time to bank and maximum achievable roll rate were both improved, at a cost of increased wing root and fold loads. The limiting constraint was the wing fold loads, which were increased to a criticality ratio limit of 0.7 at the Level 2/3 border, arbitrarily set for this investigation. This study successfully showed that CONDUIT could be an effective tool for the AAW project.

Status

Studies are currently being made, using CONDUIT, into the unique gain scheduling requirements of AAW aircraft. The design of the final AAW control laws is currently waiting for improved aerodynamics and loads models, which will be available after the parameter identification flights.

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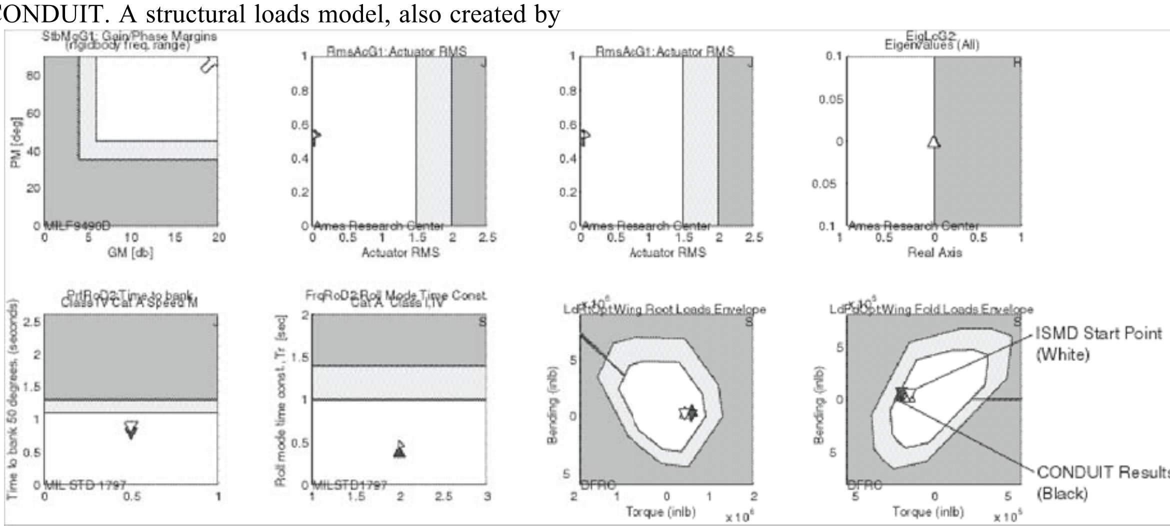

Flight Testing of the MACH Control Laws on theX-38

Summary
The X-38 program was conceived to demonstrate the technology required for the International Space Station's emergency Crew Return Vehicle (CRV). Five prototype vehicles, based on the X-24A lifting body, were developed to demonstrate the CRV concept. One of the X-38 prototypes is designed for space flight testing. It is planned that this vehicle, designated V201, will use the dynamic inversion based Multi-Application Control Law (MACH) developed at the Honeywell Technology Center in Minneapolis, Minnesota. The MACH control laws were recently flight tested on two of the unpowered, subsonic X-38 vehicles at the NASA Dryden Flight Research Center (DFRC). One of these two vehicles, designated V132, was derived directly from the X-24A and scaled to 80% of the size of V201. The second vehicle, designated V131R, was modified from V132 to be an 80% scaled version of V201. The external shapes of V131R and V201 differ from V132 due to space flight considerations.

Objective
The objective is to assess the performance of the MACH control laws in subsonic flight and offer improvements prior to the first flight of the X-38 space flight vehicle.

Justification
In theory, MACH offers an advantage over a classically designed control system because an aerodynamic model is embedded in its algorithm. This allows a full envelope design without the need for point-design gain scheduling and minimal re-design for vehicle shape changes.

Results
The first flight of the MACH control laws was performed on the third flight of V132. The MACH control laws performed well, showing improved angle of attack tracking over V132's previous conventional control system.

The second flight of the MACH control system was performed on V131R (Figure 1). Though the ability to easily adapt itself to new vehicle shapes is one of MACH's main benefits, V131R's less favorable aerodynamics and large aerodynamic uncertainties forced a redesign of the lateral-directional control laws. The redesigned system featured rudder only lateral control, and consequently, significantly degraded roll performance. The first flight of V131R proved interesting. Shortly after release from the B-52, the vehicle experienced an asymmetric yawing moment which was larger than anticipated. The flight control system, which was designed to stress stability over performance, was too low gained to counter the resulting roll rate and the vehicle performed an uncommanded 360-degree roll in flight. The recovery sequence initiated early and the vehicle landed with no notable damage.

Status
After V131R's first flight, a second redesign of the MACH control laws was initiated. The new design places more emphasis on performance and disturbance rejection than the MACH version used for first flight of V131R. Flight testing is expected to resume in May 2001.

Figure 1: X-38 V131R in first free flight.

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X-38 Transonic Rocket Assist (XTRA)

Summary

The X-38 program was conceived to demonstrate the technology required for the International Space Station's emergency Crew Return Vehicle (CRV). Five prototype vehicles, based on the X-24A lifting body, were developed to demonstrate the CRV concept. Currently subsonic, unpowered flight-testing is being conducted with one of these prototypes at the NASA Dryden Flight Research Center (DFRC). In September 1999, a team was formed at the DFRC to study the feasibility of obtaining transonic aerodynamic data by adding propulsion to this vehicle. As a result the X-38 Transonic Rocket Assist (XTRA) configuration was conceived.

Objective

The objective of the study was to determine the technical feasibility of obtaining transonic aerodynamic flight data on the X-38.

Justification

One of the X-38 prototypes is designed for space flight testing. This vehicle will be released from the Space Shuttle payload bay and reenter the earth's atmosphere, covering a Mach number range from M=25.0 through subsonic on its maiden flight. Prior to this release, the X-38 program has no plans to obtain flight data above M=0.80. Historically, wind tunnels have had difficulties in accurately predicting the transonic aerodynamics on lifting bodies. For this reason, transonic flight test data are desired to minimize the aerodynamic uncertainties for the X-38 space flight test vehicle.

Approach

Initial trade studies focused on the effects of rocket design characteristics on data collection potential. A review of commercially available rockets was then conducted to determine which one best matched the optimal characteristics. After selecting a rocket, hardware required for integration with the X-38 was designed, a loads analysis was completed, and an assessment of the required vehicle structural modifications was given. A ballast management system was designed for handling the expected large CG shifts. Extensive simulation studies were performed to determine the best trajectory and rocket positioning for maximizing supersonic flight time while remaining within dynamic pressure constraints. Further simulation work was done to study the effects of thrust dispersions. A thermal analysis was also completed.

Results

The resulting configuration featured two solid rockets, one mounted on each side of the vehicle. At rocket burnout the externally mounted rockets were to be jettisoned. The weight of X-38 required that the rockets be high thrust, and consequently the selected rockets were very large. The rockets were to be mounted with a pitch installation angle (fig. 1) to augment the vehicle lift to aid in minimizing the dynamic pressure. Initially problematic thrust dispersion effects were minimized by use of the yaw installation (fig. 2). Simulation studies predicted approximately 50 seconds of data collection starting at Mach 1.5 through subsonic.

Status

Although the XTRA configuration appeared promising, the concept was rejected - primarily due to concerns about the effects of the large externally mounted rockets on the vehicle aerodynamics. Additional studies continue using a lightweight, 25% scale model of the X-38 with an internal propulsion system. A white paper copy of the XTRA study is available.

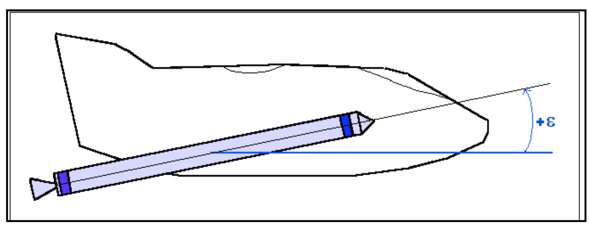

Figure 1: Side View of X-38 in XTRA Configuration

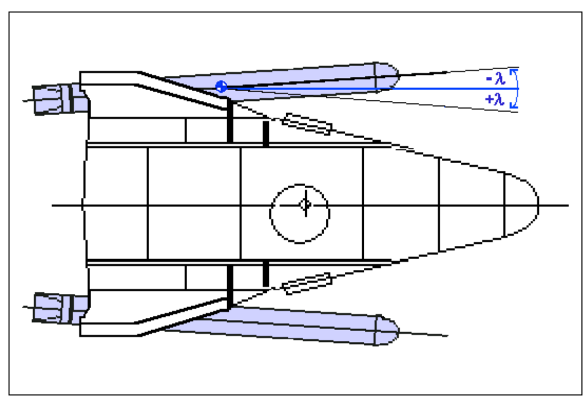

Figure 2: Top View of X-38 in XTRA Configuration

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Ground and Flight Tests of Sub-Scale Inflatable Wings

Summary

Aircraft with inflatable wings are potentially attractive for number of aerospace applications. Recent advances in the design and fabrication of high-pressure inflatable aerospace structures have made a small number of inflatable wings available to researchers at the NASA Dryden Flight Research Center. The inflatable wings were integrated into the design of two sub-scale (15-25 lb), instrumented, research aircraft configurations: a pusher-powered conventional configuration, and an un-powered winged lifting-body configuration.

Objectives

Apply conventional ground and flight test techniques to sub-scale research aircraft in order to gain an understanding of the structural, aerodynamic, and operational characteristics of vehicles with state-of-the-art inflatable wings. In addition, demonstrate in-flight, dynamic (< 1 sec) deployment of inflatable wings.

Approach

Sub-scale testing of inflatable structures is attractive for several reasons. Most ground and flight test operations are greatly simplified when the mass of the test vehicle is low. Vehicle fabrication costs, personnel costs, and test range costs are all reduced with smaller vehicles. Furthermore, the maturation of miniaturized sensor technology, GPS receivers, and micro-controller hardware by the electronics industry has enabled research-quality instrumentation systems onboard sub-scale vehicles with only a modest weight, power, and cost penalty.

Prior to flight with the inflatable wing, each research aircraft was flown with a rigid wing of similar geometry to the inflated wing. Flight operations with the rigid wing were used to test the onboard systems and to practice the required flight test maneuvers prior to commitment to flight with the inflatable wing. Research maneuvers flown with both the rigid wing and the inflatable wing allowed comparison of the trim, aerodynamic performance, and stability and control characteristics of the aircraft in both configurations.

Flight Results

The majority of the flight data collected to date has been with the conventional, pusher-prop configuration fitted with the rigid wing. However, limited comparisons of the vehicle in both a rigid and inflatable configuration are possible. Figure 2 shows a comparison of the vehicle normal-force coefficient versus angle-of-attack for the two configurations. For the limited angle-of-attack range spanned, these data show that the gross lift-generating capability of the inflatable wing is not measurably different from the rigid-wing analog. Takeoff and landing speeds for the two configurations were also similar. Roll trim of the two configurations, however, was measurably different. The inflatable wing set initially flown was (unintentionally) fabricated with a small amount of twist in each wing panel. The resulting rolling moment required approximately 10 of differential horizontal tail deflection to trim out.

Status

Static structural characterization of the inflatable wing has been completed though a series of ground tests. Flight tests have been accomplished using the pusher-prop configuration with both a rigid, and a pre-inflated wing. Construction of the lifting-body vehicle primary structure is complete. Results from initial ground testing of the in-flight, dynamic inflation system are being used in the final system design.

Figure 1 - Static structural testing of the inflatable wing

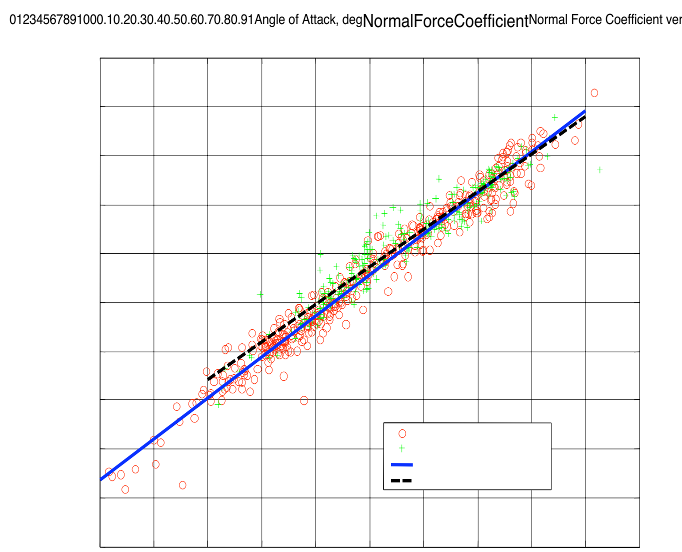

Figure 2 - Comparison of normal force coefficient for a rigid and inflatable wing

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X-34 Tow Tests

Summary

The X-34 vehicle is a technology demonstrator, designed to prove new technologies for future use in second and higher generation Reusable Launch Vehicle programs. Some of the key technologies demonstrated by the X-34 vehicle include primary and secondary composite structures, advanced thermal protection systems, low cost avionics, rapid turn around times, and autonomous flight. The tow tests are a major milestone in preparing for first flight of the X-34.

Objective

In May - July 2000, Orbital conducted tow tests of the X-34 vehicle at Dryden. These tests were designed to demonstrate the vehicle's active steering capability, along with braking effectiveness and navigation capabilities. These tests were not only intended to verify operations for the rollout phase of flight, but also used as end-to-end system integration checks.

Justification

Tow testing was the first operational evaluation of an X-34 vehicle and was intended to increase confidence in the vehicle's systems and programming prior to performing first flight. These tests will verify the functionality of all the major subsystems and interfaces that will be used during a landing rollout.

Benefits
- reduces risk prior to first flight
- verifies system performance
- characterizes controllability and ground navigation
- evaluation of braking system
- gathering of data which will be used to increase the fidelity or models and simulations
- gain operational experience

Approach

Tow Testing was designed to simulate rollout conditions of the X-34 after touchdown. The X-34 was towed by a truck using a 500 foot cable. When the test speed was reached, the X-34 was released. After release, the X-34 performed steering and braking tests.

Results

Tow tests up to 30 mph release speed were completed. These tests demonstrated the successful integration and performance of the vehicle's navigation system, flight computer processing, and steering acuation. It also verified the hydraulic power system response, as well as the correct functioning of the autopilot and flight control hardware interfaces. Telemetry software performed well in both the Mobile Ops Module and the Mission Control Room, in addition to the DGPS UHF uplink working flawlessly. Thermal models were also able to be updated and corrected with test data.

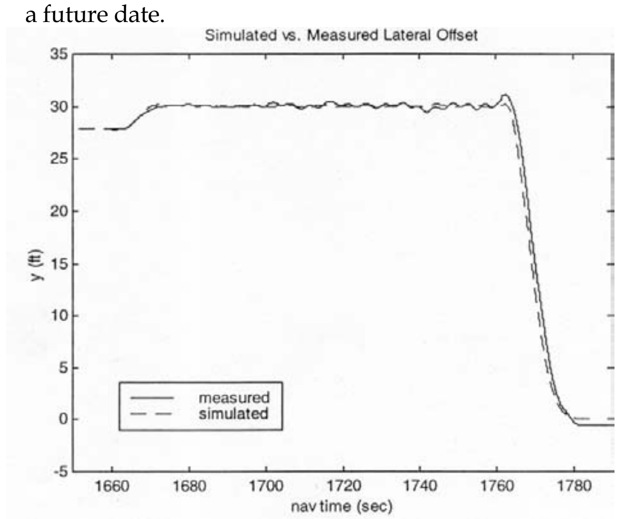
NASA Dryden Flight Research Center Photo Collection
http://www.dfrc.nasa.gov/gallery/photo/index.html
NASA Photo: EC00-0226-21 Date: July 20, 2000 Photo by: Tom Tschida

X-34 and HSTV tow vehicle on lakebed prior to tow test

Status

Following the completion of tow tests up to 80 mph at Dryden, the A-1A vehicle is scheduled to complete additional captive carry flights needed for FAA flight certification.

After acquiring FAA flight certification, the X-34 will be flight tested through a series of approach and landing tests at White Sands Space Harbor in New Mexico. Eventually, the program will incorporate powered flights into its 2-A and 3-A vehicle programs, which are slated for flight test at a future date.



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Unnamned Combat Air Vehicle

Summary: The unmanned combat air vehicle (UCAV) is a Boeing Phantom Works concept for a fully autonomous airplane designed to perform military suppression of enemy air defense (SEAD) missions. Work is currently being done to develop two UCAV demonstration vehicles, which will serve as prototypes for eventual operational vehicles. The demonstration vehicles will be required to taxi in formation, takeoff, fly in formation, execute a collision avoidance maneuver and land in a fully autonomous manner. Operations will be conducted under the supervision of a human operator who will monitor vehicle health, send basic commands and serve as an interface to air traffic control. Dryden will host the flight test program, which is expected to begin in the spring of 2001.

Objective: Demonstrate the technical feasibility for a UCAV system to effectively and affordably prosecute 21st century SEAD/strike missions within the emerging global command and control architecture.

NASA Dryden's major roles in the program are to develop the automatic taxi control laws, and algorithms for formation taxi and collision avoidance. These products will be integrated into the vehicle control system and demonstrated during flight test.

Approach: Dryden control engineers are employing Boeing methods for rapid prototyping of control laws using graphical analysis and design tools in Matrix-X. All control law development from linear analysis to hardware-in-the-loop simulation is performed within the framework of a detailed, high fidelity Matrix-X 6-DOF simulator. Utilizing a single, configuration-controlled simulation ensures that a consistent set of the latest models is used by all UCAV engineers. Ultimately, the flight control software will be obtained by auto-coding the control law block diagrams directly from the simulator.

NASA's ground control system includes steering control, pitch and roll dampers and speed control. The ground control laws share the guidance system and control mixer with the up-and-away flight control laws. This enables a smooth integration into the flight control system and minimizes the potential for problems during transition to and from flight.

Artist conception of UCAV

Dryden's UCAV team is also evaluating concepts for the integration of UAV aircraft such as the UCAV into airspace with piloted aircraft. Concepts include evaluation of a Traffic Alert and Collision Avoidance System (TCAS) to add "see and avoid" capability and the development of two autonomous ground vehicle testbeds for auto-taxi research.

Status: The ground control system has been designed and is undergoing further refinements. The control laws are integrated into Boeing's baseline 6-DOF simulator and are currently being tested in the hardware-in-the-loop simulator in Seattle. Initial formation taxi algorithms have also been developed. Further development work continues in the areas of anti-skid control, failure modeling, brake heat reduction and contingency management.

The collision avoidance and auto-taxi efforts are currently procuring hardware and developing algorithms to be used in their respective independent research programs.

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F/A-18 Autonomous Formation Flight

Summary

A formation flight autopilot was flight tested on the Systems Research Aircraft (SRA) as part of the Autonomous Formation Flight (AFF) project. The NASA 846 support aircraft was used as the formation lead. A total of 32 test points were accomplished multiple times over eleven research flights. The experiment met and, in some cases, exceeded all program objectives.

Objective

The project was to demonstrate through flight test the functionality of autonomous station keeping for a two aircraft formation in support of establishing practical operability of precision formation drag reduction. Performance of the system was measured by monitoring errors in the center of gravity location of the trail aircraft from its commanded position relative to the lead aircraft. The goal of the experiment was to maintain formation control to less than 20 feet.

Approach

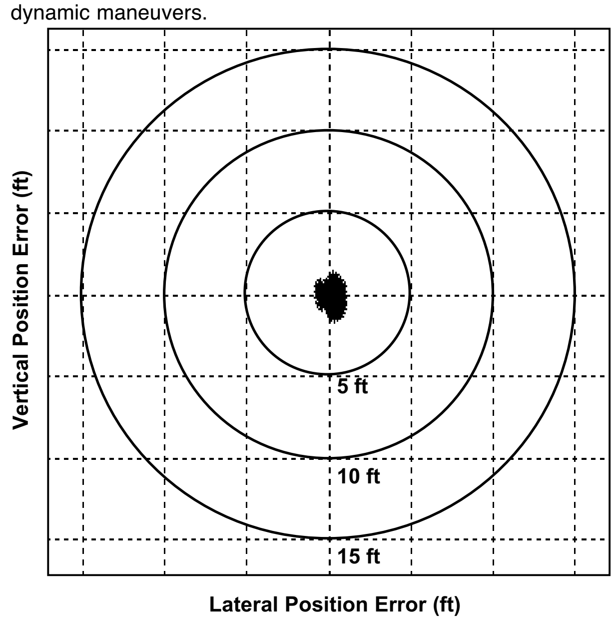
The two F/A-18 aircraft were outfitted with identical GPS receivers and an an air-to-air telemetry system for inter-ship communication. In addition, the SRA was equipped with an Airborne Research Test System (ARTS) and Production Support Flight Control Computers (PSFCCs). The formation autopilot was located in the ARTS, and pitch and roll commands were interfaced to the PSFCCs through analog multiplex/filter cards. A pushbutton display was installed in the back seat for control of the ARTS and the Attitude Reference Indicator (ARI) in the front seat was relocated next to the HUD. Lateral and vertical position errors from the ARTS were displayed on the ARI using its Instrument Landing System (ILS) needles.

In addition to quasi-static relative position measurements and automatic disengage checks, a standard test block of six maneuvers was repeated for each of four different autopilot gain sets. These maneuvers included five-minute steady state tracking tests and 30 foot commanded step inputs in each axis. Dynamic response of the system was observed by having the lead aircraft perform heading sweeps of a few degrees and altitude sweeps of several hundred feet.

Finally, the performance of the autopilot within the wingtip vortex of the lead aircraft was observed by engaging the system outside of any detectable interference effects and commanding small vertical steps toward the vortex location. This was an additional objective outside of the original scope of the experiment.

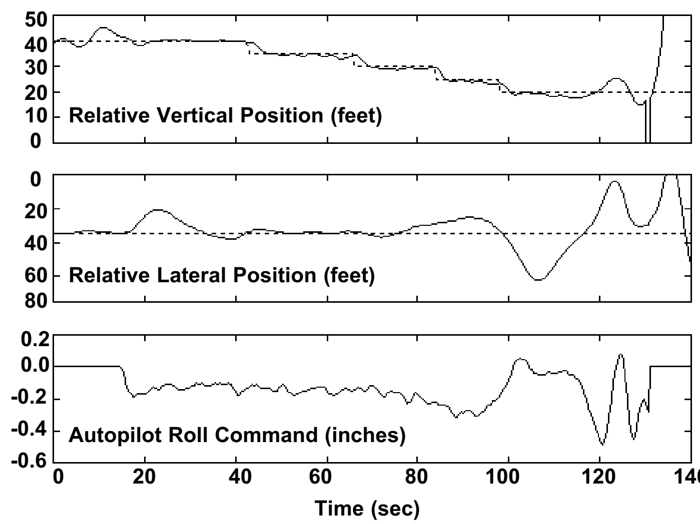
Station Keeping Results

The formation autopilot maintained relative position control to better than ±5 feet for all four gain sets during straight and level flight with turbulence levels ranging from nonexistent to light chop. Accurate, predictable tracking was observed during the step and dynamic maneuvers.



Vortex Control Results

The formation autopilot was able to stabilize within the outer regions of the lead aircraft's vortex. With the wingtips of the two aircraft approximately aligned and the the lead 40 feet above, the trail aircraft was commanded closer vertically in 5 foot increments. Upon reaching a vertical separation of 20 feet, the vortex rolling moment exceeded the bandwidth of the autopilot and caused the trail aircraft to exit the vortex. Due to large fluctuations in throttle position to maintain nose-to-tail separation, no appreciable fuel flow reduction was measured.



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Development of an Integrated, Multi-Stage Simulation for the Hyper-X Program

Summary
The Hyper-X program intends to evaluate the operability and performance of a scramjet powered, autonomous vehicle at hypersonic velocities. To meet program objectives, the X-43A research vehicle (RV) is delivered to the required test conditions in two stages that lead to a final free-flight stage. The first stage consists of a captive carry flight under the wing of a NASA B-52. During this stage, the RV forms the forebody of a modified Pegasus rocket booster, which is slung under the bomber's starboard wing. The second stage is characterized by a B-52 jettison of the Pegasus launch vehicle (LV) and dominated by rocket accelerated flight to the engine test condition. Once near the required flight condition, the RV separates from the spent booster and assumes autonomous controlled flight for the remainder of the mission.

The Hyper-X research program represents a unique challenge to traditional simulation activities conducted at Dryden. Early in the program preliminary estimates of simulation requirements indicated serious labor and technical challenges to creating an integrated 6+6 DOF simulation capable of generating results from the B-52 drop event through the X-43A splash down. Although simulation rates of 100 Hz are sufficient for many air vehicles, challenges arose from the need to simulate the rapid separation event. This event is characterized by relatively fast dynamics that require simulation frame rates in excess of 100Hz. Unfortunately, simulation rates sufficient for the separation event and utilized during the entire mission indicated a requirement to allocate significant computational resources beyond those available to the project.

Objective
The project proceeded to construct three independent, stand-alone simulations capable of simulating all phases of X-43A flight except B-52 captive carry: rocket-boosted ascent, separation of the RV from the LV, and RV autonomous free-flight. This approach, however, is vulnerable to a miscommunication of simulation end-states during phase transitions. Care has been taken to reduce exposure to those vulnerabilities. To further reduce risk, simulation outputs from the LV simulation (LVSim) and the separation simulation (SepSim) have been used as inputs to drive the RV simulation (RVSim)– in effect creating an integrated simulation. See figure below.

Two project objectives helped in defining this integrated simulation approach. The first was a desire to perform a limited verification of simulation outputs between simulations. This is accomplished by executing the RVSim using the inputs supplied by the other two simulations and comparing outputs.

The second objective identified was to evaluate the impact of INS error growth in the sensor package of the RV during the boost event. By modeling these errors during the boost event, simulation activities can be used to assess their impact on stability and performance – two important characteristics of an autonomous vehicle.

Approach
In the integrated simulation mode, the RVSim is limited to emulating only the effects of the other two simulations. For example, the RVSim does not carry a model of the rocket motor but instead relies on the simulated effect of that model carried in LVSim. There are two model effects required by RVSim to operate as an integrated simulation. The first is a profile of the boost event that is used to drive the equations of motion in the RVSim – but only during boost. The second model governs the force and moment perturbations to the research vehicle during the separation event and is referred to as the SepΔ model.

Results
The approach detailed in this report has proven useful in conducting sim-to-sim comparisons and evaluating INS error growth during boost. Though this may not be the preferred approach, it has proven useful without excessive computational resources.

Status/Plans
Continued work is planned to further evaluate and refine the approach.

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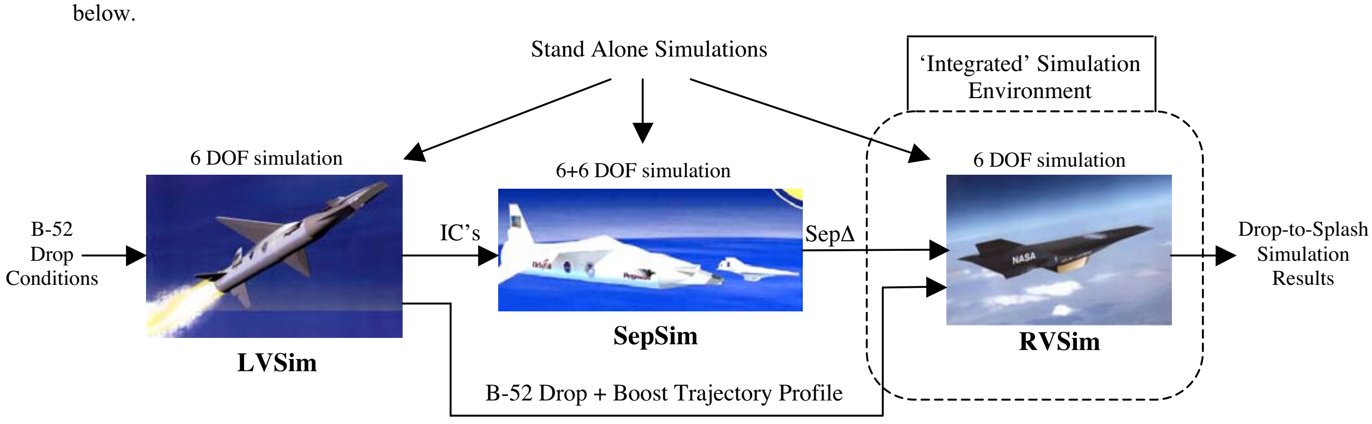

BWB LSV Actuator Performance Characterization

Summary

The Blended Wing Body (BWB) Low-Speed vehicle (LSV) is a sub-scale, remotely-piloted research aircraft intended for flight research critical to commercial BWB development. The aerodynamic actuation system for the LSV is currently composed of 16 aerodynamic surface actuators, and as many as eight actuator controller units. The actuator and controller are COTS components from an existing design, but are repackaged for use on the LSV. The LSV deflection accuracy, hinge moment envelope, and mission duration requirements are significantly different from the original actuator requirements. Therefore, it is necessary to obtain actuator performance data within the LSV envelope. This data will be used to verify actuator requirements, and to develop a simplified actuator model for real-time simulation.

Objective

Characterize the performance of flight control actuators for implementation in simulation as well as flight control design and analysis.

Justification

The COTS actuators chosen for the BWB LSV did not have a tested math model of the actuator performance. In addition, the LSV requirements are significantly different than the original requirements of the actuator. Math model development from test measurements throughout the envisioned LSV envelope provides an accurate representation of actuator performance.

Approach

A test stand was constructed for mounting three actuators and deflection measurements (Figure 1). Linear actuator motions deflect calibrated torsion rods mounted to the test stand. The torsion rods apply resistive loads to the actuator up to the actuator stall force. A test engineer uses a UNIX workstation to command the actuators during the test sequences. Frequency sweeps, step inputs, ramps, and multi-spectral inputs were used to characterize the actuator response. Torsion rod deflections were measured by an optical encoder as well as position transducers.

Results

Recorded data was analyzed in the frequency and time domains using Matlab. A discrete, non-linear actuator model was produced in Simulink to represent the actuator performance.

The reduced-order math model reproduces the actuator test data fairly well (Figure 2). The model consists of 3rd order dynamics and three non-linear force and position dependent elements: acceleration limit, rate limit, and steady state error. The model is computationally simple enough to be used 16 times (for 16 surfaces) in simulations.

Status

The latest control laws for the BWB LSV have been designed and analyzed using the math model developed from actuator testing. The capacity for additional testing and model development is retained as maturation of flight systems design may require.

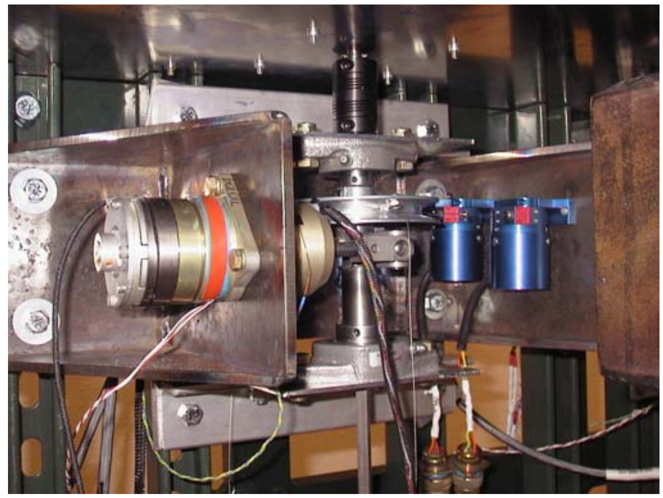

Figure 1 - Actuator mounted to bracket on left. Bellcrank, torsion rod connection, and pulley for position transducers in the center. Coupling for optical encoder in top center (encoder mounted above). Position transducers mounted on bracket on right.

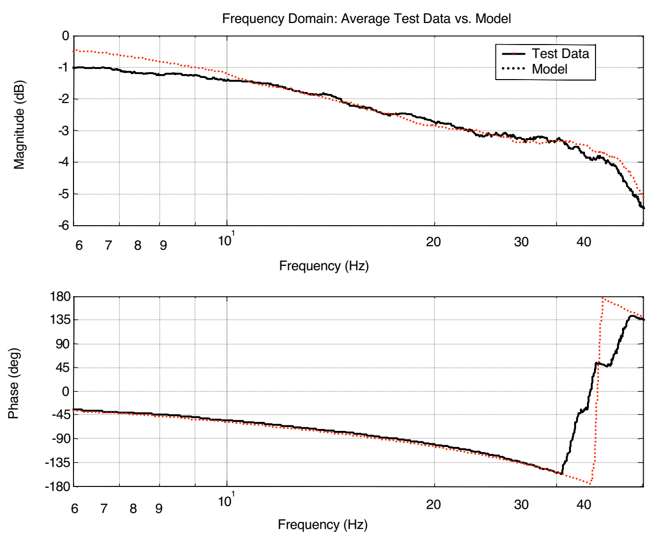

Figure 2 - Comparison of averaged test data and model computations in the frequency domain (from linear sweeps)

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Stability Margin Measurement for Trajectory Analysis

Summary
A new generation of research vehicles such as the X-33, X-34, X-37, and X-38 has been developed to provide more cost-effective access to space. All these vehicles are flown autonomously along pre-defined trajectories. These trajectories include both a boost phase and a re-entry phase where the vehicle experiences fairly rapid changes in flight condition. A new method for measuring stability margins along these trajectories has been developed and applied to the X-34.

Objective
Develop and validate a method of measuring stability margins while flying along a pre-defined trajectory. The measurement should be made over a short time span so that changes in flight condition do not effect resulting measurement.

Justification
Monte Carlo analysis has traditionally been used to evaluate robustness of a vehicle over a given trajectory. Multiple simulations runs are performed with parameters varying over an uncertainty range. Vehicle survival has been used as the measure of success. While this highlights catastrophic events it does not give a measurement of how close some of the simulations were to failure. Adding a stability margin measurement as data collected during Monte Carlo runs would highlight cases that came close to failure.

Benefits
- Method applicable to nonlinear simulation
- Short time duration required
- Simultaneous inputs to pitch, roll, and yaw axis
- Time history of margins over entire trajectory
- Additional measure of merit for Monte Carlo runs

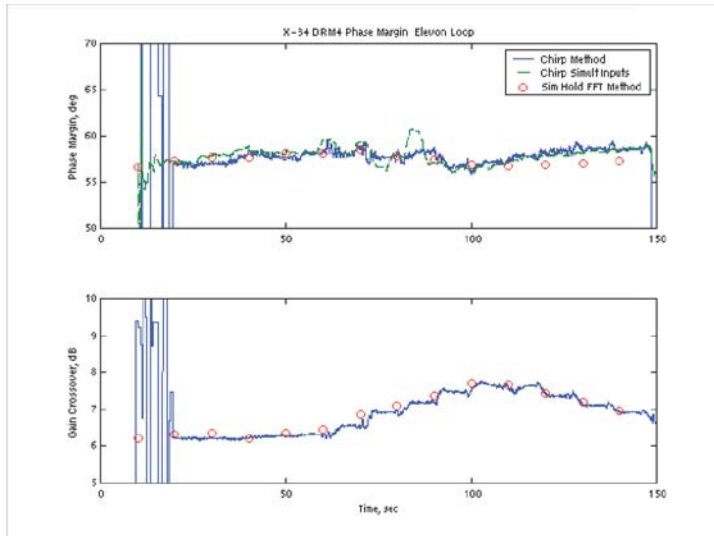
Approach
A low level tailored input is applied to each active flight control surface. Chirp-Z transformations are used to calculate the open loop transfer function for each axis. Stability margins are calculated over a sliding window of time data. Margins are output along with other simulation data.

Results
Stability margins along the X-34 approach to landing trajectory were calculated using 1) linear models, 2) conventional FFT methods with a lower degree of freedom simulation, and 3) the new method. Results were in close agreement. Measuring all three axis (pitch, roll, and yaw) at the same time produced results equivalent to measuring one loop at a time.

Advances
- Method for de-trending data developed to improve results
- Efficiency improved by reducing the number of Chirp-Z transforms required (input transform is only calculated once)

Status
The new method has been validated against proven conventional stability analysis methods. Next step is to incorporate stability margin as data collected during Monte Carlo simulations.

Method will be used in flight to measure stability margins during X-34 approach and landing tests.



Stability margin measurements obtained for the X-34 approach trajectory.

X-34 Vehicle

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X-43A Flight Systems Validation

Summary

In year 2000, X-43A flight systems efforts focused on the completion of the X-43A validation testing in preparation for first flight in 2001. This testing included hardware-in-the-loop software validation on the flight computer, aircraft-in-the-loop validation using the vehicle control surfaces and the flight computer, and end-to-end system blow-downs with pressurized fuel systems. The year concluded with the start of integration activities between the X-43 research vehicle, the vehicle to Pegasus booster adapter, and the booster. In conjunction with the test activities, the flight systems team presented an overview of the X-43 validation activities at the JANNAF conference in November.

Objectives

The objective of the flight systems validation testing is to subject the X-43A vehicle to simulated flight conditions to evaluate system response. In order to accomplish this objective, multiple tests were run with the flight software, flight hardware, and pressurized and unpressurized fuel systems.

Approach

Prior to execution of the validation tests, effort focused on development and verification of the ground test equipment. This primarily included the simulation equipment, the ground power interface, and the vehicle fuel cart. The systems group supported verification of the vehicle simulation, including the flight computer inertial simulator and the analog/digital interfaces. Multiple trajectory tests in the hardware-in-the-loop configuration were performed in order to fine-tune the trajectory. The systems group designed the vehicle power interface to provide the command and control functions available on the B-52 carrier aircraft. This resulted in a design for the monitoring station aboard the B-52 based on lessons learned during the ground tests. A flight systems engineer was selected to crew the monitoring station on the B-52 for flight operations. Additionally, the electrical design and fabrication, including power and data interfaces, for the fuel servicing cart was completed prior to the first pressurized blow-down.

Validation testing began with the execution of Failure Modes and Effects testing. Discrete signals present on the ground equipment were used to simulate flight interfaces present on the monitor station in the B-52. Additional discrete signals were provided to simulate the separation protocol with the launch vehicle, and actuate solenoid valves as required. Following completion of the hardware-in-the-loop testing, several tests were run with the flight actuators. These aircraft-in-the-loop tests validated command and control of the four surface actuators during nominal and off-nominal vehicle trajectories.

The systems team supported several fuel systems blow-downs in order to validate fuel system performance. These included three blow-downs with inert gases instead of the hydrogen and silane used for flight. An additional blow-down was conducted with hydrogen and a hydrogen/nitrogen mixture instead of silane. Each of the blow-downs used flight hardware with a simulated vehicle trajectory. A plugs-out test using inert gases was also conducted to validate systems performance without the simulation connected. The year concluded with the execution of an integrated vehicle to adapter blow-down to validate hand-over of the adapter cooling and purge to the vehicle at separation.

Results

The X-43A flight systems validation test program continues to be highly successful. To date, both mission critical anomalies and non-critical performance related anomalies have been uncovered. Discovering and correcting these anomalies demonstrates the effectiveness of the validation program.

X-43A Mated to Pegasus Booster

X-43A Fuel System Test

Contacts

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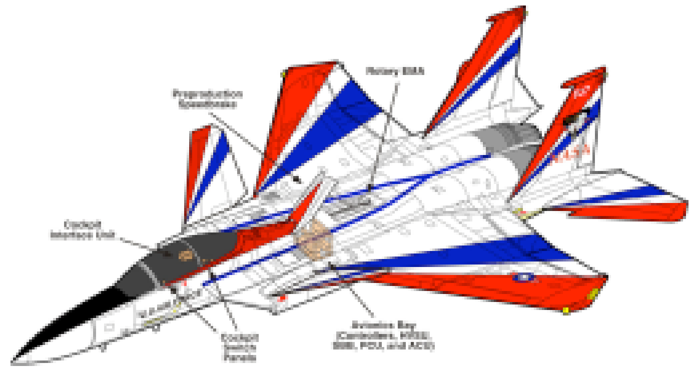
X-Actuator Control Test (X-ACT) Program

Summary

For X-ACT, the X-38 rotary rudder EMA will be installed in place of the standard hydraulic speedbrake actuator on NASA #837, a heavily modified, preproduction F-15B two seat fighter aircraft. A Dryden-designed jackscrew mechanism will be used to couple the output shaft of the EMA to the brake surface. An Actuator Control Unit (ACU) will be used to provide actuator commands normally generated by the X-38 Flight Critical Computers. Actuator commands will be specified by a Flight Test Engineer in the back seat of the aircraft via a cockpit interface system. The X-38 Failure Detection, Isolation, and Recovery (FDIR) logic will be hosted in the ACU, with full authority to reconfigure the system. Failures can be injected via the crew interface, either by modifying data passing to and from the FDIR code, or by interrupting signals between the actuator and controller.



X-ACT System Locations

Objectives

The primary objective of the X-ACT program is the following:

Test the X-38 V-201 Software (Failure Detection, Isolation, and Recovery Logic, Load Equalization Logic, and Built-In-Test Logic) in a Flight Environment.

The following are secondary objectives:

Gain operational experience with an X-38 V-201 Rudder Actuator/Controller System in a flight environment.

Develop a Multipurpose Cockpit Display System for Flight Research

Justification

The research value of the X-ACT flight test program is in the establishment of a new redundancy configuration, potentially suitable for a wide range of new vehicle applications. These flight tests will either verify the correctness of the design approach or identify deficiencies in the configuration to allow correction. Lessons learned during this program should be applicable to other high power EMA and EHA programs currently in development, both for spacecraft and aircraft applications. A successful flight test of the actuator could help further the acceptance of Power-By-Wire actuation technology by industry.

Approach

The flight test matrix will span the load and rate envelope of the actuator. The actuator will be tested throughout the full range of its stroke. Some typical X-38 reentry profiles may be flown, from an altitude of 50,000 ft. down to the moment of X-38 parachute deployment. The goal is to have 25 flight hours on the actuator by the time vehicle V-201 ships to KSC for Shuttle Integration. An additional 75 flight hours will be accrued prior to first flight of the CRV vehicle.

Results

Currently under development

Benefits

Reduced Risk for X-38.

Validation of new Redundancy Management Schemes for Power-By-Wire Actuators.

Lessons Learned applicable to multiple EMA development programs.

Contact

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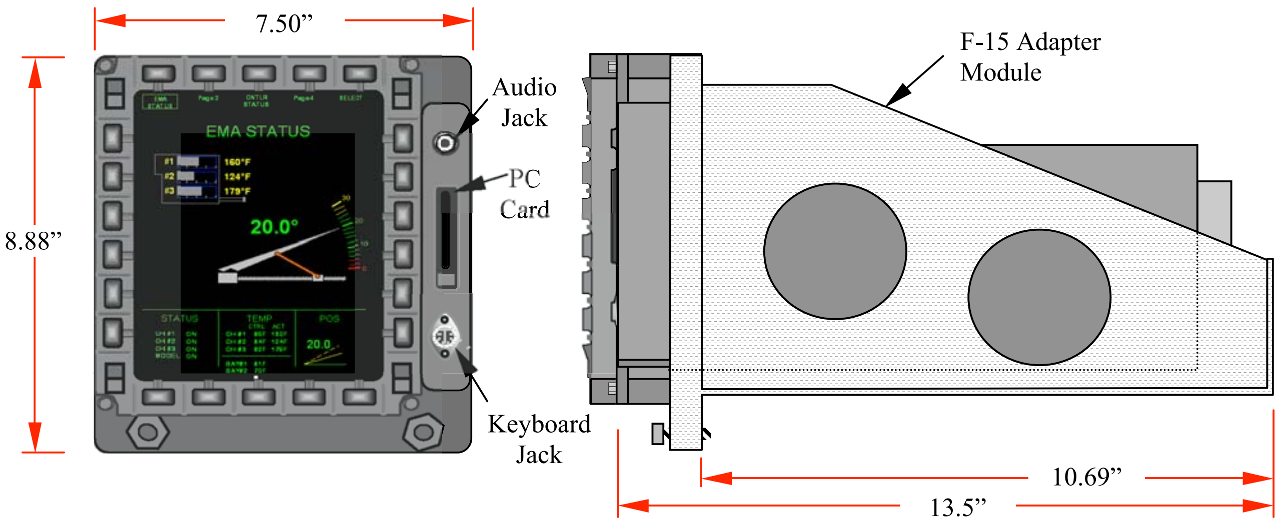
Cockpit Interface Unit (CIU)



Summary

The Cockpit Interface Unit (CIU) is a PC104 based Multi-Function Display built by Applied Display Technology to NASA/DFRC specifications. The CIU shall provide a high resolution, sunlight readable display and surrounding bezel-button switch panel as an interface to the NASA #837 F-15 back seat crewmember. The CIU capabilities also include, video display from remote cameras in the aircraft, rapid reprogramming of the display software by NASA personal, experiment monitoring and control of non-essential aircraft components. The CIU is also equipped with audio interface, PC Card (PCMCIA) interface and PC/AT compatible keyboard interface.

The CIU graphical displays are prototyped on a separate development workstation then compiled into a stand-alone executable using the VAPS software package by Virtual Prototypes Inc. Once the executable has been generated it can be uploaded to the CIU via the PC Card interface. The Embedded Windows NT operating system in the CIU has a significantly smaller footprint than standard Windows NT 4.0.

Objectives

The primary objectives of the multipurpose Cockpit Interface Unit, can be grouped into the following two categories:

X-ACT Cockpit Interface System

Development of a Cockpit Display System for the X-ACT flight test program that will allow for onboard monitoring of experiment status, including mission critical parameters. In addition to its experiment monitoring capability the X-ACT CIU will allow the crew to execute test scripts that command actuator position and trigger the hardware and software fault injection system.

Technology Demonstrator for a Generic Flight Test Display Unit

The CIU was designed with the flexibility to be used in other experiments and on different aircraft. A low cost multipurpose display unit with hardware and software that is interchangeable among several types of aircraft and ground test configurations would be valuable to any flight test facility. The USAF and at least two aerospace companies have expressed unsolicited interest in the CIU for their projects.

Benefits

Rugged low cost multipurpose display unit
Rapid in house prototyping of display software
Interchangeable among several aircraft
Useful experiment and monitoring interface

Specifications

- 800 x 600 AMLCD, 5.0 x 6.7 inch viewable NVG, sunlight readable display, 256 color
- 23 programmable switches
- Pentium 233MHz/MMX CPU
- Embedded Windows NT Operating System
- MIL-STD-1553B duel redundant RT/BC/M
- 20 Relay/20 Optoisolated Digital I/O module
- 112 MB Solid State Disk
- PC104 Quad RS-422/232 Module
- PCM Bit Sync & Decom Module (option not currently exercised)
- Possible expansion to include a 2nd remote LCD

Contact

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X-43A RF SYSTEM REDESIGN AND EFFECTS

Summary

An antenna pattern test was conducted on the X-43A, ship 2, at Micro Craft, Tullahoma, Tennessee, on September 8, 2000. A previous antenna pattern test was conducted on the X-43A at the Benefield Anechoic Facility (BAF) (report HX-DFRC-0114, April, 2000) at Edwards AFB. The antenna patterns of that test led to a redesign of the X-43 RF system.

The RF redesign included phased matched cables for the telemetry and transponder systems that handle higher power. An S-band transmitter was added to split the aft antenna from the two side antennas. Two way splitters/combiners replaced three way splitters.

Due to the extensive changes in the RF system, another antenna pattern measurement test was required. The patterns collected from the second test demonstrate that the changes made were acceptable and that the antenna coverage is adequate.

Objective

The test objective was to demonstrate that there is an improvement in the RF coverage for the X-43A by comparing antenna patterns before and after the RF system redesign.

Justification

In general, antenna pattern testing is necessary to show the RF coverage around the vehicle. Results from the previous antenna pattern test for the X-43A showed areas of high attenuation of the RF signal indicating potential loss of signal for the actual mission. The RF system was subsequently redesigned to improve the RF coverage, so a retest of the RF system was necessary.

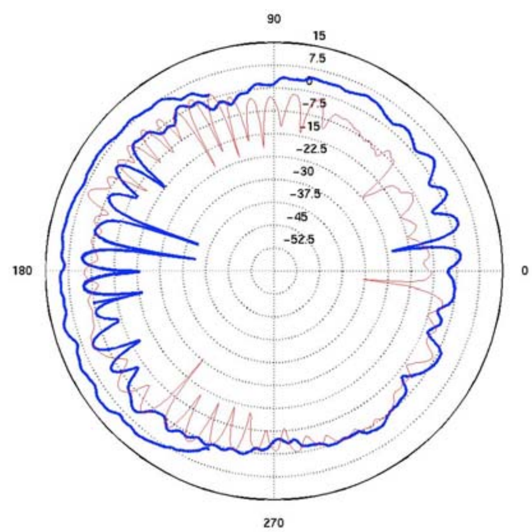
Approach

The Naval Air Warfare Center Weapons Division (NAWCWD) China Lake, California used their Miniature High Speed Radar (MHSR) to measure antenna patterns for X-43A ship 2. The signal from the radar was input into a low loss coaxial cable that was connected to an external cable on the aft of the aircraft. The aircraft was rotated by a positioner while the aircraft antennas radiated the signal back to the receive antenna of the radar. A data acquisition system and positioner controller was located in racks with the radar.

The aircraft was lifted to a height of ten feet above the ground. The receive antenna was positioned on an antenna tower with an adjustable height so that the correct elevation angle between the receive anttenna and the aircraft could be achieved. The aircraft was positioned approximately 30 feet in front of the radar. The signal gain was plotted at each azimuth angle as the X-43 was rotated. Several frequencies were tested.

Results

The antenna patterns plotted for this test were compared with the previous test. The plot below is an example of the comparison. The test results are documented in document HX-DFRC-0146, January 26, 2001. The regions of attenuation are improved on the port and starboard of the aircraft. RF coverage provided by the aft antenna is improved as well. All frequencies tested show similar results. With the redesigned RF system, no data loss is expected during the mission.



Antenna Pattern Comparison After RF Redesign

Contacts

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Airborne Parameter Display System

Summary

Originally developed as a backup to the Integrated Telemetry Analysis System (ITAS) on the B-52, the Research Instrumentation Decom and Display System (RIADDS) is now the primary system for displaying X-43/Hyper-X safety of flight parameters onboard the B-52.

Objectives

RIADDS is required to decommutate the HXRV PCM signal and graphically display selected parameters at a flight monitoring station located on the B-52 lower deck.

Justification

The ITAS system currently used is outdated and becoming increasingly difficult to support. RIADDS eliminates this problem using in-house developed software and the availability of replacement parts.

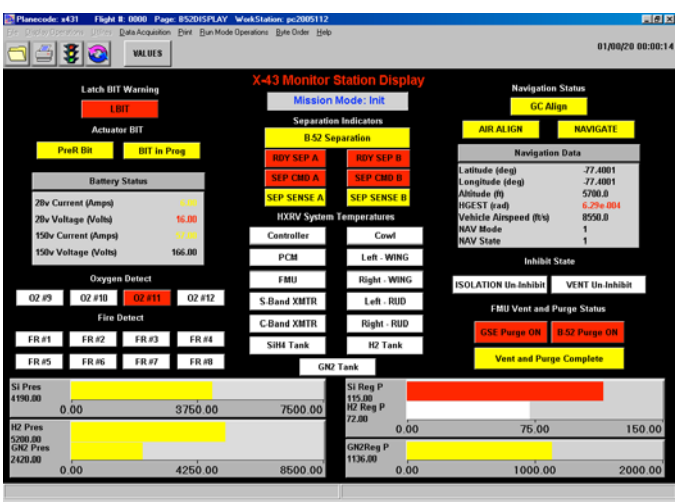
Approach

RIADDS is a VME based system housed in a ruggedized Long ATR chassis built by Macrolink (ML1000 series). For vibration isolation, RIADDS was mounted on 4 Barrymount T44-AB-10 shock mounts. In order to prevent any failures caused by vibration, all Commercial Off The Shelf (COTS) parts were ruggedized.

Software developed with VxWorks is able to decommutate, concatenate, and apply EU conversions to a subset of the CIMS file.

A utility was developed to generate both the subset of the CIMS file and a simulation file for non-real-time debugging of the display.

The standard Dryden Control Room Parameter Display System (PDS) was modified to support RIADDS' VME architecture. This allows RIADDS to generate a control room type display at the B-52 monitoring station.



Airborne Parameter Display Page using PDS
(Simulation Mode)

RIADDS Ruggedized Long ATR Chassis

System Hardware

Manufacturer	Part Number	Description
Datum	PC03V	Time Code Processor
Berg	V64	Decom
VMIC	VMI/VME 7588	Pentium
Synergy	VGM5	PowerPC
Seagate	ST32155W	Hard Drive
Netgear	DS104	Ethernet Hub

System Specifications

Weight	36.45 pounds
Size	7 _" H x 10 _" W x 21 _" L
Power Consumption	28V @ 6A
Temperature	7ºF to 120ºF (operating @ lab alt) 0ºF to 90ºF (operating @ 25,000ft) 0ºF to 130ºF (storage)
Altitude	25,000 feet
Vibration	Contact Russ Franz
Operating System	Windows NT4.0 SP6.a
Reset (Backplane)	Front panel and/or remote momentary switch
Input/Output	Monitor, SVGA (HD15) Mouse (PS/2) Keyboard (PS/2) 10/100 LAN (RJ45) PCM (BNC 75Ohm) IRIG-B (BNC 75Ohm)

Results

Installation in the B-52 is complete and initial integration tests have been conducted. All anomalies have been fixed and integration issues resolved. Continued checkout is underway.

Status

A duplicate system is currently being assembled.

Acknowledgments

Richard Hang (VxWorks development)
Stephan Hoang (PDS modifications)

Contact

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Efficient Modulation Techniques

Summary:
NASA Dryden and UCLA are investigating techniques that enhance the efficiency and reliability of the transmission of flight-test data. Current areas of research include Multiple Antenna Systems, Spatial-Temporal Processing, Orthogonal Frequency Division Multiplexing (OFDM), Adaptive Equalization and Feher Quadrature Phase Shift Keying (FQPSK)* Systems.

UCLA originally proposed the investigation of Space-Time Coding to increase the efficiency of modulation techniques currently being used or developed by NASA, DoD and other organizations for the transmission of flight-test data. The technique utilizes multiple antennas and spatial-temporal processing to improve bandwidth performance and reduce susceptibility to frequency selective fading. The use of OFDM further enhances system performance.

FQPSK modulation has been adopted as a new standard for high data transmission rates by the Range Commanders Council Telemetry Group. FQPSK is a form of Offset Quadrature Phase Shift Keying (OQPSK). The use of an equalizer can enhance the performance of QPSK systems in a frequency selective fading channel.

Objective:
Expanding data requirements force the transmission of increasing amounts of flight-test data. This research allows the transmission of flight-test data at greater rates with improved accuracy and efficiency.

Results:
Multi-Antenna Spatial-Temporal Processing when used in conjunction with OFDM techniques appears to be a promising method to achieve higher data transmission rates as well as counteract the adverse effects of frequency selective fading in a fixed bandwidth channel. UCLA has completed simulations to evaluate the performance of these systems, figure 1.

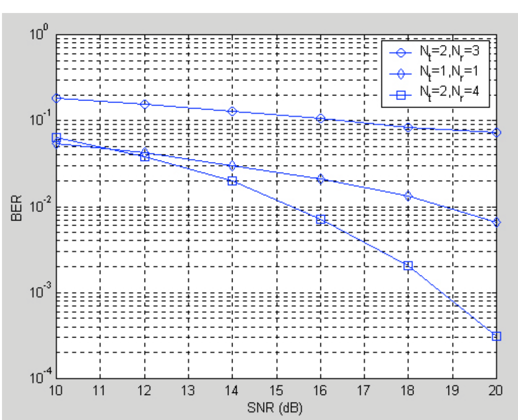


Figure 1: BER vs. SNR of OFDM with differing numbers of Transmit (Nt) and Receive (Nr) antennas. Data rate is proportional to Nt.

UCLA is investigating adaptive equalization methods to improve the current performance of OQPSK systems. The equalizer provides amplitude and phase compensation to restore the received signal to it's original transmitted form and reduce the effects of fading on the quality of received data, figure 2.

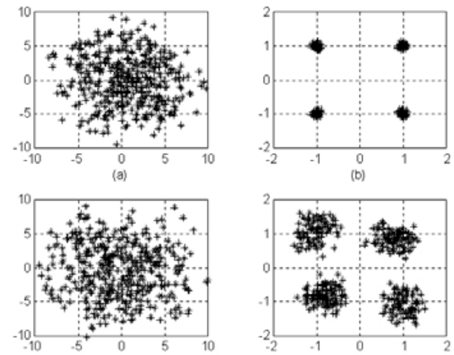


Figure 2: Signal constellations before and after equalization for QPSK (a & b) and OQPSK (c & d) modulation schemes.

Status/Plans:
Methods for the design and implementation of Multi-Antenna Spatio-Temporal Processing OFDM systems are being investigated with UCLA and possible small business partners. Efforts include the investigation of existing hardware utilized in cell phone and internet applications that meet or can be modified to meet NASA Dryden's requirements.

UCLA is continuing the modeling and simulation of adaptive equalizers for OQPSK systems and their design and optimization for Dryden's transmission channel.

Evaluation and flight-testing of FQPSK systems is in process. A PCM/FM system will be flown at the same time for comparison. Bit error rates and multipath performance will be evaluated and compared. Compatibility with existing ground station equipment is also be evaluated. Dryden is also looking at possibilities for evaluating the performance of FSPQK transmission systems in a high doppler rate environment during flight tests aboard a hypersonic vehicle.

Contacts:
Don Whiteman, NASA Dryden, RI, x3385
Kung Yao, Elec. Engr. Dept., UCLA

References:
(1) Tung, Yao and Whiteman, "Multiple-Antenna Spatio-Temporal Processing For OFDM Communications Over Frequency-Selective Fading Channels", 2000 International Telemetry Conference Proceedings

(2) Fan, Yao and Whiteman, "Adaptive Equalization For OQPSK Through A Frequency Selective Fading Channel", 2000 International Telemetry Conference Proceedings

* FQPSK is Feher patented QPSK.

Miniature 3-Axis-Vibration High-Frequency Data Logger

Summary

A miniature, stand-alone system for acquiring high frequency vibration data was developed and flown.

Objectives

Develop a miniature and easily used accelerometer data logger. System can be quickly installed in aircraft areas to acquire high fidelity information used to characterize vibrational environments.

Justification

A need was recognized to quantify ER-2 payload bay vibration environments. Existing published data was considered outdated due to several aircraft modifications and engine upgrades. Data is critical to aid developers to flight qualify scientific instruments for ER-2 payload bays.

Approach

Existing data logger hardware developed by Code RI Flight Instrumentation was applied as part of the solution. A commercial tri-axial accelerometer package was procured and the data logger hardware was modified to accept accelerometer data. Data was digitized at 10,000 samples per second per channel and stored onboard on hard disk. Data was retrieved postflight and processed using MATLAB-based programs to generate frequency spectral response.

Status

The system flew on the NASA B-52 to characterize cabin vibration in support of X-43 flight monitoring equipment (see Reference). System is ready to fly on ER-2 predicated on flight opportunities. Inquires from F-18 Autonomous Formation Flight project have been received to support flight qualification of experimental equipment.

Photograph of data logger (tall box), optional control box, and three-axis accelerometer

Contact

Phil.Hamory@dfrc.nasa.gov, (661)276-3090; Ryan Heine; Russ.Franz@dfrc.nasa.gov, (661)276-2022.

Reference

Russ Franz, “Measurement and Analysis of B-52 Cabin Vibration Levels”, 2000 Research Engineering Annual Report.

Specifications of Present Configuration

Full Scale Range	+/- 25g
Resolution	0.012g
Noise Level	0.020g
Accuracy	+/- 10% (*)
Sample Rate	10,000 Hz
Frequency Response	2 Hz to 2000 Hz
3-pole Butterworth anti-aliasing filter cutoff frequency	2900 Hz
1-pole AC coupling filter cutoff frequency	1.6 Hz
Accelerometer	Endevco 5253A-100

Recording Time	10 records lasting 34 seconds each
Recording Medium	Laptop-type hard drive
Triggering	Manual
Size	6.425” (L) x 4.350” (W) x 4.700” (H) as shown in photo
Weight	3 pounds
Power Consumption	250 mA @ 28V
Temperature	0 F to 170 F
Altitude	up to 50,000 ft
Vibration	System has passed DFRC Process Spec 21-2 Curve B

(*) The main limitations are cross-axis sensitivity and frequency dependence of the accelerometer chosen.

Measurement and Analysis of B-52 Cabin Vibration Levels

Summary

In support of environmental qualification for the Research Instrumentation Airborne Decom and Display System, RIADDS (see Reference 1), a cabin vibration level study was conducted. For hardware description, see Reference 2.

Objectives

Collect high frequency accelerometer data during typical B-52 flight conditions.

Complete spectral analysis on this data to determine the vibration levels from 2Hz to 2000Hz.

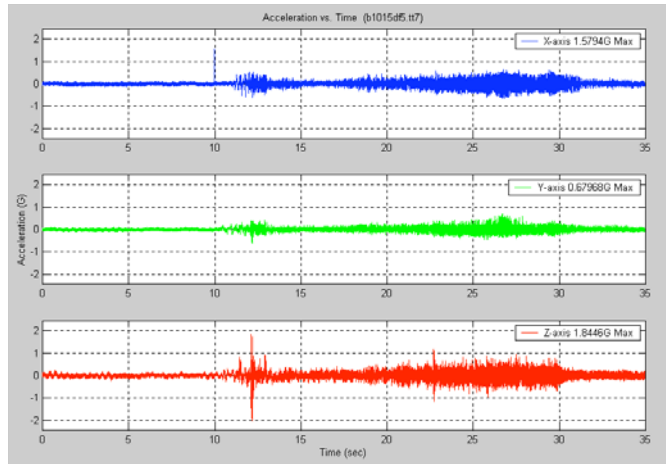
Compare the actual cabin environment to DFRC Process Spec 21-2 curves to determine test applicability.

Justification

Typical B-52 cabin vibration levels were needed to determine if testing to power levels less than Curve A would be reasonable for the environmental qualification of RIADDS. Some of the critical electronics in RIADDS are Commercial Off The Shelf (COTS) and unnecessarily severe testing during environmental qualification was to be avoided, if possible.

Approach

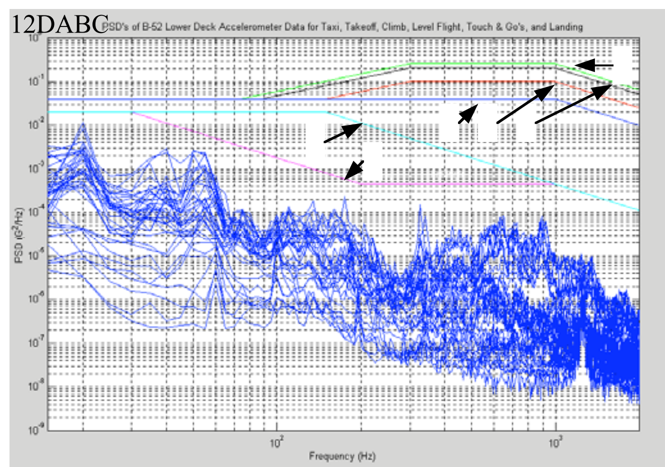
The accelerometer system, installed on the top shelf of the equipment rack located in the lower cabin of the B-52, collected data on a proficiency flight (Flight 1015.) Ten files were logged capturing engine start, departing taxi, takeoff, climb, cruise, 3 touch and go's, landing, and returning taxi. The data was offloaded and imported into MATLAB for processing. Processing included time-series analysis of accelerations and Power Spectral Density (PSD). A MATLAB PSD analysis routine was used with a 5 Hz frequency resolution to process the spectra.



Time Series of the Touch & Go
(Lateral, Longitudinal, & Vertical axis, respectively)

Results

The maximum power density (0.012 G^2/Hz) occurred at 20Hz during a touch and go and was derived over a 2-second time window of the vertical axis. In general, power densities dropped off at a rate steeper than –6dB/Octave to 2000Hz. At 20Hz the maximum power density is 1/4th of Curve A and at 2000Hz the power density is 4 orders of magnitude less than Curve A. Due to the significantly lower vibration levels in the B-52 cabin, Curves 1 and 2 were used to qualify RIADDS, Curve 1 maintaining at least 6dB above B-52 levels observed.



PSD's of various flight conditions compared to
DFRC Process Spec 21-2 curves. (Flight 1015)

Status

Currently, the time-series data is archived on CD and available upon request. The accelerometer measurement system and analysis technique are general purpose and available for other applications to quantify environmental vibration levels and spectra.

Analysis Summary

Sample Rate	10,000 Hz
File length	~34 seconds
PSD frequency range	15–2000 Hz
PSD frequency resolution	5 Hz
Maximum Acceleration	1.84G's (Vertical axis)
Maximum Power Density	0.012G^2/Hz @ 20 Hz

References

1) *Airborne Parameter Display System*, 2000 Research Engineering Annual Report, Russ Franz, RI.

2) *Miniature 3-Axis-Vibration High-Frequency Data Logger*, 2000 Research Engineering Annual Report, Phil Hamory, RI.

Contacts

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Adjustable-Height Skin Friction and Flow Direction Sensor

Summary

Laboratory testing of an adjustable-height skin friction and flow direction sensor was conducted in a low-speed wind tunnel. Results are encouraging.

Objectives

The objective of this work is to advance the state-of-the-art in pressure-based indirect measurement of shear stress. The goal is a device that measures both the magnitude and direction of shear stress.

Justification

Skin friction (or shear stress) is an essential parameter in flight research for performance evaluation and safety assessment. It is a difficult parameter to measure accurately, and there is no single preferred measurement technique. Indirect measurement techniques are popular because they are easy to use. For example, Preston tubes and Stanton gages measure the pressure difference caused by a surface obstacle, and this pressure difference is interpreted via calibration as shear stress. Indirect techniques have important limitations however. For example, Preston tubes and Stanton gages are limited by factors such as pressure gradient, flow direction, and physical size.

Approach

The concept for the sensor is that of an adjustable, omnidirectional probe (or surface obstacle) operating at minimum protrusion levels (heights). The sensor was designed by Professor Raimo Hakkinen of Washington University in St. Louis, MO, and is shown in Figure 1. The obstacle's diameter is 10 mm (0.393"). It protrudes into the flow from the tunnel wall over a range of 0 to 2.4 mm (0.094"). Around its circumference are 12 pressure ports spaced 30 degrees apart. Beneath the sensor is a precision actuator for adjusting the height of the probe. The present construction is for developing the concept and obtaining calibrations. Thereafter, a version for routine use in research flights could presumably be more compact.

Results

The sensor was tested at one laminar and two turbulent conditions in Washington University's low-speed wind tunnel. Results from the lower speed turbulent case (flow velocity of 62 ft/s, Reynolds number of 1.4 million) are shown in Figure 2 which plots pressure against port number. Port number is defined with respect to flow direction; port 1 is in the back and port 7 is in the front of the sensor. Each curve represents readings for a given protrusion height.

With peaks at port 7, the results make it clear that the sensor's ability to detect flow direction is highly pronounced. This represents an advancement in the state-of-the-art for pressure-based skin friction gages. Furthermore, the directional capability is evident even at small protrusion heights. That is a significant result because that means the obstacle only needs to protrude far enough into the flow to get a measurable pressure difference, and minimal protrusion is desirable for minimal disturbance to the flow. Magnitude readings are consistent with values published for Stanton and Preston tubes.

Figure 1. Photograph of skin friction sensor and reference instrumentation in Washington University Low-Speed Wind Tunnel

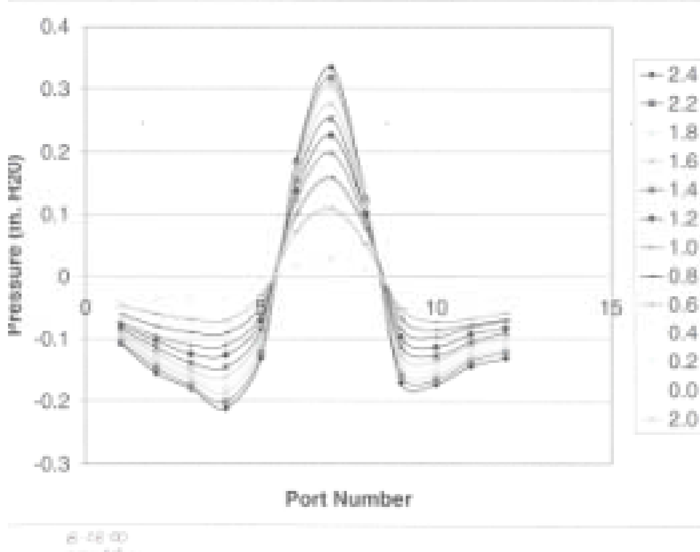

Figure 2. Raw data from the sensor's 12 pressure ports at 12 protrusion heights.

Status

Testing at higher Reynolds numbers and in yawed velocity profiles has been proposed next and is waiting for funding.

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Drag Reduction Validation Methodology for Autonomous Formation Flight

Summary The Autonomous Formation Flight (AFF) Project goal is to achieve 10% fuel savings in commercial flight operations by flying two or more aircraft in formation. This goal will be achieved by advancing
the concept of formation-flight drag reduction from the experimental proof-of-concept stage to a prototype demonstration within three years. A critical requirement to achieve the goal is an accurate determination of both vehicle and engine performance during flight. Accurate in-flight thrust measurement is important for the evaluation of both aircraft and engine performance.

Performance Objectives

The objectives are:

•Quantify the reduction in aircraft drag and engine fuel consumption during formation flight
•Assess the effect of dynamic throttle usage on aircraft performance during formation flight by investigating how various throttle rates affect fuel consumption and evaluating performance accuracy during unsteady throttle conditions
•Assess the accuracy of predicted performance benefits
•Compare accelerometer, INS, and energy methods for determination of excess thrust
• Improve understanding of basic physics: “drag reduction” vs upwash effect

Research Scope and Approach

The general research scope and approach are as follows:

•Instrument flight vehicle with a dedicated thrust/performance measurement system
•Develop real-time models and displays to assure data and maneuver quality are acceptable during tests
•Develop post-flight analysis methods and models to determine “standardized” thrust and performance values
•Conduct engine ground test using the AFFTC Aircraft Horizontal Thrust Stand (pad #18)
•Conduct flight test to obtain engine thrust and vehicle performance data and develop optimum flight maneuver technique for AFF evaluation
•Monitor flight tests for data and maneuver quality

Research Scope and Approach (cont.)

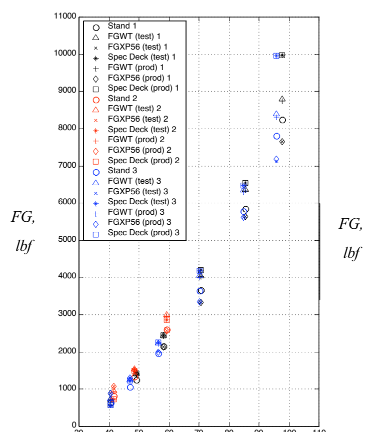
•Analyze in-flight thrust and aircraft performance data and calculate post-flight, standardized thrust and aircraft performance values
•Investigate how dynamic throttle use affects accuracy of thrust model

Status and Preliminary Results

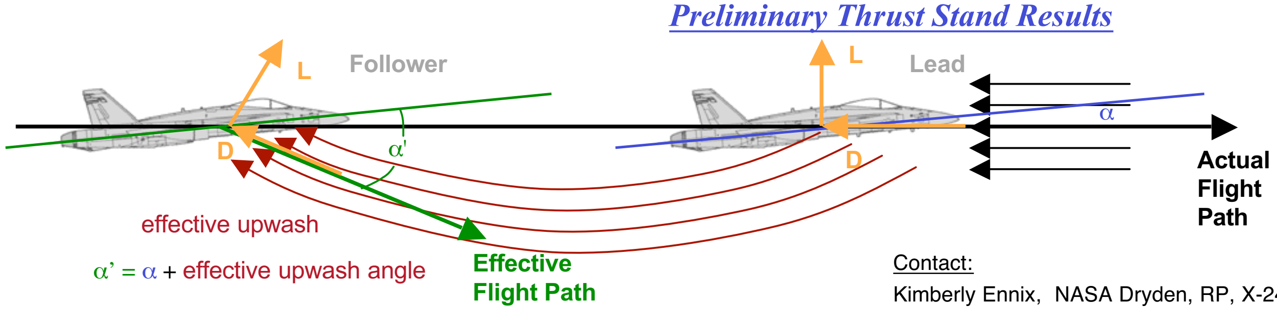
•Completed installation and check-out of engine specific performance instrumentation:
- Flight test fuel-flow meters
- 20 probe PT56 (turbine discharge pressure) rakes

•Developed real-time and post flight performance models
- Verified real-time models and displays.

•Successfully completed thrust stand ground test
- Validated In-flight Thrust Model.



Preliminary Thrust Stand Results



The Physics of Formation Flight (over-simplified)

The upwash effect rotates the Lift and Drag vectors reducing the overall thrust and thus, the fuel flow required to maintain level flight.

Contact:
Kimberly Ennix, NASA Dryden, RP, X-2479
Ron Ray, NASA Dryden, RP, X-3687
Kevin Walsh, NASA Dryden, RP, X-3686
Clint St. John, NASA Dryden, RP, X-5306
Jake Vachon, NASA Dryden, RP, X-2450

F-15B Propulsion Flight Test Fixture (PFTF)

The F-15B Propulsion Flight Test Fixture (PFTF) is a unique, low-cost flight facility for the development and flight test of advanced propulsion systems (Figure 1). Flight research data can be obtained at subsonic, transonic, and supersonic speeds up to about Mach 1.8 and 1,100 psf dynamic pressure. Propulsion experiments, including "cold" (unfueled) and "hot" (fueled) experiments such as inlet research, Rocket Based Combined Cycle (RBCC), and Pulse Detonation Engine (PDE) experiments may be tested. The PFTF incorporates an extensive instrumentation and telemetry system, including a 6-component, in-flight force balance.

Figure 1. NASA F-15B with PFTF Propulsion Experiment (conceptual).

The PFTF has two main components: the pylon and the flight experiment. The pylon provides volume for instrumentation and systems for the flight research experiment. The complete PFTF, pylon and experiment, has a maximum weight of 1600 lbs. The maximum allowable experiment weight is approximately 500 lbs. Experiment dimensions of approximately 107 inches in length and 12 inches in diameter can be accommodated.

There are three main bays within the PFTF pylon. Each bay contains a removable rack which allows for ease of experiment systems integration and access. The bays can house instrumentation and experiment systems including tankage, valves, plumbing, etc.

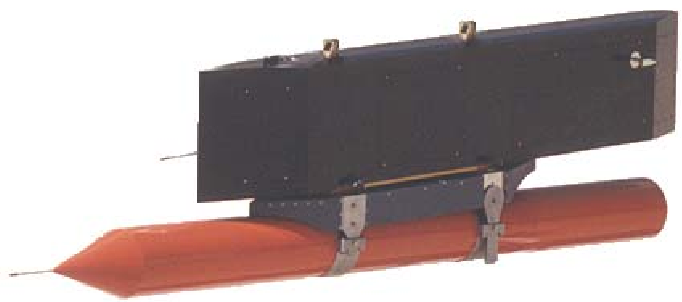


Figure 2. PFTF with Cone-Tube Experiment.

Standard PFTF instrumentation include measurement of pressures, temperatures, and accelerations. The data can simultaneously be recorded on board the aircraft and telemetered to a ground control room for display and recording. On board signal conditioning and instrumentation power is provided. Individual data requirements can be integrated into the PFTF.

Envelope expansion flight tests of the F-15B/PFTF will be conducted in the summer of 2001. These flights will be used to assess the flying qualities and performance capabilities of the F-15B/PFTF configuration. A simple cone-tube experiment will be carried to verify the function of the in-flight force balance (Figure 2). The aerodynamic flow quality underneath the pylon will also be investigated.

Planned PFTF flight tests include cold flow inlet tests of a simple normal shock inlet with measurements using a distortion rake (Figure 3). Flight tests of a methane-oxygen fueled rocket based combined cycle (RBCC) experiment are planned in 2002.

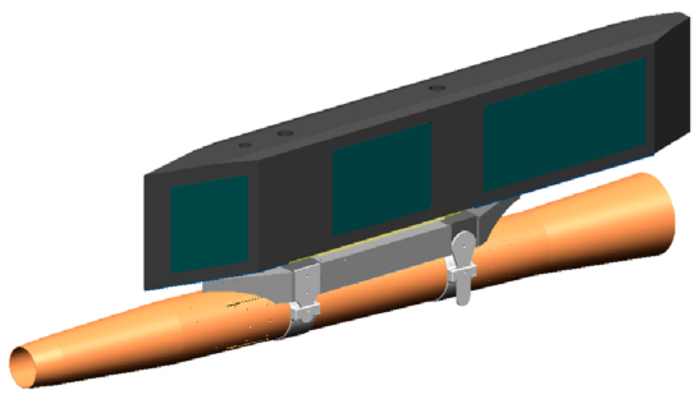


Figure 3. PFTF with Normal Shock Inlet Experiment.

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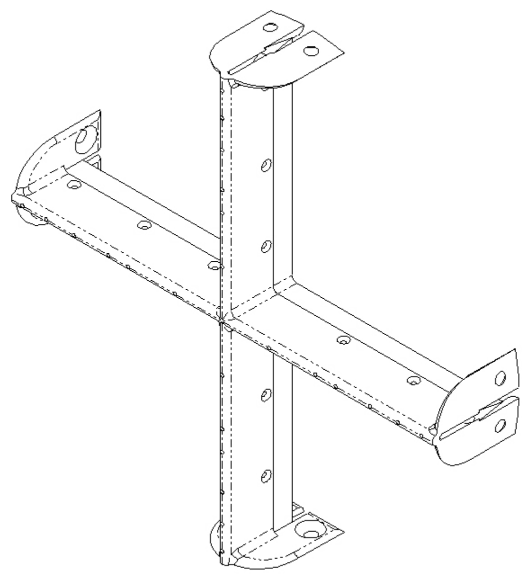
F-15B Propulsion Flight Test Fixture (PFTF)

Inlet Total Pressure Distortion Rake

NASA F-15B with PFTF Experiment (conceptual)

The Propulsion Flight Test Fixture (PFTF), flown on the F-15B, provides a unique, low-cost flight facility for the development and flight test of advanced propulsion systems. Flight research data can be obtained at subsonic, transonic, and supersonic speeds up to about Mach 2.0 and 1400 psf dynamic pressure. Propulsion experiments, including "cold" (unfueled) and "hot" (fueled) experiments such as inlet research, Rocket Based Combined Cycle (RBCC), and Pulse Detonation Engine (PDE) experiments may be carried. The PFTF incorporates an extensive instrumentation and telemetry system, including an in-flight force balance for the direct measurement of propulsive thrust and aerodynamic drag. This unique feature may be useful for aerodynamic and other flight experiments in addition to propulsion research.

To facilitate development of the PFTF as a research testbed, a Pitot-type total pressure distortion rake has been designed to provide a low-resolution map of the internal flowfield of a baseline propulsion experiment. The rake can be mounted in several positions on the PFTF. Depending upon the experiment, the rake can be placed upstream of the main propulsive device to map the distortion behind research inlets or downstream of the main propulsive device to map distortion associated with the device itself and support structure.



PFTF Inlet Total Pressure Distortion Rake

In concert with the PFTF design philosophy, the rake had to be inexpensive, modular, and relatively simple to incorporate. The rake is designed similar to the inlet distortion rake pioneered on the NASA High Alpha Research Vehicle (HARV) and incorporates a simplified wagon-wheel design with high- and low-response pneumatic probes. The rake provides 21 total pressure measurements on four spokes. Four of the pressure measurements are taken by high-response, temperature-compensated pressure transducers. One thermocouple is also installed on each of the four rake spokes. A single Pitot-static probe is mounted at the rake centerbody location..

The rake will fly on the PFTF as part of Phase II, scheduled for testing during FY '02.

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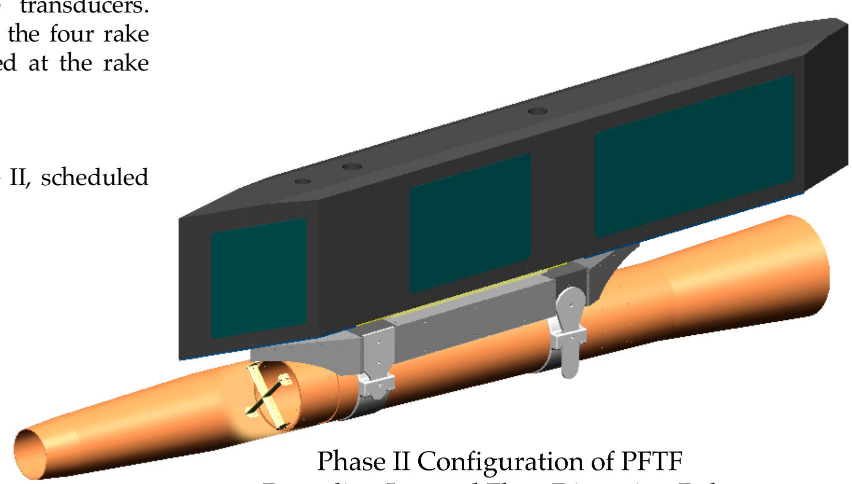


Phase II Configuration of PFTF
Revealing Internal Flow Distortion Rake

The Rocket Vehicle Integration Test Stand (RVITS)

Summary

NASA Dryden, in conjunction with the AFFTC and AFRL, is establishing a Rocket Vehicle Integration Test Stand (RVITS) at the site of the historic X-15 Engine Test Stand.

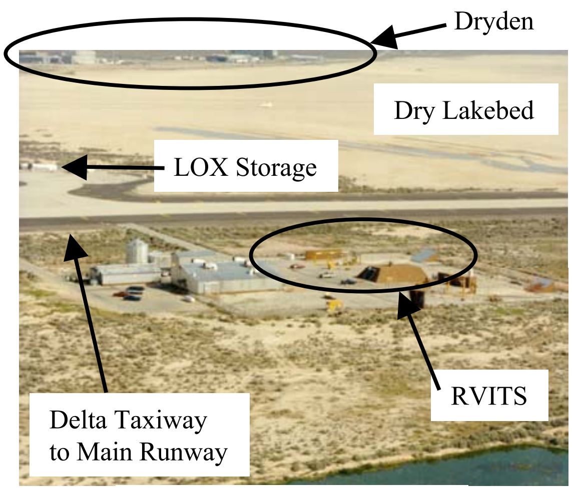
Objective

Edwards AFB will reestablish the capability of supporting preflight operations for Space Launch Initiative (SLI) programs and related technologies. Potential vehicles requiring this facility in the near future may include X-34, X-37, RBCC, and PDE. This facility will be used to:

- Provide fully integrated vehicle validation
- Troubleshoot after propulsion system anomalies and modifications
- Reduce technical and operational risks
- Hot fire installed engines in a controlled environment that is LOX compatible

Justification

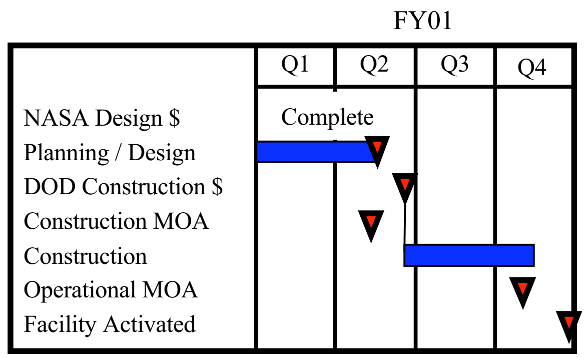
NASA is undergoing an agency-wide push to develop new and advanced Access to Space technologies. Dryden has a key role in the flight development of these technologies. RVITS will provide a critical ground test facility at Edwards for supporting flight operations, and conduct integrated vehicle/propulsion system check-out of Access to Space vehicles. Ground Test Vs. Flight Test engine run time has shown that even a well developed engine (such as the XLR-99 in the X-15) will have anomalies which result in required ground testing prior to powered flights. Since ground tests will be required during powered flight research, RVITS will continue to minimize costs by being optimally located near both ground testing and powered flight locations.



RVITS Area - Aerial View

Approach

The process and schedule for RVITS rehabilitation is shown below. The project has completed feasibility, clean-up, and assessment phases. NASA Dryden has provided funds for RVITS design work which will lead to completion of the general requirements of a typical rocket vehicle. Upon facility activation, these baseline requirements may be supplemented with enhancements as necessary. Design work by AFRL/Sverdrup is now nearing completion ahead of schedule. Construction is being funded by the DoD. A Memorandum Of Agreement (MOA) is being drafted that incorporates the strength of the Edwards Alliance, utilizing our rocket testing infrastructure and experience of the AFFTC, AFRL (Phillips Lab), and NASA Dryden. The X-34 is anticipated to be the first customer of the new RVITS facility.



RVITS Process and Schedule

RVITS Test Stand Before Rehabilitation

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02/08/01

Development of Scramjet Skin Friction Gages

Summary

NASA Dryden is working with Virginia Polytechnic Institute and State University (Virginia Tech) to develop a skin friction gage for scramjet flight test application on the X-43 Mach 10 research vehicle. Although the Virginia Tech skin friction gage has been used extensively in scramjet wind tunnel tests, it has never been used in flight. As the results, an extensive development, testing, and validation program is being pursued to help develop the best possible gage technology for the X-43 scramjet flight application.

Objective

Develop and validate sensor technologies for skin friction measurement inside scramjet engines of hypersonic flight vehicles such as the X-43.

Justification

Surface skin friction force is one of the important forces affecting hypersonic flight vehicles and their propulsion systems. For scramjet engines such as those used in the X-43, scramjet skin friction drag is a significant portion of the net engine thrust. Accurate measurement of the skin friction force is important to validate analysis tools, provide wind tunnel to flight correlation, and determine scramjet engine efficiency.

Approach

In the proposed approach, two skin friction gages and one heat flux gage are installed on the body side in the internal nozzle section of the X-43 Mach 10 research vehicle. The heat flux gage, when tested together with the skin friction gages, allows the Reynolds Analogy to be tested for the Mach 10 scramjet nozzle flow. The heat flux gage also provides some redundancy for the skin friction gages, since the skin friction stress can be deduced from heat flux measurements.

Results

Based on recent results from the scramjet skin friction gage tests at GASL and NASA Langley, as well as the F-15 skin friction gage flight tests at NASA Dryden, a new skin friction gage design is being designed by Virginia Tech for use in the X-43 Mach 10 research vehicle. A cut-away view of the gage is shown in figure 1. The top of the sensor head (red) is mounted flush with the X-43 engine wall. As the sensor head is displaced by the skin friction shear, it produces stresses on the spokes of the wheel (green) which is rigidly attached to the sensor head. Outputs from strain gages mounted on the spokes of the wheel indicate the shear stress levels.

This gage design offers many advantages to previous gages. The beam-wheel arrangement results in a shorter, more compact gage. Also, increased gage sensitivity is possible with this gage design, allowing the use of normal foil-type strain gages. Finally, the counter-balancing mass (blue) acts both as a heat sink and a dynamic mass damper. As the results, water cooling and oil fill are not required.

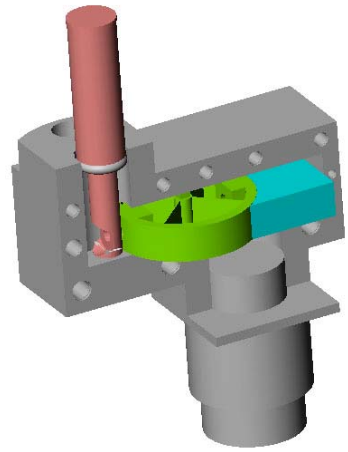

Figure 1. Cut-Away View of the X-43 Mach 10 Skin Friction Gage.

To measure heat flux simultaneously with skin friction, a new heat flux gage is also being designed and built for this experiment. The design must work well for the thick wall and highly transient heat transfer conditions inside the X-43 scramjet engine. Figure 2 shows a prototype of a heat flux gage from Virginia Tech that would be used in the flight test. The heat flux gage will install into the flight vehicle the same way as the skin friction gage.

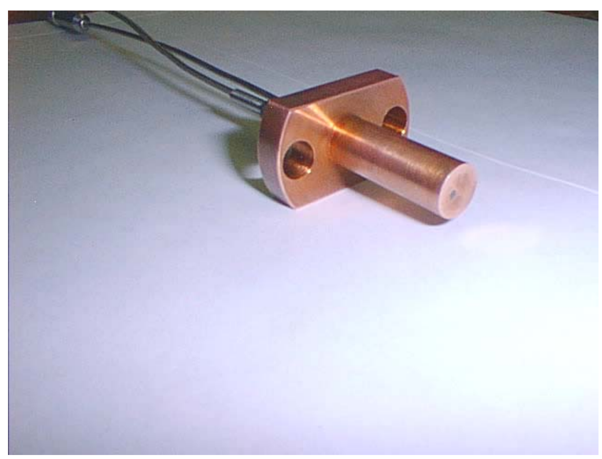

Figure 2. X-43 Mach 10 Heat Flux Gage.

Work is underway to machine gage installation holes in the X-43 Mach 10 flight engine. Plans are also being made to incorporate the skin friction and heat flux gages into the flight vehicle signal conditioning system. After the gages are built, they will be evaluated in ground tests at Virginia Tech, NASA Langley, and NASA Dryden. Final installation into the X-43 research vehicle is currently planned for April 2001.

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F-15 Skin Friction Flight Test

Summary

NASA Dryden is working with Virginia Polytechnic Institute and State University (Virginia Tech) to develop a skin friction gage for scramjet flight test application on the X-43 Mach 10 research vehicle. Although the Virginia Tech skin friction gages has been used extensively in wind tunnel testing on the ground, they have never been used on a flight vehicle. The F-15B/flight test fixture (FTF) provides an excellent opportunity for testing the Virginia Tech gage concept in the flight environment.

Objective

Evaluate the Virginia Tech F-15 skin friction gage concept in flight using the F-15B/FTF.

Justification

Surface skin friction force is one of the important forces affecting hypersonic flight vehicles and their propulsion systems. For scramjet engines such as those used in the X-43, skin friction drag is a significant portion of the net engine thrust. Accurate measurement of the skin friction force is important to validate analysis tools, provide wind tunnel to flight correlation, and determine scramjet engine efficiency. The F-15B flight tests provide valuable experience in developing skin friction gages for flight applications.

Approach

The configuration of the skin friction sensor complex is shown in figure 1. This complex was designed to fit into existing 8-inch hatches on either side of the FTF, facilitating joint flight testing with other FTF experiments. The complex includes the boundary layer rake, RTD temperature sensors, heat flux gages, and a Preston tube. The skin friction is measured simultaneously by the skin friction gage, the Preston tube (using the Preston tube method), and the rake (using the Clauser plot method). The flight conditions are chosen so that the flow over the FTF II approximates the simple flat plate flow, and a good estimate of the skin friction can be obtained using the Preston tube and the Clauser plot methods. These estimates can then be used to evaluate the accuracy of the Virginia Tech F-15 skin friction gage.

Results

In FY00, the F-15B/FTF skin friction sensor complex has been flown piggy-backed with the F-15B/FTF Hot Wire and Boeing Door Seal experiments. A dedicated skin friction flight was also conducted in December, 2000. Extensive flight data were obtained for flight Mach numbers ranging from 0.5 to 1.7 and at altitudes between 5000 ft and 45,000 ft. Good turbulent boundary layer correlation with theory was obtained for subsonic flights. For supersonic flights, possible shock-boundary layer interactions on the FTF caused the boundary layer to differ significantly from the typical flat plate equilibrium boundary layer profile, and no correlation was possible.

Figure 2 summarizes the typical subsonic boundary layer rake and the Preston tube results for an F-15B flight. It can be seen that consistent and repeatable shear levels can be obtained on the FTF.

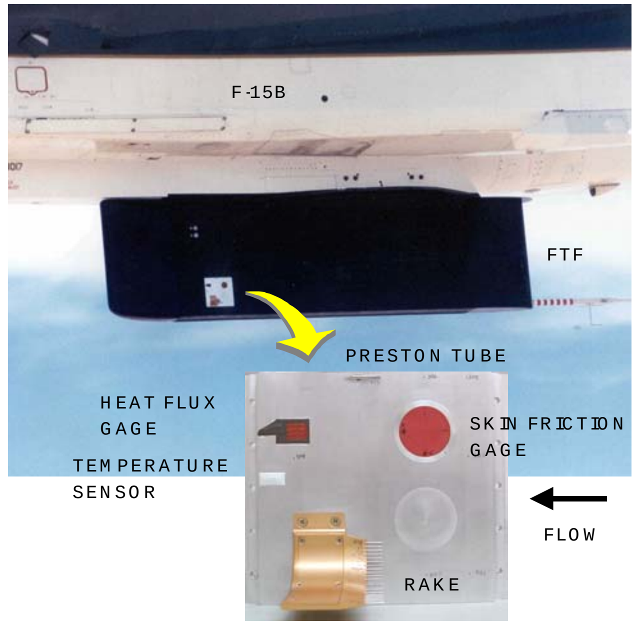

Figure 1. Flight Test Configuration for the F-15B/FTF Skin Friction Sensor Complex.

The shear levels obtained from the rake and the Preston tube data were used to evaluate the accuracy of the skin friction gage. The skin friction gage was found to have significant and unacceptable sensitivity to ambient air temperature changes in flight. This was caused primarily by the red rubber sheet which covers the gage sensing surface. The rubber sheet was used to dampen in-flight vibrations. However, it also contracts and expands in flight due to temperature changes, producing large artificial changes in gage outputs. As the results, it is recommended that rubber or polymer materials are avoided in the construction of skin friction gages for flight applications.

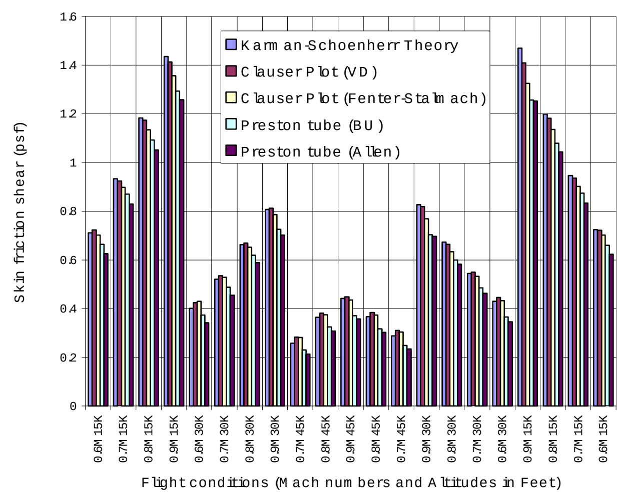

Figure 2. Preston Tube and Rake Results for F-15B Flight Number 179.

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Regression of Loads Derivatives Using F-18/SRA AAW Early PID Flight Data

Summary

An evaluation of the F-18 baseline loads model was performed as part of the Active Aeroelastic Wing (AAW) Risk Reduction Experiment on the Systems Research Aircraft (SRA) aircraft. The F-18 loads model was developed by Boeing for use with the flight simulator to determine loads at discrete locations on the aircraft. The loads model was modified to reflect the increased flexibility of the AAW aircraft so the model could be used in the development of the AAW control law. The control laws are being developed to maximize roll performance while maintaining load limits. Currently, the loads models for both the AAW aircraft and the baseline F-18 are predicting higher than expected loads especially for the outboard leading-edge flap which has become the limiting factor in the AAW control law development. In order to ensure the success of the AAW program, a reasonably accurate loads model must be developed. The purpose of this experiment was to develop a method for reducing loads derivatives, which can be used to create a more accurate loads model.

Approach

Three control surface hinge moments were used in this experiment: the right outboard leading edge flap, the left aileron, and the left stabilator. These measured control surface hinge moments were monitored in flight along with loads model calculated values. The loads model predictions were obtained by incorporating the loads model and database into the real-time F-18 simulator and using 23 aircraft measured flight parameters including: Mach, altitude, and control surface deflections as inputs to the model.

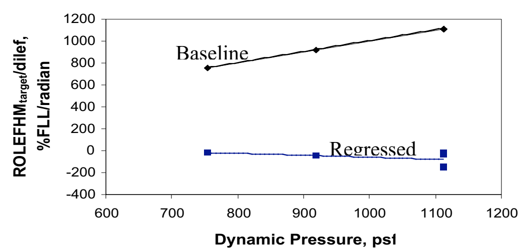
Longitudinal and lateral directional doublets were performed during which each individual control surface was deflected. The flight-measured hinge moments were compared to loads model calculated hinge moments at twenty flight conditions ranging from Mach .85 to 1.3 and altitude from 5000 to 25,000 feet. Then, based on the comparison, a linear regression technique was used to establish the load derivatives, which is an effect on a load due to a unit change in an aircraft parameter such as control surface deflection.

Results

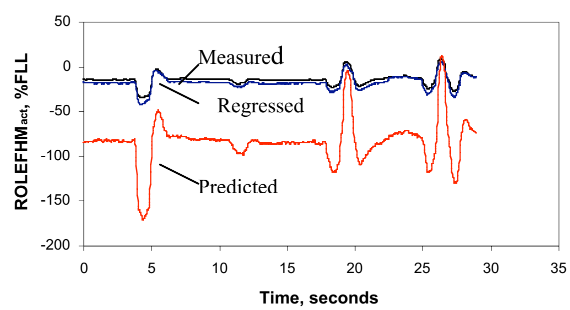
Flight test has been completed on the F-18 SRA. The first stage of data analysis, which compared measured and predicted hinge moments at each test point, showed that the loads model was over predicting the measured load by as much as 130% of the absolute value of the flight limit load (FLL_a) subsonically and 100% FLL_a supersonically for the leading edge flap hinge moment. This technique was used to develop the 23 loads derivatives at 20 flight conditions for this control surface (see figure). The new load derivatives were placed back in the loads model to create a new regressed load which reduced the error to within 10%FLL_a subsonically and 2%FLL_a supersonically.

Future Plans

This technique will be used during the F-18 AAW Parameter Identification flights to create a new loads model for all of the aircraft measured loads. This loads model then will be used in the development of the AAW control laws.



Otbd LEF Hinge Moment due to Inbd LEF Deflection vs. q at 0.95 Mach



Contact

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Service Life Analysis of X-38 Hooks

Summary

The B-52 pylon hooks are used to carry different air launched vehicles (e.g., X-38, Pegasus rocket, X-43). Each hook has a critical stress point where micro cracks could initiate after excess number of test flights (cumulated load cycling). For safety purposes, the fatigue life (i.e. the number of available flights) of each hook must be established before any flight-test program.

The present fatigue analysis focuses on the X-38 hooks. The current results show that the X-38 hooks have considerable fatigue life for safe flight-test operations of the X-38 captive carry on the NASA Dryden B-52.

Objectives

To perform finite-element hook stress analysis and then service life (fatigue life) analysis of the X-38 hooks and to set the limit of "safe" number of test flights permitted for the X-38 hooks without failure.

Approach

The hook fatigue (service life) analysis requires finite element stress analysis of each hook to locate the critical stress point, and to establish the equation describing the tensile stress at the critical stress point as a function of the hook load for each hook. Then, to calculate the fictitious initial crack size based on the limit hook load using the fracture mechanics. Next, using the random load spectrum obtained from the first flight, and the half cycle theory to calculate the fictitious crack growth after the first flight. Finally, to calculate the remaining flights available for the X-38 hooks.

Results

- The X-38 hook was found to fail in tension rather than in shear.
- The critical tensile stress point is located on the X-38 hook inner boundary at $\theta = 24.75°$ for the concentrated hook loads, where θ is an angle measured from the horizontal axis.
- The maximum shear stress point is located in the cross section near the root of the X-38 hook horizontal arm and is near the upper horizontal boundary.
- The service life of each X-38 hook is predicted to be approximately 231 flights.

Contact

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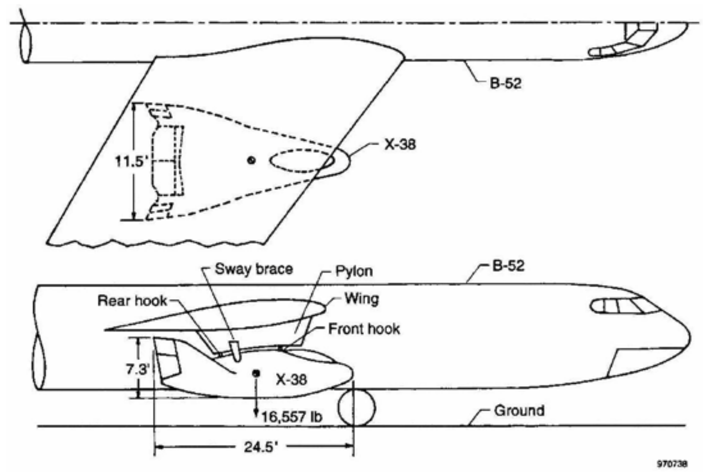
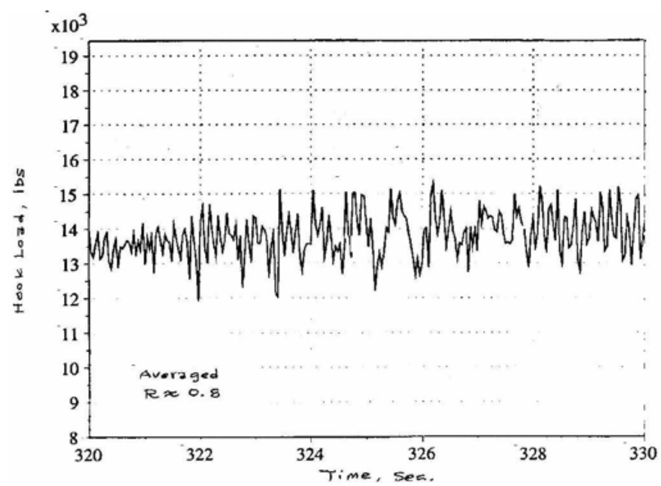
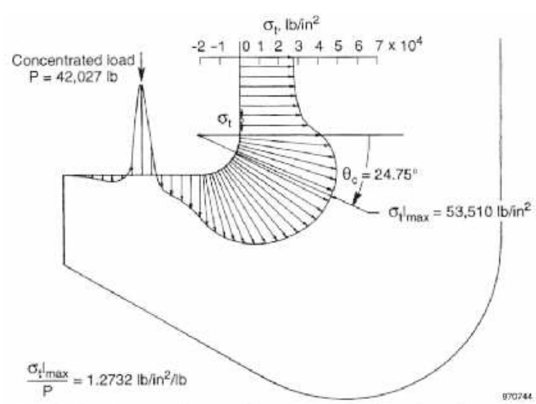


Sideview of the X-38 lifting body atmospheric test vehicle mated to the B-52 pylon through two new hooks.



Hook loading cycles during take-off run of B-52 carrying X-38 V.131 vehicle.



Distribution of tangential stress σ_t along inner boundary of the X-38 hook; concentrated load $P = 42,027$ lb.

Carbon Composite Control Surface Test Program

Summary

Work continued in the Flight Loads Laboratory (FLL) to establish a 3000°F test capability in support of the Spaceliner 100 Carbon Composite Control Surface (CCCS) test program. The significant areas of development included the development of an optical temperature measurement technique, the evaluation of quartz lamp heaters, and the completion of a full-up lamp/zone check.

An optical temperature measurement technique was developed utilizing existing lightpipe sensor technology. The sensor was successfully used in temperature control up to 2400°F. The lightpipe sensor correlated well with conventional thermocouples.

The performance of the quartz lamp heating system was successfully evaluated in a small nitrogen chamber. The heaters were used to heat a carbon-carbon test specimen to 2400°F in a nitrogen atmosphere. Additionally, the heaters being used to heat the CCCS successfully completed a full-up lamp and zone check.

Objectives

- Establish a 3000° F nitrogen atmosphere test capability.
- Develop high-temperature instrumentation for use on C/C and C/SiC materials.
- Develop high-temperature test techniques for testing structures up to 3000°F.
- Perform thermal/mechanical test up to 3000°F.
- Support the Spaceliner 100 carbon composite structures development work through test and analysis of the CCCS.

Benefits

- Provides valuable test data to the technical community on a flight-weight carbon composite control surface.
- Establishes methods of testing oxidation sensitive structures at high temperatures.
- Serves as a testbed to test innovative sensors.

Contacts

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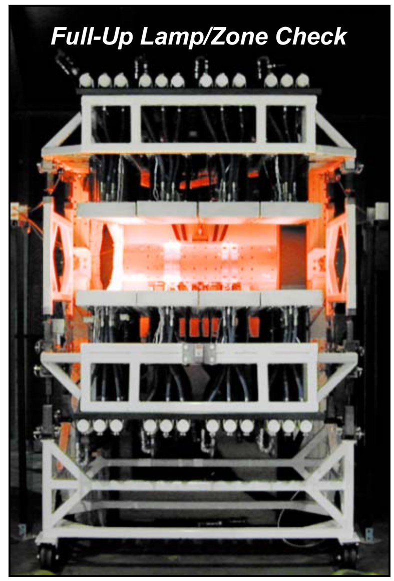
Tom Horn, Test Engr., DFRC, (661) 276-2232
Dr. Wayne Sawyer, Project Engr., LaRC, (757) 864-5432

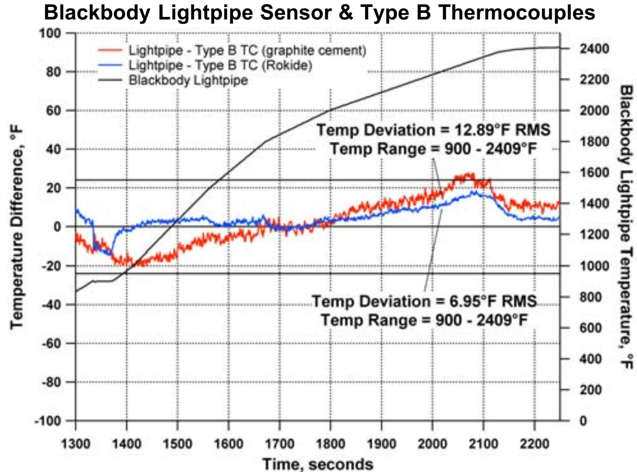

Carbon Composite Control Surface


Full-Up Lamp/Zone Check


Heater & Lightpipe Sensor Evaluation


2400°F Heating Test in the Small Nitrogen Chamber Testing

Temperature Comparison Between a Blackbody Lightpipe Sensor & Type B Thermocouples



Load Pad Development for the AAW Wing Strain Gage Loads Calibration Test

Background:

In preparation for the Wing Strain Gage Loads Calibration Test on the Active Aeroelastic Wing (AAW) flight research program, some material testing was completed to develop adequate load pads and a bonding agent for the loads calibration test. One hundred and four load pads will be symmetrically bonded to the lower wing surfaces over the main wing box, and leading and trailing edge control surfaces of a modified F-18. These load pads will cover 83 ft^2 or 60% of the lower surface of each wing and provide the elastomer interface to the AAW aircraft for the tensile and compressive single point and distributed loads. The Loads Calibration Test will begin in late Spring of 2001.

Objectives:

The objectives of the load pad development research were to obtain an elastomer interface material and bonding adhesive that would be used in the AAW Loads Calibration Test. Desired characteristics include:

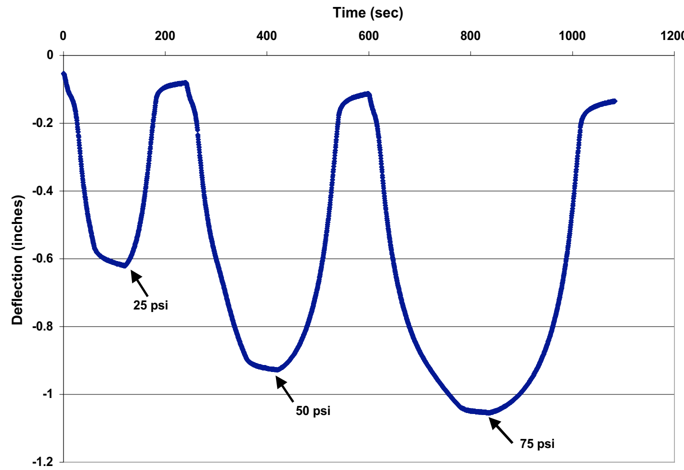
- Load Pads ⇒ Neoprene material with ideally 2" thickness and has "safe" compressive characteristics and high tensile and creep strength characteristics.
- Adhesive ⇒ Adequate strength, removable adhesive with rubber, metallic, and composite bonding capabilities and can withstand cyclic loading.

Approach:

Several material tests were conducted on five types of Neoprene rubber ranging from 1" to 2" thicknesses and varying durometer (hardness of the material) scales. Two adhesives were utilized throughout these tests. The following tests were conducted in Dryden's Flight Loads Laboratory (FLL):

- Compression ⇒ Testing to determine the load pad deformation and Young's Modulus under compressive loads.
- Creep ⇒ Testing to determine that the load pads could be placed under tensile load for a period of time without tearing or splitting apart. Secondly, testing to determine the bonding reliability of the adhesive along with gaining experience with bonding applications.
- Ultimate Strength ⇒ Testing to determine that the load pads could withstand "sufficient" tensile loads without failure. Secondly, test to gain confidence in the bonding reliability of the adhesive under high tensile loads along with additional experience with the bonding procedures.
- Cyclic ⇒ Testing to determine the rubber and adhesive bonding reliability and fatigue strength under low-cyclic loading.
- Pad Removal ⇒ Testing to develop a technique for removing the load pads from the AAW aircraft.

Initially all five Neoprene pads types were to undergo the above testing, but as pads started to fail necessary AAW Loads Calibration Test requirements they were eliminated from the testing matrix.



Compression Data for the 1.5" thick Rubberlite Load Pad

2" thick Neoprene load pad failed during Ultimate Strength Testing using a Material Test Stand (MTS) machine

Results:

The load pad that met all the material properties for the AAW Loads Calibration Test was Rubberlite's 1.5" thick, 30-50 Shore OO Neoprene rubber. Under compressive loads the Young's Modulus is non-linear and the pad deflects 0.6" at 25 psi which is the maximum expected compressive pressure during the ground test. During tensile loading the pad elongates 0.2" at 25 psi which is the maximum expected tensile pressure for the loads test. A factor of safety of 2 was determined for the ultimate strength of the load pad. The adhesive which will be used in the loads test is B-1/2 polysulfide fuel tank sealant. This adhesive turned out to be gap filling, more reliable, and easier to remove than the industry's standard contact cement type of adhesive.

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Radiant Heat Flux Gage Calibration System Characterization

Summary

Heat flux measurements typically have large uncertainties (typically 10% to 20%) associated with them. This issue was further highlighted in 2000 when a heat flux gage vendor claimed a difference of 15% between the vendors calibration technique and the technique used at the National Institute of Standards and Technology (NIST). Further testing at other facilities has revealed a similar difference between recent NIST calibrations and heat flux measurements derived from slug calorimeters.

The developments mentioned above have caused renewed interest in DFRC heat flux calibration research. The Marshall Space Flight Center (MSFC) Engineering Directorate has requested Dryden support in resolving the issues surrounding heat flux calibrations as applied to materials qualification testing at MSFC. High-quality heat flux measurements are required at MSFC in order to ensure that the proper boundary conditions are imposed on materials test samples. Large heat flux measurement uncertainty requires conservative test conditions, which may result in excess thermal protection system weight on vehicles such as the Space Shuttle.

Plans for 2001 include continued development of the blackbody cavity thermal models. Model development will be supported with additional testing as required. An attempt to duplicate and analyze NIST and industry calibration techniques will also be undertaken.

Recent Developments:

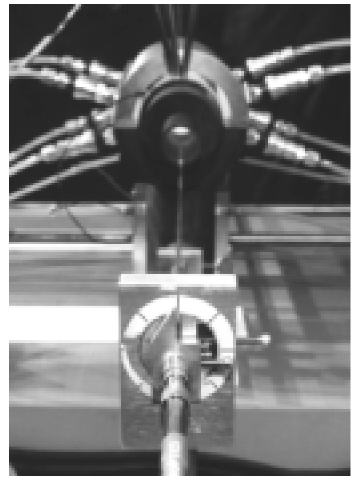
- Publication of papers describing the experimental and numerical characterization of the blackbody cavity @ 1100°C (2012°F) and flat plate heater. (see references)
- A heat flux gage vendor discovered a difference between NIST heat flux calibrations and calibrations obtained by other methods as well as heat flux measurements derived from slug calorimeters.

Future Work:

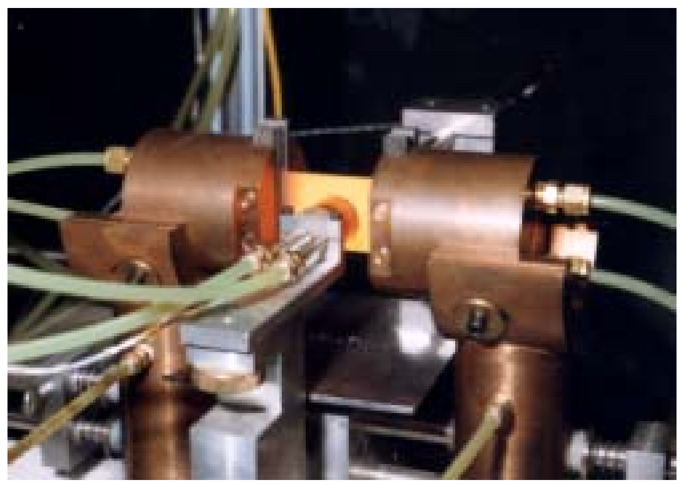
- Determine how DFRC calibration techniques compare to NIST and other calibration/measurement methods.
- Develop 3-D model of the flat plate heater for use in evaluating calibration configurations
- Develop transient numerical models of blackbody/heat flux gage calibration process
- Obtain Electrically Calibrated Radiometer in order to duplicate NIST calibration process.

Contact:

Thomas J. Horn, NASA Dryden, RS, (661)276-2232



Blackbody Source with Wall Temperature Measurement Probe



Flat Plate Heat Source for Heat Flux Calibration Transfer

References:

Jiang, S., Horn, T. J., and Dhir, V. K., 1998, "Numerical Analysis of a Radiant Heat Flux Calibration System," HTD-Vol. 361, *Proceedings of the ASME Heat Transfer Division, V. 5*, pp. 609 – 616.

Jiang, S. Horn, T. J., and Dhir, V. K., 2000, "Numerical Analysis of a Radiant Heat Flux Calibration System," International Journal of Thermophysics, Vol. 21, No. 4, pp. 941 – 963.

Horn, Thomas J. and Abdelmessih, Amanie N., "Experimental and Numerical Characterization of a Steady State Cylindrical Blackbody Cavity at 1100 Degrees Celsius", NASA/TM-2000-209022, August 2000.

Vortex Effects of F/A-18 Formation Flight

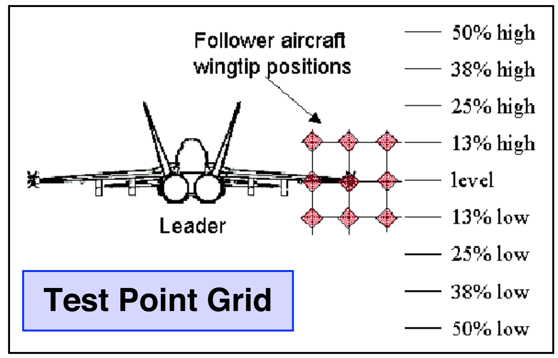
Summary

The vortex influence of formation flight was mapped as part of the Autonomous Formation Flight (AFF) project. The vortex effects on the trail aircraft's moments and side force were mapped at two flight conditions and four longitudinal distances. With the pilot in-the-loop, 27 flights and more than 280 test points were accomplished using NASA's Systems Research Aircraft (SRA) as the formation lead, and the NASA 847 support aircraft as the trail aircraft.

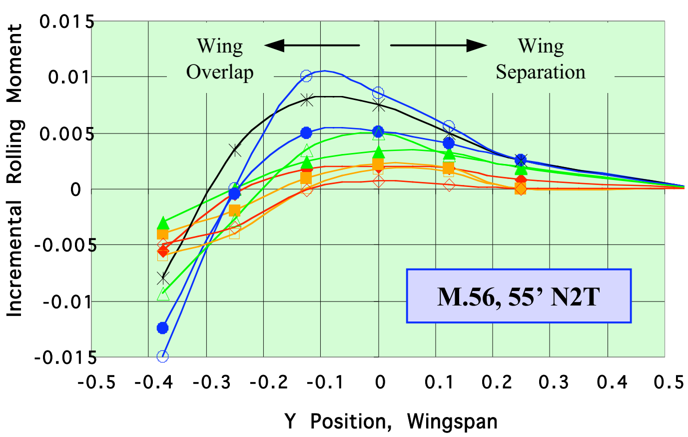
Objective

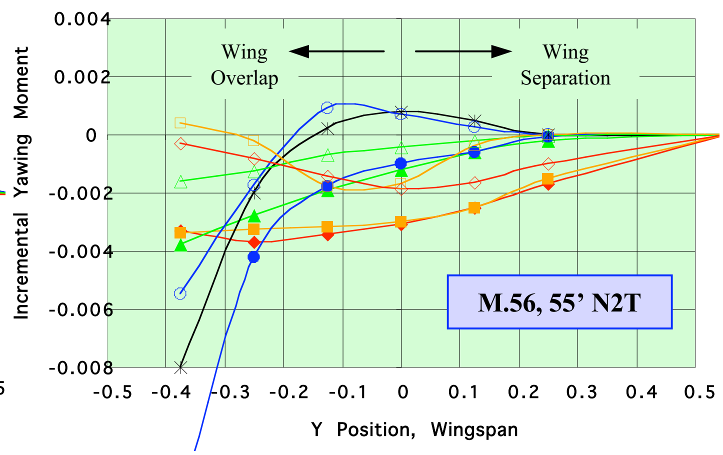
The goal of the project was to map the vortex influence on the rolling, pitching and yawing moments, and side force, as a function of flight condition (subsonic: Mach=0.56 @ 25,000 ft or transonic: M=0.86 @ 36,000 ft) and position in the vortex. Each test point was to be flown with the pilot in full control of the aircraft.

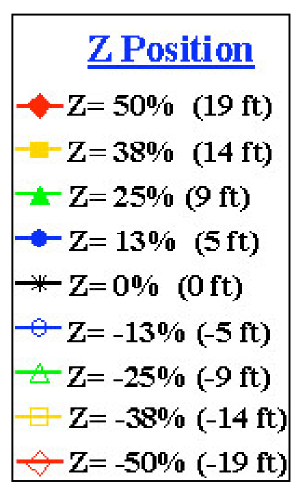
Approach



Each aircraft was outfitted with a GPS receiver, an air-to-air telemetry system and an advanced guidance system to give the pilot of the trail aircraft real-time position data. Using the ILS (Instrument Landing System) needles in the HUD, the pilot guided the trail aircraft into position behind the formation lead. To acquire a complete map of the vortex effects on the trail aircraft, stable test points were defined over a test point grid in the lateral-vertical (Y-Z) plane. The test points were spaced about 13% of a wingspan (~4.9 ft) apart, and were flown at four longitudinal distances (X = 20', 55', 110' and 190' measured nose-to-tail).







Results

The vortex-induced incremental side force and moments first peaked at a lateral position of 13% wing overlap and vertical positions of 0% (level) and 13% below the lead aircraft. At the subsonic flight condition, the flight data also suggests the vortex effects with increased X distance were weaker in pitch and roll, but stronger in yaw and side force when compared with those at the closer X distance. At the transonic flight condition, the vortex effects were weaker at close X distances. Further away from the formation lead, the vortex effects were once again weaker in pitch and roll, but stronger in yaw and side force when compared with the effects at the subsonic condition.

The peak vortex effects coincided with the position for maximum drag reduction, and the incremental force and moments were more sensitive to formation position as wing overlap increased. Despite this, the vortex effects were well within the capability of the pilot; and therefore, the aerodynamic effects did not appear to jeopardize the success of a formation flight controller design for this type of aircraft.

DFRC Contact: Jennifer Hansen x2052 or Brent Cobleigh x2249

Meteorological Summary of the Helios Prototype 100,000-Foot Mission

Summary

In the summer of 2001, the Helios Prototype Solar Powered UAV was deployed to the Pacific Missile Range Facility (PMRF), Kauai, HI, in an attempt to fly above 100,000 feet, a NASA ERAST Level-I milestone. In support of this goal, meteorologists from the NASA Dryden Flight Research Center were sent to PMRF, as part of the flight crew, to provide current and forecast weather information to the pilots, mission directors and planners. This information is critical to maintain a successful and safe flight operation. In general, the primary weather concern for ground and flight operations are wind speeds. Due to the aircraft's long wingspan (247') and low weight (1500-1600 pounds), the Helios is sensitive to wind speeds exceeding 7 knots at the surface. Also, clouds are a concern due to effects of sun blockage on the solar array that provide power to the aircraft. Vertical wind shear is very closely monitored in order to prevent damage, or loss of control, of Helios due to turbulence. Two flights were successfully accomplished at PMRF during the deployment. The last flight, on August 13-14, 2001, set a world altitude record for a non rocket-powered aircraft, climbing to 96,863 feet (over 11,000 feet higher than the SR-71 record).

Objective: To provide flight team current and forecast weather conditions that will improve the likelihood of a safe and successful flight test operation.

Justification: Meteorological support is critical to help optimize flight conditions for aircraft peak performance to achieve mission success and avoid adverse weather conditions.

Approach:

- Issue daily meteorological forecasts of surface and upper level conditions starting 48 hours before flight day.
- Provide current and forecast conditions at crew brief.
- Provide early morning weather briefing for go/no-go to rollout aircraft from hangar.
- Final weather go/no-go briefing given 2 hours prior to takeoff is issued.
- After takeoff, periodic updates based on weather balloon and satellite data are provided to the pilot and mission planner.
- Approximately 2 hours before landing a final weather forecast is issued to estimate earliest possible landing time and a selection of a runway.
- After landing, surface conditions are monitored until aircraft is safely in the hangar.

Status: Two successful flights were accomplished

July 14, 2001:

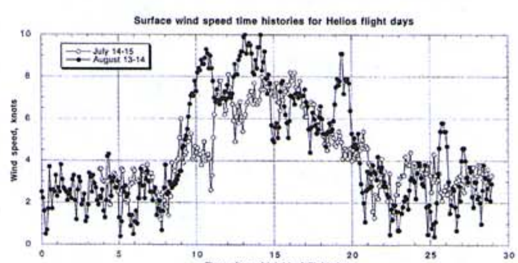
- 20-minute delay in takeoff due to clouds.
- Convection over runway at takeoff generated moderate turbulence during takeoff.
- Aircraft climbed to maximum altitude of 78,000 feet.
- Aircraft landed in stable conditions after 15+ hours of flight.

August 13, 2001:

- 45-minute delay in takeoff due to clouds.
- Strong wind shear at 2000 feet due to strong trade winds and island wake.
- Aircraft climbed to new altitude record of 96,863 feet.
- Aircraft landed safely after a last minute change in runway due to winds.
- Flight satisfied NASA milestone for extreme altitude mission.

Cloud cover as seen from the runway after sunrise

Helios Prototype takeoff on its way to a record altitude flight. August13, 2001



Surface wind time histories for Jul 14-15, Aug 13-14.

Contacts: Casey Donohue, AS&M, X-2768

Active Aeroelastic Wing

Summary
The Active Aeroelastic Wing (AAW) technology demonstration program proposes a synergistic benefit of reduced wing internal loads during maneuvering flight by utilizing elastic wing twist, active controls, and aerodynamics in an integrated fashion. A new technique for development of a control surface mixing strategy is being examined.

Objective
The objective is to develop control laws for the F-18 AAW aircraft that produce fighter type roll performance without using the rolling tail.

Justification
The development of flight proven reductions in internal wing loads can lead to airplane designs with lighter wings.

Benefits
This approach develops an optimal trajectory from wings level roll attitude to a specific bank angle in a specified time. It utilizes the airplane's full linearized dynamics in the optimization problem. The technique does not produce a control strategy directly, but requires that the control system designer extract the control system mixer from multiple optimum trajectories.

Approach
Previous attempts for control law design concentrated on developing a control surface mixer design for many steady rolling maneuvers. To find the mixer that minimized the wing internal loads a constrained parameter optimization algorithm was used. In the current work the optimization problem was formulated as a problem in the calculus of variations. For the cost function the integral of the wing internal loads was chosen. Included in the mathematical formulation are appropriate terminal constraints and differential constraints which are the full linearized lateral-directional equations of motion. For the numerical solution of this optimization problem the conjugate gradient technique, as formulated in Ref.1, was used.

Results
Work up to date shows that the conjugate gradient algorithm exhibits good convergence properties. A typical case is shown in Figure 1. However, the numerical solutions also reveal certain compatibility problems between the aerodynamic and the internal loads models.

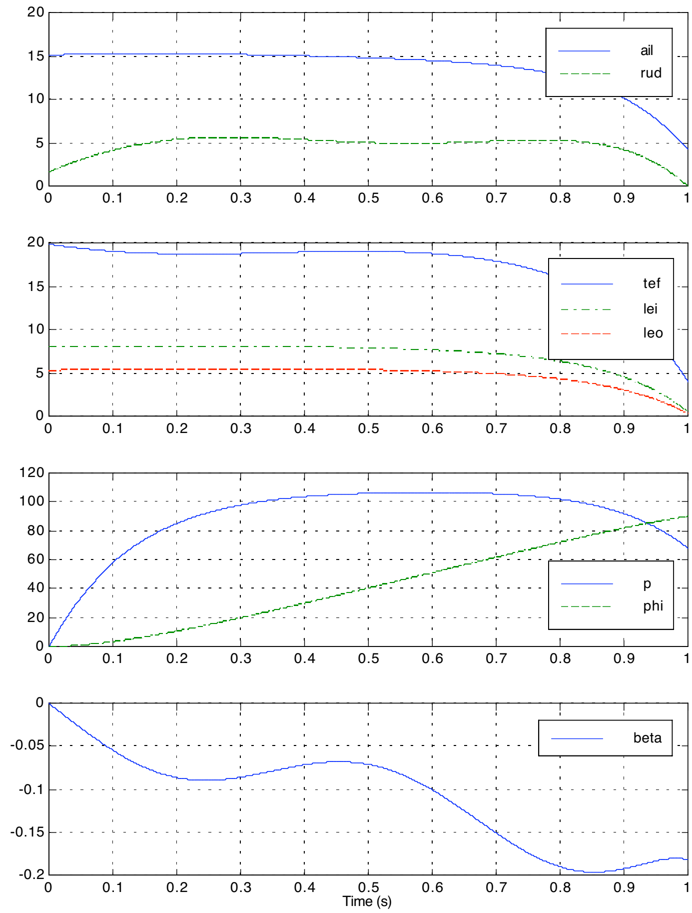


Figure 1. Optimum trajectory for a 90° bank roll maneuver in 1 second at Mach = 0.85 and altitude = 5,000 feet.

Status
To resolve the previously mentioned compatibility issues a series of flight tests of the AAW airplane are planned for spring FY02. These flight tests will provide the definitive aerodynamic and loads models to be used in the closed-loop control law designs. The particular F-18 selected for the AAW program has a modified wing structure and additional control surfaces to allow the utilization of the elastic deformation of the wing for flight control.

References
Lintereur, Louis. J., Optimal Test Trajectories for Calibrating Inertial Systems. CSDL-T-1265, Master of Science Thesis, MIT, June 1996

Contact
Robert Clarke, NASA Dryden, RC, X-3799
Joseph Gera, NASA Dryden, AS&M, X-7917

Autonomous Formation Flight Phase 1 Risk Reduction: Independent Separation Measurement System Relative Position Algorithms

Summary

The primary objectives of the Autonomous Formation Flight (AFF) project have been to quantify the drag reduction effects caused by the wing tip vortex of a lead aircraft on a trailing aircraft flying in close formation and to develop the guidance and control technologies to enable close formation flight. The AFF program is currently using two NASA F/A-18 aircraft, tail numbers 845 (lead) and 847 (trail), for this purpose.

Objective

The principle objectives of the Phase 1 risk reduction portion of the AFF project were to mature a relative position sensor through flight test and to more accurately quantify the vortex flow field for control law development. In order for the wing tip vortex drag reduction effects to be accurately and systematically characterized, Independent Separation Measurement System (ISMS) relative position algorithms were developed to provide real-time three-dimensional relative position information between the two aircraft. The ISMS relative position information is used to drive the Instrument Landing System (ILS) needles on the cockpit Head-Up Display (HUD) of the trail aircraft, providing lateral and vertical guidance to the pilot for manual positioning of the aircraft at pre-defined points inside the leader's wing tip vortex. Longitudinal (nose-to-tail) guidance from the ISMS is relayed to the trail aircraft pilot via mission control room radio calls.

Approach

In brief, the ISMS algorithms use 2 Hz GPS latitude, longitude, altitude, speed-over-ground, course-over-ground, and time for each aircraft as inputs. Additional inputs specific to the trail aircraft are INS true heading, and the time delay caused by processing and transmission of the leader's navigation solution to the trail aircraft via a radio modem link. Outputs of the ISMS are formation reference frame X, Y, and Z relative positions and relative position errors (with respect to the desired position inside the vortex) at a common time at a rate of 10 Hz. The ISMS relative position algorithms use the appropriate coordinate system transformations, data synchronization, and complementary filtering required to arrive at these outputs.

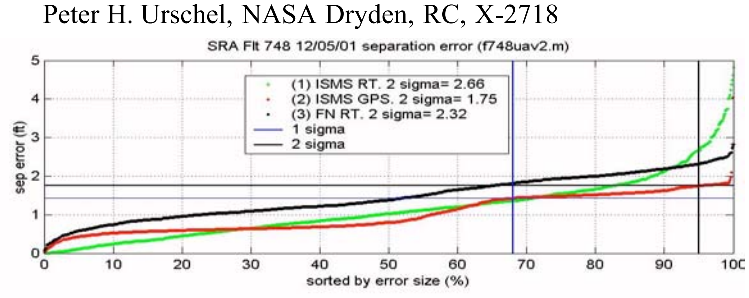
The initial ISMS relative position algorithm design and testing was accomplished using Simulink and Matlab. Testing in Simulink included the use of previous flight data for input to the algorithms, with algorithm output also being compared to previous flight data. Upon completion of the initial design and testing in the Simulink/Matlab environment, the ISMS relative position algorithms were translated into C code for integration with flight hardware.

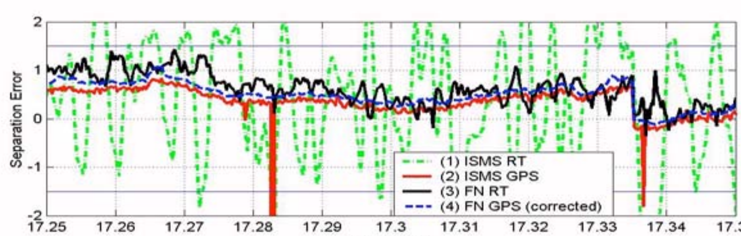
Status

At the time of this writing, the ISMS relative position algorithms have flown successfully and have performed as designed, enabling the AFF Phase 1 Risk Reduction program to collect wing tip vortex drag reduction data. Data analysis is underway. See the ISMS/formation needle (FN) flight data from SRA/AFF flight 748.

Contact:

Peter H. Urschel, NASA Dryden, RC, X-2718





Guidance, Navigation and Control Flight Test Preparations, and Limited Flight Test Results, for the X-43A Research Vehicle

Summary

The Hyper-X program intends to evaluate the operability and performance of an autonomous hypersonic vehicle powered by an airframe integrated scramjet engine. To meet program objectives, the X-43A autonomous research vehicle (RV) is delivered to the required hypersonic test conditions in two stages. The first stage consists of a captive carry flight under the wing of a NASA B-52. During this stage, the RV forms the forebody of a modified Pegasus rocket booster (collectively called the Launch Vehicle, or LV), which is slung under the bomber's starboard wing. The second stage is characterized by a B-52 jettison of the LV which is then dominated by rocket accelerated flight to the engine test condition at nearly 100,000 feet. Once near the required flight condition, the RV separates from the spent booster and assumes autonomous controlled flight for the remainder of the mission.

Objectives

During CY01, the Hyper-X program was at the flight readiness level of maturity. Activities such as flight planning, control room operations development and test data reduction preparedness proceeded smoothly toward an expectant summer launch. Specific items that required project attention included: mission timeline development, checklist generation, emergency procedure development and exercise, control room display development and checkout, and control room training. Duties of the Guidance, Navigation and Control (GNC) group included reviewing flight cards, ensuring adequate anticipation of failure scenarios and definition of emergency procedure response, defining in-flight maneuvers to characterize system health (which included the actuation system built-in-test and navigation performance assessed through B-52 predefined maneuvering, called phasing maneuvers), defining control room displays, and actively participating in control room checkout exercises. Other duties included the preparation of post-flight data reduction tools designed to facilitate rapid flight data analysis efforts.

Flight Results

The Hyper-X research mission was launched on June 2, 2001. The B-52 bomber, carrying the LV beneath its starboard wing, departed from Edwards Air Force Base at midday and headed west towards to Pacific Ocean, as shown in Figure 1. When the B-52 reached the California coastline north of Santa Barbara it turned south towards a predefined racetrack holding pattern. The holding pattern was designed to conduct final system checks of the RV and LV and to configure the B-52 to drop at the intended flight condition. After successfully separating from the B-52 (see Figure 2) and following a preprogrammed short duration free-fall, the Launch Vehicle's rocket motor ignited and began to accelerate the stack to the scramjet test conditions. Shortly after ignition, the Launch Vehicle began a maneuver to arrest the initial sink rate and initiate an ascent to the test conditions at nearly 100,000 feet. Approximately 14 seconds into the mission, however, the Launch Vehicle departed from controlled flight. The Navy's Pt. Mugu Range Safety Office successfully initiated the flight termination system, thus ending the mission after which the debris fell harmlessly within a pre-cleared test range over the Pacific ocean.

Status

The flight preparations that consumed much of the program in the first half of CY01 were successfully validated on June 2. Vehicle and control room operations went smoothly, aided by repeated control room exercises – you can never train too much. Some updates are expected to the GNC control room displays to more efficiently display flight data for real-time monitoring and quick data reduction activities.

The second half of CY01 was dominated by an investigation of the mishap; the investigation ended early in CY02. An official report detailing the investigations findings has been prepared, but as of this writing is not available.

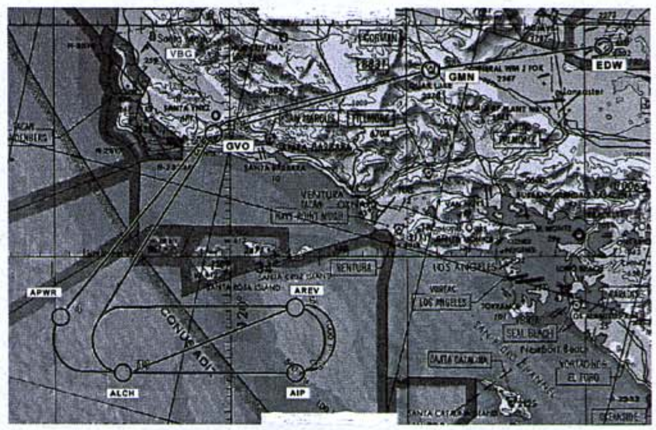


Figure 1 - Captive Carry Flight Track

NASA Dryden Flight Research Center Photo Collection
http://www.dfrc.nasa.gov/gallery/photo/index.html
NASA Photo: EC01-0182-20 Date: June 2, 2001 Photo by: Jim Ross
Moments after release from NASA's B-52 carrier aircraft, the X-43A/Pegasus "stack" is seen before ignition of the Pegasus rocket motor on.

Figure 2 - Hyper-X Launch Vehicle dropping from NASA's B-52 Mothership

Contacts:
•Mark Stephenson, NASA Dryden, RC, X-2583
•Ethan Baumann, NASA Dryden, RC, X-3417
•Mike Richard, NASA Dryden, RC, X-3543

Autonomous Taxi Testbed Vehicle (ATTV)

Summary

The goal of ATTV is to provide two flexible, low-cost, and safe testbeds for evaluating and demonstrating technologies related to the integration of unmanned aerial vehicles (UAV) into ground operations with traditional aircraft. This effort includes developing autonomous ground control and waypoint guidance algorithms and establishing communication between two autonomous vehicles, as well as operating both vehicles simultaneously.

There is a build-up approach to implementing and testing guidance, navigation, and control (GNC) software and hardware in the testbeds. This effort concentrates first on one vehicle only, and then, after extensive testing, will be focused on a second testbed for tandem operation.

Objective

The ATTV program provides fully autonomous testbeds, operating on the Edwards dry lakebed, for future Dyden UAV programs and SBIR/University experiments, capable of working either independently or in tandem.

Approach

The ATTV's are Ford E150 Club Wagon vans outfitted with adaptive driving controls designed for use by drivers with either a limited range of motion or limited strength, which allows a convenient and safe mechanism to interface computer control commands to the vehicles.

A portable Autonomous Taxi Control Computer is located on each vehicle, performing all guidance and control functions. Navigation is accomplished by a Formation Flight Instrumentation System (FFIS) under the guise of the Autonomous Formation Flight Program (AFF). The FFIS computers establish a communication link between the two ATTV's and provide state information for each vehicle.

The primary test objectives are to demonstrate the ability of both a single ATTV and two ATTV's working cooperatively to perform GNC functions to a level sufficient to provide useful comparison to UAVs in a ground operations. The 4-step build-up approach for testing is as follows:

Single Vehicle, Open-Loop Control
Single Vehicle, Closed-Loop Control
Dual Vehicle, Open-Loop Control
Dual Vehicle, Closed-Loop Control

Status & Future Work

Early tasks have focused on selecting and integrating hardware and designing an interface panel to communicate with the vehicle throttle and steering controllers. There has been some initial work on GNC software, which will continue after upcoming vehicle system identification testing, eventually leading to the 4-step series of evaluations.

Possible future experiments will include adding collision avoidance sensors and implementing formation taxi control laws.

Contacts:

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Shane Petersen, DFRC, RF, (661) 276-5664

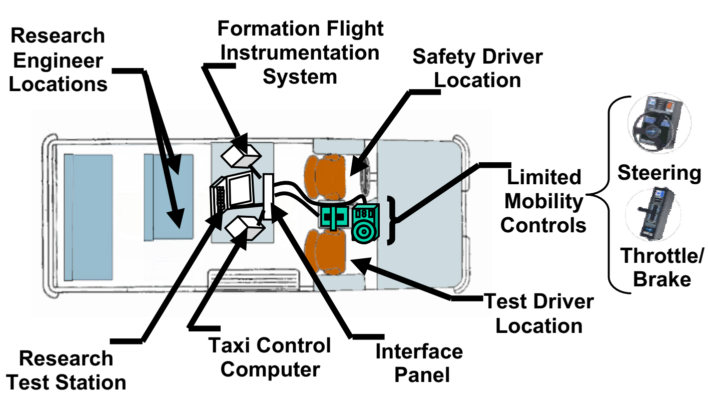


Figure 1: Autonomous Taxi Testbed Internal (Concept)

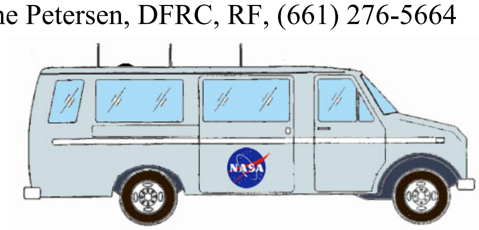
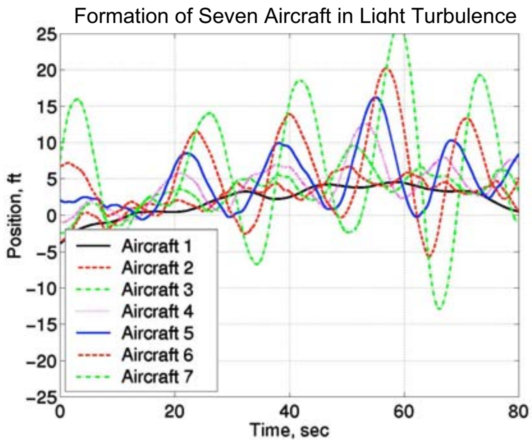


Figure 2: Autonomous Taxi Testbed External (Concept)

String Stability Analysis of an Autonomous Formation Flight System



Summary

String stability (sometimes called the slinky-effect) describes how position errors propagate from one vehicle to another in a cascade system. Typically, any position changes to the first aircraft are reacted to by the second aircraft with slight overshoot. Each aircraft overshoots the motion of the previous aircraft. This can cause large motion of the last aircraft in the string. The AFF phase-0 system magnifies position errors caused by the first aircraft and is thus string unstable. For two-ship formations, this is not a problem because the magnification is not time dependent. Since very little work has been published about the possible use of a string unstable system, a study of the AFF system for larger formations was conducted.

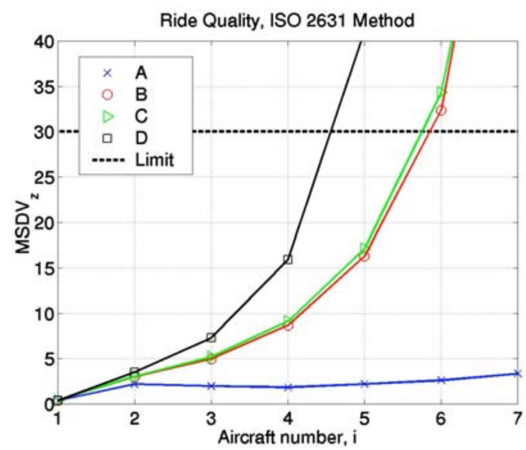
Objective

The principle objectives of the study were to determine analysis methods for use with future formation flight control systems.

Approach

A linear technique was found by expanding an equation used to describe the string stability of cars on the automated highway system. Worst-case input was used to ensure that the technique would conservatively predict the motion of the i^{th} aircraft in formation. This technique can be used in the design of a formation flight system. A unique aspect of this method is that string unstable systems are allowed under certain conditions with limited-size formations.

Two remaining analysis methods both used the nonlinear simulation. Each aircraft's time history was recorded and played back for the next aircraft to follow. Each aircraft in formation was then checked for relative-position tracking-performance and ride quality. Four different controller gainsets were used in this analysis.



The A-gains gave lower Motion Sickness Dose Values (MSDV) than the other three gainsets. This is because the A-gains cause the system to react at a frequency that is not conducive to motion sickness.

Status

Detailed results of this work will be presented at the AIAA Guidance, Navigation, and Control conference in Monterey, CA on August 6. A paper entitled "String Stability of a Linear Formation Flight Control System" will follow.

Contact:

Michael Allen, NASA Dryden, RC, X-2784

Ground and Flight Tests of Sub-Scale Inflatable Wings

Summary

Aircraft with inflatable wings are potentially attractive for a number of aerospace applications including planetary exploration and RLVs. Recent government R&D contracts have made a small number of inflatable wings available to researchers at the NASA Dryden Flight Research Center. The inflatable wings were integrated into the design of two small-scale (15–25 lb), instrumented, research aircraft configurations: a pusher-powered conventional configuration (I-2000), and an unpowered winged lifting-body configuration. Conventional ground and flight test techniques were applied to the research aircraft to gain experience with and an understanding of the structural, aerodynamic, and operational characteristics of vehicles with state-of-the-art inflatable wings.

Objectives

Apply conventional ground and flight test techniques to the research aircraft to gain experience with and an understanding of the structural, aerodynamic, and operational characteristics of vehicles with state-of-the art inflatable wings. In addition, demonstrate in-flight, dynamic (< 1 sec) deployment of the inflatable wings.

Approach

Sub-scale testing of inflatable structures is attractive because the vehicle fabrication costs, personnel costs, and test range costs are all reduced with smaller vehicles.

Prior to flight with the inflatable wing, each research aircraft was flown with a rigid wing of similar geometry to the inflated wing. Flight operations with the rigid wing were used to test the onboard systems and to practice the required flight test maneuvers prior to commitment to flight with the inflatable wing. Research maneuvers flown with both the rigid wing and the inflatable wing allowed comparison of the trim, aerodynamic performance, and stability and control characteristics of the aircraft in both configurations.

Flight Results

The majority of the flight data collected to date has been with the conventional, pusher-prop configuration (I-2000). Comparisons of the vehicle in both a rigid and inflatable configurations are possible. Figure 2 shows a comparison of the vehicle normal-force coefficient versus angle-of-attack for three configurations; rigid, pre-inflated, and in-flight inflated. For the limited angle-of-attack range spanned, these data show that the gross lift-generating capability of the inflatable wing configurations is not measurably different from the rigid-wing analog. Takeoff and landing speeds for the two configurations were also similar. Roll trim of the two configurations, however, was measurably different. The inflatable wing set initially flown was (unintentionally) fabricated with a small amount of twist in each wing panel. The resulting rolling moment required approximately 10 of differential horizontal tail deflection to trim out. A small trim tab was affixed to the left wing panel to correct the roll trim.

Status

Flight tests have been completed on the I-2000 using the pusher-prop configuration with both a rigid, and a pre-inflated wing. In-flight deployment of the wing was demonstrated, and measured flight data were used to illustrate the dynamic characteristics during wing inflation and transition to controlled lifting flight.

Two flights of the winged lifting-body have been accomplished with the rigid wing. Ground tests of the wing retention and deployment system have been successfully completed. Drop tests of the lifting body with an in-flight deployment of the wing are anticipated within this calendar year.

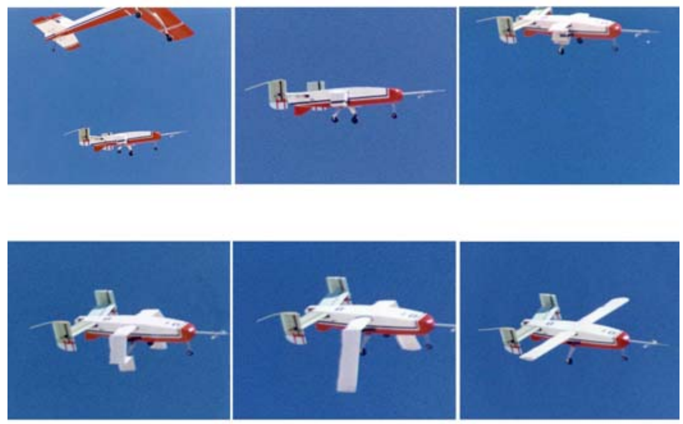


Figure 1 - First In-flight Deployment of the Inflatable Wing on the I-2000

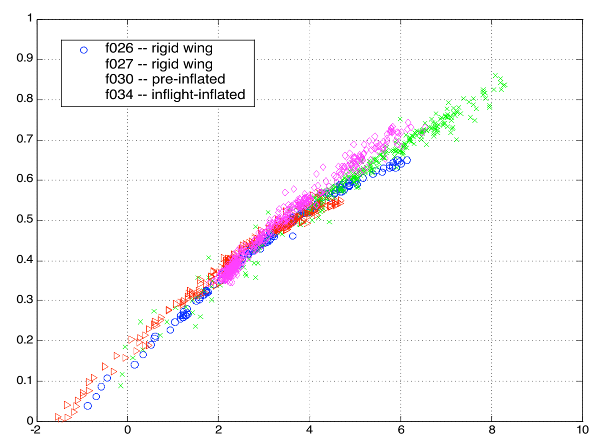


Figure 2 - Comparison of normal force coefficient for a rigid and inflatable wing

Contacts:

•Joe Pahle, NASA Dryden, RC, X-3185
•Jim Murray, NASA Dryden, RA, X-2629

X-38 Boost-Launched Aerodynamic System Testbed (BLAST)

Summary

The goal of the X-38 program is to demonstrate the technology required for the International Space Station's emergency Crew Return Vehicle (CRV). Five prototype vehicles, based on the X-24A lifting body, were developed to demonstrate the CRV concept. One of the X-38 prototypes is designed for space flight testing. This vehicle will be released from the Space Shuttle payload bay and reenter the earth's atmosphere, covering a Mach number range from M=25.0 through subsonic on its maiden flight. Prior to this flight, the X-38 program has no plans to obtain flight data above Mach 0.80. Historically, wind tunnels have had difficulties in accurately predicting the transonic aerodynamics for lifting bodies. For this reason, transonic flight test data are desired to minimize the aerodynamic uncertainties for the X-38 space flight test vehicle. The BLAST program was formed at DFRC to study potential methods for obtaining transonic aerodynamic data for the X-38 program.

Objective

The BLAST Program is a conceptual design study investigating the feasibility of using a sub-scale vehicle to obtain transonic flight data for the X-38 program.

Approach

The BLAST program was started by investigating the possibility of getting a small scale, lightweight model of the X-38 to transonic speeds by launching it from a rocket sled. In simulation runs the high speed, low altitude launch condition resulted in excessively high dynamic pressure and the associated drag caused the vehicle to slow down rapidly. Therefore, the BLAST concept was adapted to use a B-52 as the launch platform. Initial studies with a 25% scale, 3000 lb. unpowered model showed that supersonic speeds could not be achieved when launching from the maximum altitude and mach number obtainable with the B-52. Balloon launches from higher altitudes were considered, but the concept was abandoned due to inability to predict exact launch locations. A liquid rocket propulsion system was added to the model and simulated B-52 launches with this configuration showed promising results. After launch, the vehicle would first climb to about 60,000 feet, then accelerate and glide back on the space flight vehicle trajectory. Launching from an F-15 was proposed to reduce the fuel required for the vehicle to climb. Simulations showed Mach numbers as high as 2.0 where achievable with this configuration. However, the F-15 launch option became infeasible once the model scale was increased to 50% due to refined program requirements. With the larger scale model (50%), focus was returned to unpowered launches using a tailcone to reduce drag. Simulation studies were preformed to study the effect of varying model weight and flight profile. Low supersonic speeds (1.05 – 1.10) were achieved with models weighing greater than 15,000 lbs. To increase the maximum Mach number, a propulsion system was once again added to the vehicle. This time, solid rockets were chosen over the previous liquid rocket engine to reduce propulsion system complexity. Adding internal solid rocket propulsion increased the maximum Mach number to about 1.35. However, with solid rockets, if a problem was encountered during the rocket burn segment of flight, terminating the rocket burn would likely result in loss of the vehicle. Integrating the solid rocket with the tailcone was proposed to as a potential solution to this problem, but this option has not been thoroughly investigated.

Status & Future Work

The BLAST program has been indefinitely delayed. Simulation studies have isolated potential configurations that could be used to obtain the desired transonic data. Future work would focus on refining requirements and applying constraints to narrow down vehicle configuration options.

Figure 1: X-38 V201 (Artist Rendition)

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Flight Controls Laboratory

Summary

The Flight Controls Branch has developed and is improving a small laboratory primarily for testing control law theories. Single input, multiple output hardware in the loop (HIL) control of an unstable system using a desktop PC is implemented to learn, teach, or experiment.

Objective

A flight controls laboratory has been developed to provide HIL control law experimentation such that engineers gain experience with a design before flight control applications are attempted.

Justification

Many new hires have not designed control laws for 'real world' applications. Complementarily, theoretical control laws with promise for use as flight controls laws may have not been implemented in 'real world' HIL control systems. The HIL laboratory provides a proving ground for junior engineers' skills as well as new control law theories.

Benefits

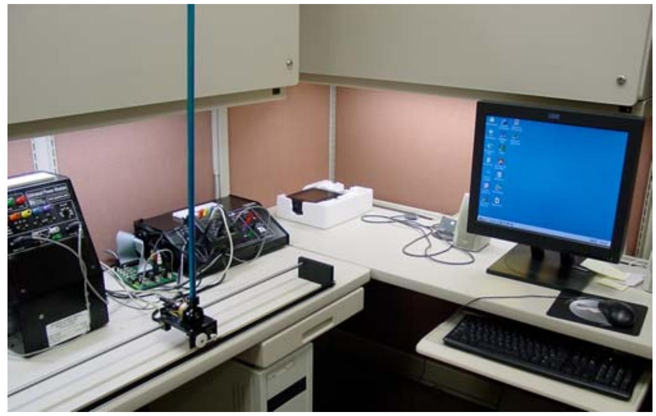
- preliminary evaluation of control theories for use in flight
- gain experience in control law design; decrease the learning curve for new hires

Approach

Since Matlab and Simulink are the primary development tools for flight controls, a desktop PC is used to provide the user with an environment to design and test control laws with familiar tools. A HIL system was integrated with the PC via a data acquisition card in an expansion slot. The hardware currently used is an inverted pendulum with analog or digital angle and position sensors and an electric motor for position control. To allow more complex control systems, further expansion of hardware configurations is underway.

Results

In the previous year, one new hire designed and tested multiple controllers for the inverted pendulum, including LQR and LQR servo theories. Also, a co-op student helped in using Parameter Identification (PID) theories to characterize the inverted pendulum. This was undertaken to increase the fidelity of the linear model such that a classical controller design (which would be less robust) could be implemented.



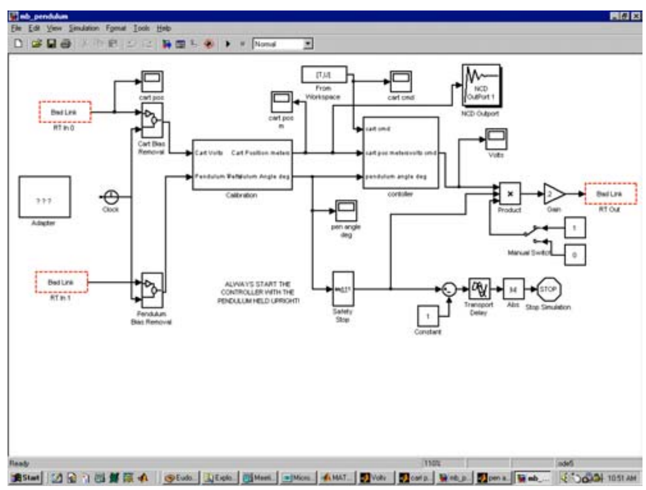
Inverted Pendulum and PC system

Status

A new desktop PC has been acquired and integrated with the HIL. Thus, control step times can be reduced for quicker control.

New experiments with multiple inputs and multiple outputs are being integrated for more challenging control design.

Testing of Neural Network control theories in the laboratory is planned. This could help in testing multiple theories as well as gaining experience in designing these control laws.



Screenshot: Simulink control law design software

Contact:

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Parameter Estimation Analysis of X-38 Lifting Body Flight Data

Summary

The NASA Johnson Space Center is currently developin the technology to design and build a lifeboat for the spa station. The program, designated the X-38, is currently using an experimental vehicle, designated V131R, to investigate the transition of controlled aircraft-like flight through the deployment of a guided parafoil landing system to touchdown. In July of 2001, V131R (fig. 1) w launched from a B-52 carrier aircraft at 35000feet, .6 Mac Approximately 30s after launch a drogue chute initiated the deployment of the parafoil system, allowing for a successful landing at a pre-designated landing zone on t ground. During the 30 second free-flight (see fig. 2) portion of the flight Programmed Test Inputs (PTIs) we performed.

Figure 1: V131r during free flight #2

Objective

The objective of the PTI's were : 1) to investigate a potential yawing moment asymmetry triggered by aileron deflection observed in some wind tunnel testing and 2) to extract aerodynamic derivatives from flight for comparison with wind tunnel prediction, including a yawing moment curve which is non-linear for varying sideslip (fig. 3).

Justification

The accuracy of the wind tunnel estimates and the existence of the aileron triggered yawing moment asymmetry will influence the flight control design requirements for the next research vehicle, designated V201, which will be launched from the space shuttle.

Approach

To investigate objective #1 a PTI was designed consisting of two lateral steps with opposite signs. About 4s seconds was left at the end of each step to observe trim changes in the vehicle which might imply that the yawing moment asymmetry was triggered. To extract aerodynamic derivatives, objective #2, a rudder doublet was inserted in the command path after the lateral steps. This doublet was combined with the second lateral step in parameter estimation of the aerodynamic derivatives.

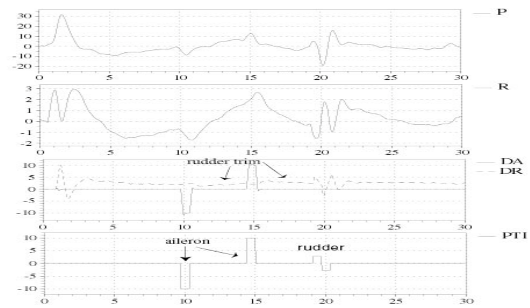

Figure 2: V131r free flight time history

Results

Observation of the rudder time history (fig. 2) after each lateral step reveals only small rudder trim changes. This implies that the aileron deflection did not trigger the yawing moment asymmetry observed in some of the wind tunnel data.

Parameter estimation analysis resulted in a set of aerodynamic derivatives which produced good motion matching when compared to the flight data and reasonable uncertainty estimates. When compared to the wind tunnel data, most of the key derivatives were estimated to be within the wind tunnel predicted uncertainty. However, there were two notable exceptions. The rudder derivatives were approximately 30% more effective than predicted, and the the yawing moment vs sideslip curve was more linear and more stable than predicted.

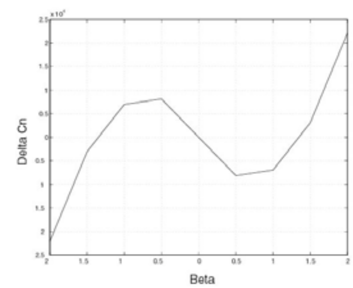

Figure 3: Predicted yawing moment vs. sideslip

Status

The results indicated above are based on one flight, one data point. More data is necessary to confirm trends. For the next flight of V131R the launch altitude will be approximately 45000 feet, resulting in about 50s of free-flight time. Four lateral/directional doublets and one longitudinal doublet are being planned for parameter estimation purposes.

Contact: Timothy H. Cox / Oscar Murillo RC, X2126

Sidestick Controller Evaluation

Summary

Two sidestick controllers were designed and built by Dynamic Controls, Inc. to test ideas for fighter aircraft control. Unlike previous sidesticks that offer only a limited range of motion, these sticks have a large displacement. The sticks have motion characteristics unlike other sticks – one translates for roll control while the other rotates, and both translate for pitch. A sliding elbow cup is provided to reduce arm fatigue by involving the entire arm in the motion. The sticks are undergoing testing in a fixed base simulator to determine how handling qualities of a fighter aircraft are effected relative to the standard centerstick.

Objective

While the wide range of motion may alleviate some of the problems present in current small displacement sidesticks, it is not known how pilots will adapt to the unusual motions. The sticks are being tested in a simulator running F-18 software. By obtaining feedback from pilots flying fighter maneuvers (air to air tracking, gross acquisition, and landing from an offset approach) the merits and drawbacks of the stick's design can be evaluated. Data from the simulator will shed some light on any differences in piloting technique that arise from use of the sidesticks.

Justification

This experiment is a follow-on to testing performed several years ago by the Air Force. The data collected will expand on the previous tests and should offer new insight into the comparison. Specifically, the previous tests only compared the sidesticks with an F-16 sidestick, whereas a more common centerstick forms the basis of this test. In addition to the pilot feedback, simulator data (such as stick displacement time histories) will give additional information as to how pilots adapt to unusual controllers.

Benefits

This experiment will expand on the database of aircraft control sticks. In particular, the relationship between pilot workload and opinion will be analyzed for the different motion types tested. This study will provide future aircraft designers with possible solutions to some of the difficulties currently encountered in the pilot-vehicle interface. It will also provide suggestions for future research into this area.

Approach

The sidesticks were installed in a six degree of freedom nonlinear simulation of an F-18. Test pilots are given three maneuvers to fly (air to air tracking, gross acquisition, and landing from an offset approach) with each of the three sticks. Cooper Harper ratings, pilot comments, and time histories are recorded for analysis. In addition to the overall ranking of the sticks, factors influencing the pilots' opinions will be explored.

Results

While testing is currently under way, results from the initial pilots have yielded several interesting conclusions. For example, two of the four pilots expressed preference for the centerstick over both sidesticks and ranked Sidestick #2 (translating roll motion) last. The other two pilots actually preferred Sidestick #2 over the others, and ranked the centerstick second. Pilot workload may play into effect, as the pilots who were very aggressive tended to have difficulties using the sidesticks.

A second factor warranting further examination is piloting style. With the centerstick, most pilots made roll and pitch commands simultaneously. On Sidestick #2, the pilots tended to make pulses in either pitch or roll, commanding each axis individually. The significance of this and its relation to pilot ratings is under investigation.

Status

Pilot evaluations are currently being conducted. Data analysis tools and procedures have been developed, and data collected in the first runs has been processed. More pilots are currently scheduled to perform evaluations in the near future, bringing the total to six or seven. Upon collection and reduction of the data, trends in both pilot opinion and time histories will be examined. The project should be completed by early January.

Contact

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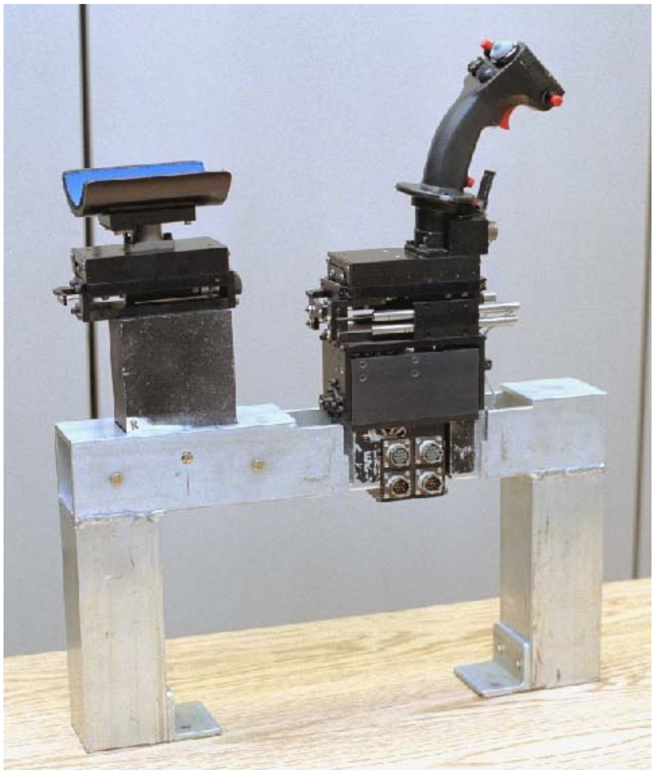
Jann Mayer, NASA Dryden, RC, x5696



C-17 REFLCS Objective

The intent of the C-17 REsearch FLight Computing System (REFLCS) is to demonstrate Intelligent Flight Systems (IFS) technologies in a real flight environment. One application of IFS technology to be demonstrated in flight is neural-net based Intelligent Flight Control System (IFCS) software. In a simulation environment, IFCS software has already demonstrated the ability to automatically compensate for degraded vehicle characteristics that may result from damage, control surface failures, or mis-predicted aerodynamics. The benefits range from the ability to land an airliner with failed hydraulic systems to return of a battle-damaged combat aircraft to increased robustness for a re-entry vehicle. Additionally, IFCS is a prerequisite for NASA's futuristic "morphing aircraft." Another application of IFS technology to be fight tested is an Integrated Vehicle Health Management (IVHM) system which will enable vehicle maintenance to be 'need' based as opposed to 'schedule' based. Also, the enhanced IVHM fault detection and reporting capabilities can be used to provide the capability to host IVHM elements within the IFCS flight controllers.

Approach

The REFLCS will be composed of a quad redundant set of Research Flight Control Computers (RFCC's) as shown in figure 1.

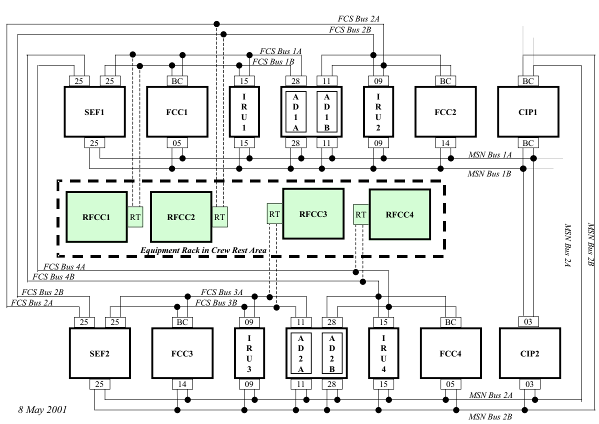


Figure 1: C-17 REFLCS Avionics Architecture

In order to achieve the full complement of IFS objectives, the capability of the C-17 REFLCS will be enhanced through incremental builds.

Build 1 of the REFLCS will consist of replication C-17 control laws (CLAWS) in order to demonstrate that the REFLCS 'tool-set' will function in a safe and effective manner to flight test IFS technologies. Build 1 will function only in a limited (up and away) Class B envelope. Build 2 of the REFLCS will house Gen. 2.0 IFCS CLAWS also in a Class B envelope with 'simulated' fault insertion capability. Build 3 of the RELFCS will be a full envelope (CLASS A) qualified system and will incorporate engine control and initial IVHM fault monitoring and detection. In order to directly compare the benefits of IFS technology with a 'conventional' flight control system, the REFLCS will provide the capability to switch between research CLAWS and the 'baseline' C-17 replication CLAWS while REFLCS is engaged in flight as shown in figure 2 below.

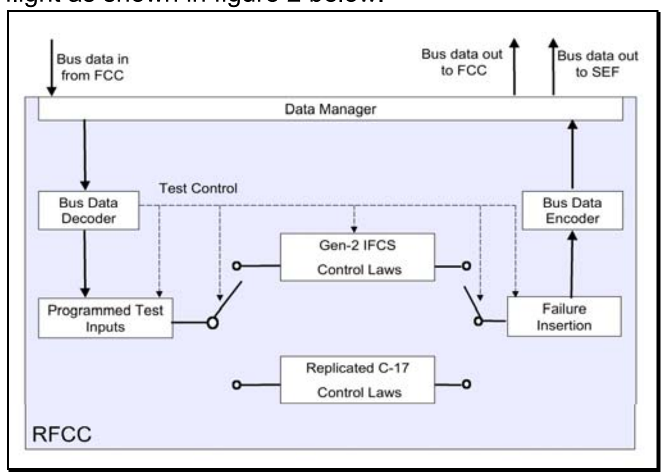


Figure 2: RFCC Software Architecture

Status & Future Work

Currently the project is focused on the development of REFLCS Build 1. Subsystem and system level integration and test is currently set for early 2003. Flight test of REFLCS Build 1 is currently scheduled to begin near the end of FY 2003. Also, the Build 2.0 REFLCS System Requirements Specification (SRS) is currently being drafted to support Gen. 2.0 CLAW flight research. The REFLCS Build 2 / Gen. 2.0 flight tests are currently scheduled to occur before the end of FY 2004. Concurrently, an effort is under way to build up an in-house Hardware-In-the-Loop-Simulation (HILS) of the C-17 REFLCS. The C-17 REFLCS HILS will support development, integration and test of future REFLCS builds to support IFS. The HILS development is currently planned to be completed by the end of FY 2003. Also, an effort to develop an SRS for Build 3 of the REFLCS will begin near the middle of FY 2003. REFLCS Build 3 is currently scheduled to fly before the end of FY 2005.

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NASA F-15B #837 Intelligent Flight Control Systems

Summary

The NASA F-15 #837 Intelligent Flight Control System (IFCS) project has re-initiated an effort to flight demonstrate neural network based flight control technologies. These revolutionary new concepts will be used to optimize aircraft performance in both normal and failed conditions. The NASA 837 aircraft will be augmented with a second generation Airborne Research Test System (ARTS II) computer processor to provide for the increased computational requirements of these advanced flight control technologies. This is a joint effort teaming NASA with the Institute for Scientific Research (ISR) and Boeing Phantom Works.

Objective

The IFCS program objective is to utilize neural network technologies to efficiently identify aircraft stability and control characteristics and utilize this information to optimize aircraft performance in both normal and failure conditions. The ARTS II processor is the new enabling technology that provides the computer power to achieve this objective.

Justification

Existing flight qualified computers are far behind the state of the art in memory and computational capabilities. To demonstrate advanced control systems with neural networks requires increased processor speed and memory capacity for data processing and storage. Additionally, the ARTS computer will allow engineers to rapid prototype control concepts for use within a limited flight envelope with non-flight critical software.

Approach

The ARTS II system will be implemented as a single-string element that communicates with the flight control processor through a MIL-STD 1553B Multiplex data bus. Computationally intensive portions of the flight control task can be executed in the ARTS II computer. The redundant flight control computer provides safety-of-flight checks on the computations prior to using the data provided by the ARTS II.

The initial IFCS Advanced Concept Program (ACP) originated in the 1990's. The original objectives of the IFCS ACP consisted of three development phases:

Phase I: Pre-Trained Neural Network (PTNN).
Phase II: On-Line Neural Network (OLNN).
Phase III: Neural Network Flight Controller.

The Phase I PTNN software was developed and implemented in the existing research computer. Phase III utilized the Phase I PTNN to provide real-time aircraft stability and control derivatives to a Stochastic Optimal Feedforward and Feeback Technique (SOFFT) adaptive controller developed by NASA Langley. This combined Phase I/III system was successfully flight-demonstrated on the NASA 837 test aircraft in 1999. The current effort will implement a Phase II OLNN in the new ARTS II computer. The Phase II/III software will be flight-demonstrated and evaluated on NASA 837 in 2003.

NASA F-15 # 837

Results

As of August 14, 2002 the ARTS II has been developed by ISR and environmentally tested by BPW. Test pilots and engineers from NASA Dryden Flight Research Center (DFRC) performed Hardware-In-Loop Simulation (HILS) testing at the BPW St Louis MO. facility. NASA 837 successfully returned to flight in July 2002 after a two-year hiatus. Initial flight validation of the ARTS II HW/SW integration with NASA 837 will begin in the fall of 2002.

Reference

Intelligent Flight Control: Advanced Concept Program, Final Report, Boeing-STL 99P0040 (NAS2-14181)

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MINIATURE FLIGHT CONTROL SYSTEM

SUMMARY

The uINS system developed by Microbics for Dryden is a small footprint flight control system with instrumentation and telemetry capabilities. The system consists of three units, uINS with DSP and GPS, Magnetometer assembly with barometer and host micro controller (Motorola 68332). The instrumentation located on the host controller has 16 digital and analog channels and 4 pulse width modulation ports. A wireless data transceiver is the telemetry system.

OBJECTIVE

Dryden wants the ability to participate in the unmanned aerial vehicles (UAV) arena and the Miniature Fight Control system (MFCS) will be a beginning for such participation. The investigation of using auto-coding tools such as Simulink will be the main focus. Instrumentation and telemetry capabilities will also be addressed.

JUSTIFICATION

Because of its small size and intended target platforms little resources would be required for development. Development of techniques for auto-coding will lessen the burden of flight control software design.

APPROACH

Installation of the MFCS into Dryden Utility model plane for uINS and telemetry system checkout will be the first order of business. Adding the host microcontroller and running Simulink code will be the next.

STATUS

A calibration of the rate and pressure sensors has been performed. Currently a data terminal is being worked that will house a smart display and the receiver for the telemetry system, this task will remove the laptop, which is now used to log data.

CONTACTS:

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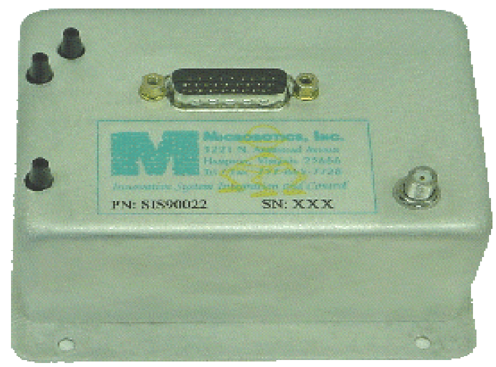
Joe Pahle, NASA Dryden, RC, X-3185
Jim Murray, NASA Dryden, RA, X-2629

uINS with DSP and GPS
2 _ X 2 _ X 1 _ inches



Magnetometer assembly with barometer
2 _ X 2 _ X 1 _ inches

DFRC UTILITY UAV
Wing span 10 Feet
Weight 29 pounds
80cc twin cylinder gas power engine

X-37 Approach and Landing Test Vehicle (ALTV)

Summary

The X-37 ALT system is an advanced technology flight demonstrator vehicle and companion Flight Operations Control Center (FOCC), developed and flown as a high technology testbed system under a cooperative agreement between Boeing and NASA. The X-37 ALT vehicle, containing embedded technologies and experiments, uses the configuration, database, and software derived from the Air Force Space Maneuver Vehicle (SMV)/X-40A program.

Objective

The X-37 research is expected to result in a low cost, reusable launch vehicle to test future launch technologies in both the orbital and re-entry phases of flight, enabling routine, safe, low-cost access to space with high reliability, and fast turnaround – similar to commercial airline operations. Toward these objectives the X-37 is required to fly autonomously with a minimum support staff.

Approach

The X-37 ALTV is being fabricated, assembled and tested by Boeing with Marshall and Dryden NASA participation. The flight test program follows a progressive capability expansion, starting with tests of the X-37 vehicle being towed along a runway by a ground vehicle. Free taxi tests are followed by atmospheric final approach and landing tests, using a B-52H aircraft to carry the X-37 aloft and release it from high altitude. The vehicle will be suspended via pylon under the wing of the B-52H. During flight, operators at the B-52 LPO station and inside the FOCC will monitor systems and assure vehicle readiness for a safe separation. The vehicle lands autonomously and does not have a remote piloting capability.

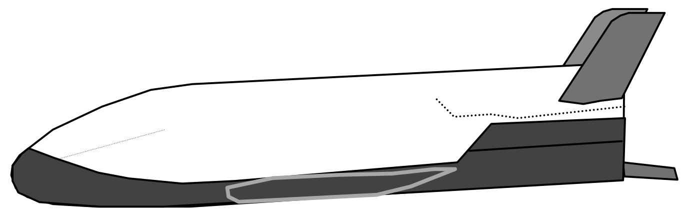


Figure 1: X-37 ALTV

X-37 key milestones include:

- Simulator and Major Assembly Tests
- Vehicle Integration and Test
- Towed Vehicle test
- Captive Carry with B-52H
- B-52H High Altitude Launch Flight Test

Status & Future Work

As of August 2002, the X-37 is progressing toward resolution of B-52 separation issues, analyzed and shared with Marshall and Boeing counterparts. This discovery as well as a systems requirements verification has resulted in additional requirements and clarification/ resolution of existing requirements. The vehicle itself is being assembled at Palmdale facilities. Originally intended to be identical to the space counterpart, the ALTV has become somewhat unique.

Possible future experiments will include 5 flights.

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X-43A Flight Systems 1st Flight Overview

Summary

The first flight attempt of the X-43A Research Vehicle (RV) occurred on June 2, 2001. Although the flight was unsuccessful due to the failed boost attempt, several secondary flight systems test objectives were met. These included successful ground processing operations, smooth flight and range operations, and nominal RV systems performance through flight termination. This report summarizes the first flight systems activities by detailing the systematic approach to vehicle integration and flight operations taken by the X-43A flight systems team.

Objectives

The primary objective of the X-43A first flight was to validate the scramjet engine performance at the desired test condition. This objective was lost once the booster departed from controlled flight. However, several secondary objectives were accomplished. These included the execution of the research vehicle flight systems pre-flight, nominal performance of the research vehicle monitor and control station onboard the B-52 carrier aircraft, successful control room operations, and nominal performance of the RV flight systems through flight termination.

Approach

The systems team designed the RV pre-flight test procedure to verify performance of the vehicle systems prior to start of fueling operations and subsequent flight operations. The procedure was developed and used during execution of all prior validation testing, including fuel system blow-downs and the captive-carry flight. The procedure verified nominal performance of the major flight systems by conducting Built-In-Tests (BITs) on the vehicle control surfaces and fuel system motorized control valves. Additionally, all fuel system solenoid valves, valve inhibits, and emergency purge sequences were verified. The operation of the flight computer, Inertial Navigation System (INS), and telemetry interfaces were also checked.

As part of the systems pre-flight, functions at the B-52 research vehicle monitoring station were also verified. Critical control functions at the monitor station verified during pre-flight included control of the fuel system inhibits, manual vents, emergency vent and purge, actuator BIT, and internal battery select. The station operator, a member of the flight systems team, was also responsible for monitoring of critical RV safety data, including fire detect sensors and system pressures. In addition, the station operator monitored the status of the hooks in the B-52 pylon and pylon adapter, and removed the safety pins in the pylon adapter hooks prior to drop initiation. Just prior to release, the operator removed safety inhibits on the fuel system solenoid valves, commanded the RV adapter on-board cameras to on, switched the vehicle to internal power, and moded the flight computer to the pure-inertial navigation mode. All of the monitor station functions performed nominally during 1st flight operations. The activation of the camera system, proper navigation moding of the research vehicle flight computer, and successful transfer of system power from the B-52 to the internal battery demonstrated this.

After drop of the booster stack from the B-52, power for the research vehicle flight systems was provided solely from the internal battery. All vehicle systems remained active and telemetry lock was achieved on both vehicle transmitters. Following termination of the booster, separation between the booster stack and the RV occurred. This was indicated both in telemetry and through video acquired from the adapter cameras. The vehicle systems continued to operate nominally until splashdown in the ocean.

Results

The X-43A flight systems performance during the extreme conditions encountered during the first flight demonstrate the completeness of the X-43A verification and validation program, as well as the overall effectiveness of the flight systems qualification program. The research vehicle flight systems data collected, in concert with the booster data, continue to provide insights into possible causes of the boost mishap.

X43A 1st Flight Take-Off

X43A Stack Drop

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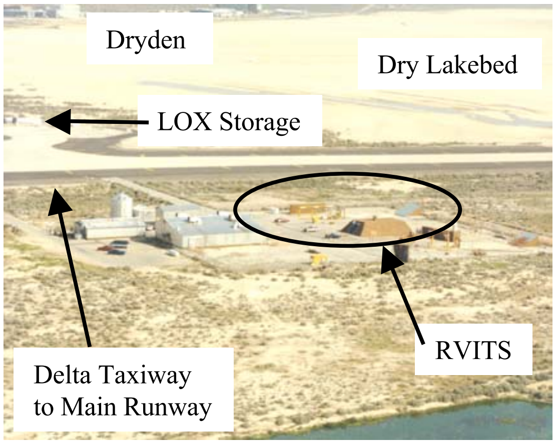
The Rocket Vehicle Integration Test Stand (RVITS)

Summary

NASA Dryden, in conjunction with the AFFTC and AFRL, is establishing a Rocket Vehicle Integration Test Stand (RVITS) at the site of the historic X-15 Rocket Engine Test Facility.



RVITS Area – Aerial View

Objective

Edwards Air Force Base (EAFB) will reestablish the capability of supporting preflight operations for Space Launch Initiative (SLI) programs and related technologies. Potential vehicles and projects requiring this facility in the near future may include X-37, RBCC, TBCC, X-43B, and PDE. This facility will be used to:

- Provide fully integrated vehicle validation
- Trouble-shoot after propulsion system anomalies and modifications
- Reduce technical and operational risks
- Hot-fire installed engines in a controlled environment that is compatible with several types of propellant requirements.

Justification

NASA is undergoing an agency-wide push to develop new and advanced Access to Space technologies. Dryden has a key role in the flight development of these technologies. RVITS will provide a critical ground test facility at Edwards for supporting flight operations, and conduct integrated vehicle/propulsion system check-out of Access to Space vehicles.

A continuing flight program requires these ground testing capabilities, not only initially, but typically throughout the flight program. Having this capability helps to insure mitigation of risks, which lead to an increase in flight safety and increased probability of reaching mission success.

Approach

Once the need was recognized for a ground testing infrastructure such as this at EAFB, a survey was taken to determine the best approach to reestablish this capability. The ability to use the existing infrastructure of the X-15 test site proved advantageous in reducing the cost and schedule in comparison with rebuilding these capabilities. The optimal location of the site was also a major advantage, being located directly off of the delta taxiway for easy access, but also remotely located enough for safety considerations.

Since this site had not been used for decades, thousands of pounds of debris and weeds had to be removed in a thorough cleanup effort. A feasibility analysis followed soon after, assessing the structural integrity of the existing infrastructure for future use. This analysis then showed the team which structures could be used as well as what modifications would be required.

Design and detail drawings for rehabilitation were initiated focusing on general requirements of a typical rocket-powered vehicle utilizing the stand. These drawings also included the modifications that would be required as determined from the feasibility analysis. This design effort was lead by Dryden and tasked to AFRL, and all drawings are now released to AFRL's configuration control system. Upon facility activation, these baseline requirements may be supplemented with enhancements as necessary. Construction will be initiated at the site when a specific project expresses a need for the facility.

RVITS Test Stand Before Rehabilitation

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07/17/02

Autonomous Formation Flight Performance Results

Summary
The Autonomous Formation Flight (AFF) program has demonstrated significant vehicle performance improvements during formation flight using two highly-instrumented F/A-18 aircraft. A matrix of trailing positions were flown at two flight conditions to map-out the performance benefits. Positioning cues were provided to the pilot on a Heads-Up Display. The changes in lift and drag coefficient were evaluated for each maneuver by comparing data while in the vortex to baseline values as shown. Contour plots of drag reduction were generated as a function of lateral and vertical position to help summarize the results and determine the optimal position.

Objective
Quantify the reduction in aircraft drag and engine fuel consumption during formation flight as a function of trailing position.

Results
Maximum drag reductions of over 20% were calculated at 0.56 Mach number, 25,000 feet altitude for longitudinal separations of X=3.0 and 4.4 wingspan aft (nose-to-nose). Total fuel flow reductions were demonstrated up to 18%. Other longitudinal stations (X=2.0 and 6.6) showed similar results. Although simple theory predicts the trailing aircraft can influence performance parameters for the lead aircraft at close separation, no improvements in drag or fuel flow were demonstrated for the lead aircraft at any of the conditions tested during this program.

The optimum drag reduction region was found to be at a vertical position range of –10 to 0% and a lateral position range of –20 to –10% wingspan. All data show more sensitivity (steeper gradients) as the trailing aircraft moves inboard of the lead (increasing the wing-tip overlap) compared to the outboard positions. In fact, significant drag *increases* were calculated at some high wing overlap positions, verifying the importance of proper station-keeping to obtain the best results.

Induced drag results compare favorably with a simple horseshoe vortex prediction model. The induced drag showed marked improvement up to 50% at all flight conditions and longitudinal stations. A comparison of the induced drag change for the two flight conditions revealed differences in the size of Y-Z region of benefit, indicating the vortex size may vary as a function of flight condition.

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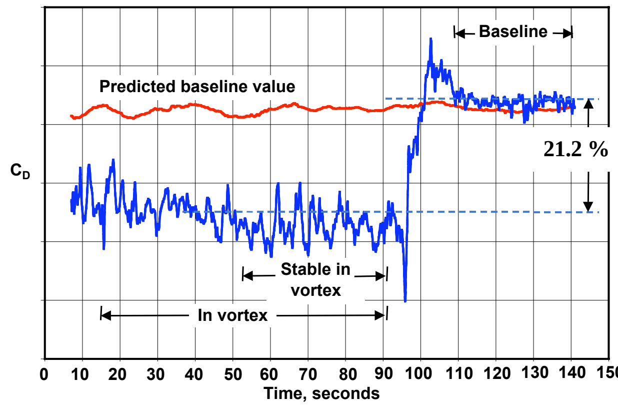
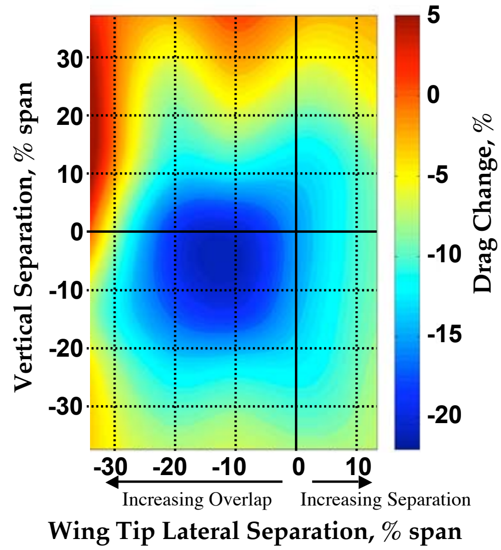
NASA F-18's in Formation with Smokewinder Activated



Example AFF Drag Results
Mach 0.56, 25,000 feet
55 feet aft nose to tail separation



F-15B Propulsion Flight Test Fixture (PFTF) Envelope Expansion

The Propulsion Flight Test Fixture (PFTF), flown on the F-15B, provides a unique, low-cost flight facility for the development and flight test of advanced propulsion systems. Flight research data can be obtained at subsonic, transonic, and supersonic speeds up to about Mach 2.0 and 1400 psf dynamic pressure. Propulsion experiments, including "cold" (unfueled) and "hot" (fueled) experiments such as inlet research, Rocket Based Combined Cycle (RBCC), and Pulse Detonation Engine (PDE) experiments may be carried.

The PFTF incorporates an extensive instrumentation and telemetry system, including a unique 6-axis force balance that provides 3 forces and 3 moments for the direct, in-flight measurement of propulsive thrust and aerodynamic drag. This unique feature may be useful for aerodynamic and other flight experiments in addition to propulsion research.

The PFTF has two main components: the pylon and the flight experiment. The pylon is mounted to the F-15B centerline station and transfers loads into the F-15B aircraft. The pylon provides volume for instrumentation and systems for the flight research experiment. The complete PFTF, pylon and experiment, has a maximum weight of 1600 lbs. The maximum allowable experiment weight is approximately 500 lbs. Experiment dimensions of approximately 107 inches in length and 12 inches in diameter can be accommodated.

The first phase of PFTF flight tests consisted of envelope expansion of the F-15B/PFTF configuration with a simple conical drag device mounted as an experiment. While primarily determining the flying qualities of the aircraft configuration, Phase One also validated performance of the force balance.

The second phase of PFTF flight tests will include a Pitot inlet, a total pressure flow distortion rake (downstream of the inlet), and a flow nozzle in the fall of 2002. During this phase, experimental hydrogen sensors will also be flown in support of the X-43 program.

Another flight test program in early 2003 will investigate performance of a proprietary inlet developed under an SBIR grant.

Phase III, with a fully integrated rocket, will bring the RBCC concept to flight test in the latter half of 2003.

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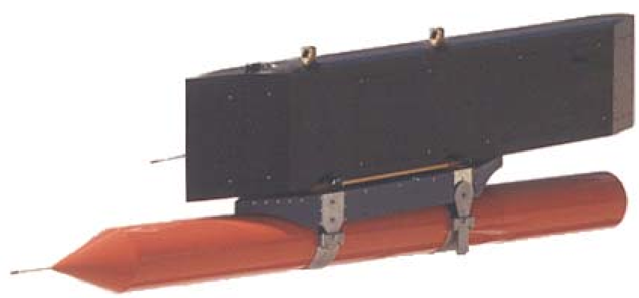
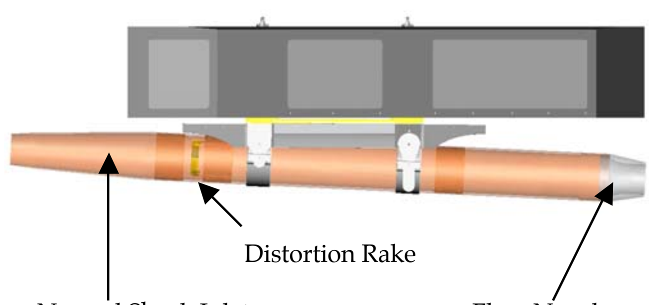
NASA F-15B with PFTF Experiment In-Flight



The Propulsion Flight Test Fixture
Shown in Phase I Configuration



PFTF Phase II Configuration

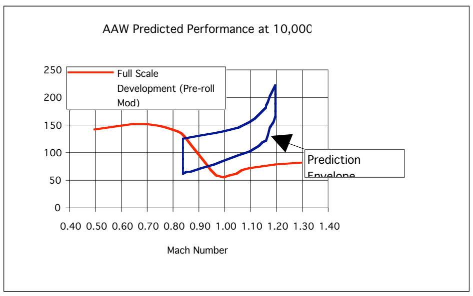
Active Aeroelastic Wing Flight Experiment

Objectives

The goal of the AAW Flight Research Program is to validate a new design paradigm in which a lighter, more flexible wing is used to improve overall aircraft performance. The AAW design philosophy contrasts sharply with conventional design practice in which wing flexibility is avoided at the cost of added structural weight. The F/A-18 aircraft is an ideal choice for AAW technology demonstration because of its thin flexible wing with multiple control surfaces and its high-speed flight capability, which together produce aeroelastic characteristics that can be exploited by the application of AAW technology.

Background

AAW Technology employs wing aeroelastic flexibility for a net benefit through the use of multiple leading and trailing edge control surfaces. The concept uses active leading and trailing edge control surfaces on a flexible wing structure to control structural deflections. The deflections or flexibility can then be used in roll control of the aircraft, which results in an optimal roll control effectiveness.



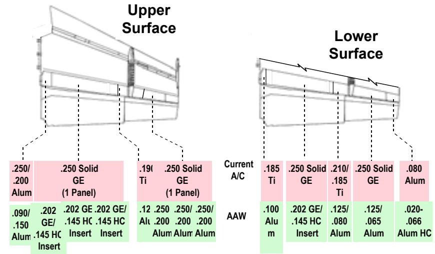
AAW Predicted Roll Performance

Aeroelastic flexibility characteristics were discovered in early F/A-18 flight-testing but were certainly not regarded as beneficial at the time. It was found then that aircraft roll performance dropped dramatically in the transonic flight regime due to the aeroelastic phenomenon known as aileron reversal, which, due to wing twisting, leads to roll in a direction opposite to that commanded.

The approach of the AAW program is to essentially restore the wings of the AAW flight test aircraft to their original more flexible structural configuration and then to design optimal control surface usage that exploits the aeroelastic effects resulting from this increased flexibility.

Aircraft Structural Modifications

The major change to the aircraft structure was in the aft box skins. These skins were changed to reduce the structural torsional stiffness and consisted of thinner skins and some material changes as shown below.



AAW Wing Panel Modifications

Control System

Aircraft Flight Systems Modifications

The leading edge flap drive system was modified to permit the portion of the flap outboard of the wing fold to operate independently as a maneuvering control surface from the inboard leading edge flap. Additional Power Drive Units (PDU's), as well as, Asymmetry Control Units (ACU's) were used to provide equivalent control to the OBLEF's as are currently present with the IBLEF's.

The Flight Control Computer's utilize a 68040 Research Processor. The extended memory and increased speed will provide control law development the flexibility needed for robust control law design. The interface between 701E/68040 shall be via Dual Port Ram (DPR).

Verification and Validation of the Flight control system has been conducted over the past several months and is nearing completion. This culminated from a joint effort between Boeing - St Louis and NASA-Dryden, in which, Boeing took the lead during Verification Testing and Dryden took the lead on Validation testing. Validation testing, which includes Piloted FMET /handling qualities assessment, will be complete by August 23, 2002.

Flight Test Plan

AAW technology flight-testing will commence through three phases to ensure a safe, thorough evaluation that addresses the complete set of research objectives.

The data from these phases of flight will be used to create a set of AAW control laws that will use optimal control surface usage to exploit the aeroelastic effects resulting from the increased flexibility.

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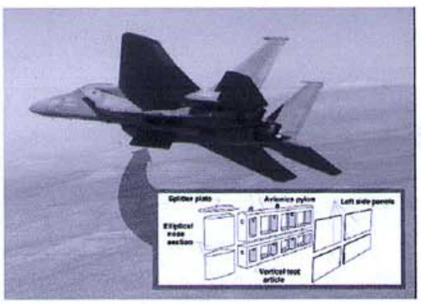
Aerostructures Test Wing

Background/Objectives:
The AeroStructures Test Wing (ASTW) provided a functional flight test planform to validate stability prediction algorithms and flight test techniques. The next generation tools for robust stability predictions will rèduce the currently high levels of risk and cost associated with flight flutter testing. The design flutter point is 0.80 Mach at 10,000 feet. The actual flutter point was found to be at 0.83 Mach at 10,000 feet. The primary objective of this project was to investigate pre-flight and on-line analysis tools for load predictions and stability estimation. In-flight flutter estimation algorithms were evaluated to determine their accuracy in safely predicting the onset of flutter instabilities and applicability towards an on-line flutterometer concept.



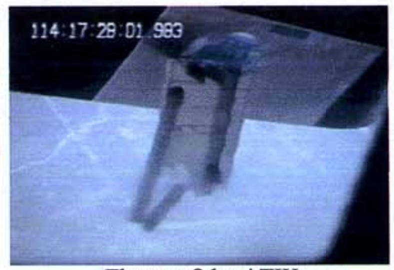
F-15B with Flight Test Fixture Aerostructures Test Wing Flutter Analysis Results

Approach
The research approach of the ASTW consisted of four main components: analytical modeling, ground tests, flight tests, and technology transfer. The main objective of the flight test was to gather static and dynamic load data for load predictions and stability estimations. A build up approach from 25% to 99% of the flutter speed was completed on the ASTW.

Results
The ground vibration test showed that the main structural modes were first bending at 13.79 Hz and first torsion at 20.79Hz.

The ATW was flown on 4 flight tests during April 2001. These flights included 21 test points with Mach numbers between 0.50 to 0.83 and altitudes between 10,000 and 20,000 *ft* such that the envelope expansion always increased dynamic pressure. The aircraft arrived on condition and then flew straight and level for 30 seconds to gather information about turbulence levels. After the stabilized run, the excitation system on the ATW was activated and response data was measured.

The ATW experienced the onset of flutter during an envelope expansion. Specifically, the system was being accelerated after the final test point at Mach 0.825 and altitude of 10,000 *ft*. The pilot was doing a very slow acceleration of approximately .01 Mach per second at constant altitude. The onset of flutter occurred at approximately Mach 0.83 and altitude of 10,000 *ft*.



Flutter of the ATW

Conclusions
The Aerostructures Test Wing was a successful experiment. Flight tests of the ATW were indeed able to demonstrate flutter. Data recorded during these flights have been used to predict the onset of flutter and demonstrate strengths and weaknesses of the corresponding prediction methods. The predictions from the damping method were initially poor but improved dramatically as the envelope was expanded. Conversely, the predictions from the flutterometer were initially slightly conservative but remained so throughout the flight testing even though more data was gathered. These results indicate a method to perform envelope expansion. The flight test should be initiated using the flutterometer at the low-speed test points to get an initial conservative estimate of the flutter speed. The envelope expansion at high-speed conditions should rely more heavily on the data-driven methods to finalize an accurate prediction of the exact speed at which flutter will be encountered. The combination of these approaches will allow for a more efficient flight test program.

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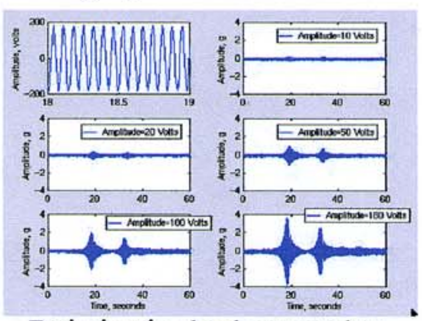
Piezoelectric Excitation for Ground and Flight Testing

Background/Objectives

A flight flutter experiment at NASA Dryden Flight Research Center, Edwards, California, used an 18-inch half-span composite model called the Aerostructures Test Wing (ATW). The ATW was mounted on a centerline flight test fixture on the NASA F-15B and used distributed piezoelectric strain actuators for in-flight structural excitation. One objective of the ATW was to investigate the performance of the piezoelectric actuators and test their ability to excite the first bending and torsion modes of the ATW on the ground and in flight.

Approach

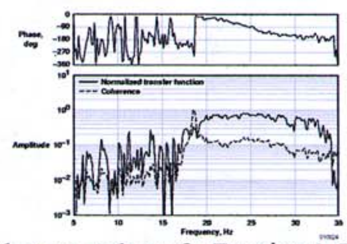
Structural excitation is an important requirement in any flight flutter test. Adequate excitation provides energy to excite all of the selected vibration modes at sufficient magnitudes to accurately assess stability from the response data. The piezoelectric actuation system on the ATW was designed and built to excite the wing with enough power to observe the structural modes during flight.



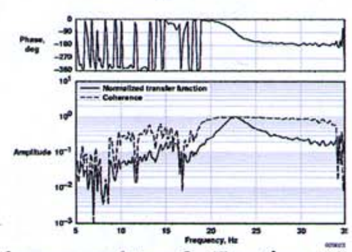
Excitation signal and response data

Conclusions

The piezoelectric excitation during flight consistently created a higher response level in the structure. However, the higher response did not always generate a better transfer function that could be used in signal processing. The coherence function was used as a measure of the quality of the transfer function. In order to get a good coherence above 0.90, the piezoelectric excitation level had to be approximately 3.5 times that of the turbulence levels. Stability algorithms that rely on the transfer function data should have some uncertainties incorporated to account for the poor coherence and transfer functions.

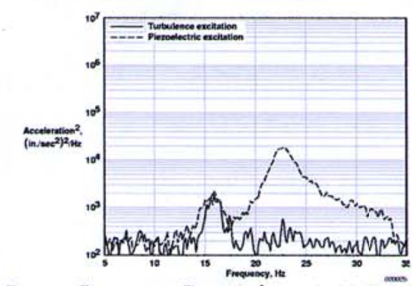


Coherance and transfer Function, 0.6Mach, 20,000 Feet



Coherance and transfer Function, 0.8Mach, 10,000 Feet

In flight flutter testing, it is important that all the critical structural modes be observable At the higher dynamic pressures where the turbulence levels were high, the autospectrums did not differ much from the piezoelectric excitation, showing that turbulence might have been good enough for basic frequency and damping calculations. But, at the lower dynamic pressures, the turbulence levels were not enough to excite the torsion mode of the wing, whereas the piezoelectric excitation excited this mode by an order of magnitude or greater.



Power Spectrum Comparison, 0.6 Mach, 10,000 Feet

Only the piezoelectric excitation was able to excite the two structural modes throughout the desired flight envelope, which is critical for any flight stability estimates.

Contacts

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Aging Theories for Estimating the Safe Flight Test Life of Air Borne Structural Components

Summary

B-52 aircraft pylon hooks are used to carry various types of air launch vehicles. Each hook has a critical stress point, a potential fatigue crack nucleation site after excess number of test flights (fatigue). The safety of the flight test hinges upon the structural integrity of the hooks. It is of vital importance to estimate the safe service life span (number of flights) of each hook before the initiation of any flight-test program.

New aging theories were developed to set the safe service life span of the hooks. The new theories predict more conservative service life of the hooks as compared with the previous Ko first- and Ko second-order aging theories.

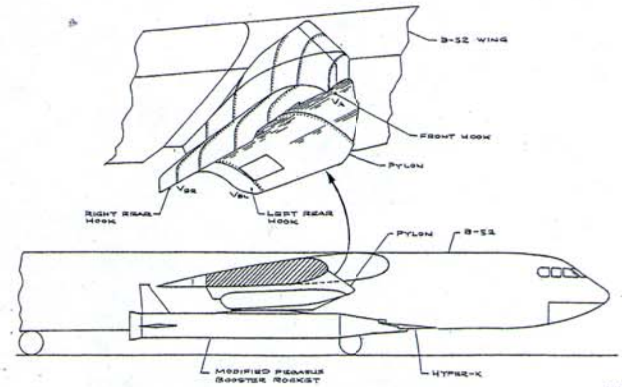

Figure 1. B-52 launching aircraft pylon carrying winged Pegasus rocket.

Objective

To develop more accurate aging theories for the prediction of safe flight-test life span of B-52 pylon hooks which will carry the Pegasus winged rocket.

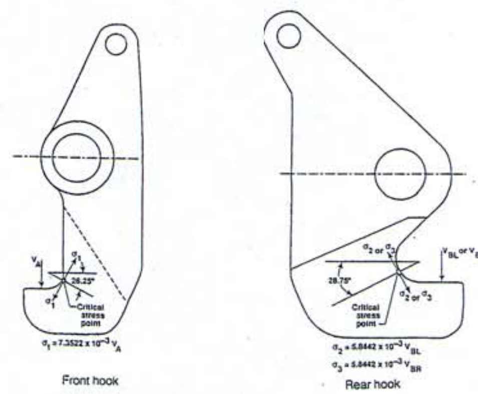

Figure 2. Critical stress points in B-52 pylon front and rear hook.

Approach

Fracture mechanics, Walker crack growth theory, and half-cycle theory are combined to develop the first new aging theory.

Also, random load spectra were represented with equivalent constant amplitude loading spectra for establishing the second new aging theory.

The initial crack size based on the limit hook load was calculated using the fracture mechanics. Next, the half cycle theory was used to calculate the crack growth using the random loading spectrum of each hook. Then the safe number of flights was established for each hook carrying the Pegasus winged rocket.

Results

For the case of B-52 aircraft carrying the Pegasus winged rocket, the available number of flights calculated from the new and previous aging theories are compared in Table 1. The new aging theories give more conservative results and ensure safety of the flight tests.

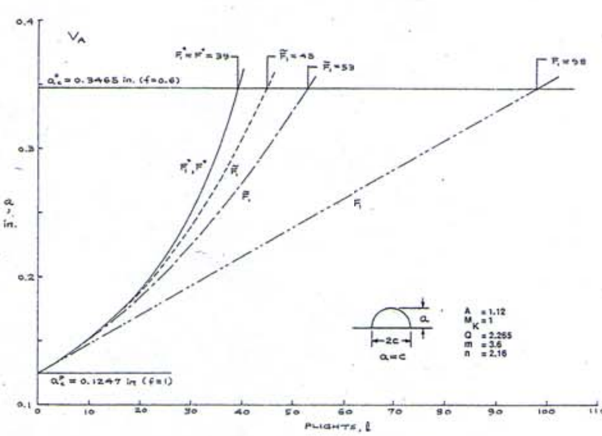

Figure 3. Crack growth curves in *a – flights* space calculated from different aging theories; front hook.

Table 1. Available number of flights for B-52 hooks carrying Pegasus winged rocket

Flights	Ko 1st order	Ko 2nd order	New Ko theories
Front hook	53	45	39
Rear hook	100	85	77

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Thermostructural Analysis of Hyper-X Hypersonic Flight Research Vehicle Wing Structures

Summary

Heat transfer and thermal buckling analyses were performed on the wing structures of Hyper-X hypersonic flight research vehicle. The wing mid-span segment was chosen for the heat transfer analysis. Thermal stress and thermal buckling analyses were conducted on the entire wing panels. The region with the weakest thermal buckling strength was identified. The calculated buckling temperature of the weakest panel was found to be about 20% higher than the wing panel peak temperature, and therefore, the Hyper-X wing panels will not buckle, and the panel/spar welded sites will not fail in shear during the Mach 7 mission.

Objective

To investigate the thermal buckling strength of Hyper-X wing skin panels and the strength of the wing panel/spar welded sites subjected to Mach 7 aerodynamic heating. To establish thermal buckling temperature magnification factors to relate the buckling solutions for the dome -shaped temperature profile heating to those for the uniform temperature profile heating.

Approach

Lockhheed Thermal Analyzer (generalized FORTRAN program based on an electrical analogy of capacitors and resistors) was used for the heat transfer analysis to calculate the wing structural temperatures.
SPAR finite element computer program was used to calculate thermal stresses and buckling temperatures of the wing panels.

Results

1. The region with the lowest thermal buckling strength of the Hyper-X wing panels is located in the center region of the aft wing panels.
2. The buckling temperature of the weakest Hyper-X wing panel is 1,389 ˚F, about 20% higher than the peak wing panel temperature 1,153 ˚F. Therefore, the Hyper-X wing panels will not buckle during the Mach 7 mission.
3. The maximum shear stress at a wing panel/spar welded site is about 71% of the shear failure stress of the welding material. Therefore, welded site will not fail in shear during Mach 7 mission.
4. Thermal buckling temperature magnification factors were established to scale up the buckling solutions for the uniform temperature profile heating to give the buckling solutions for the dome -shaped temperature profile heating.

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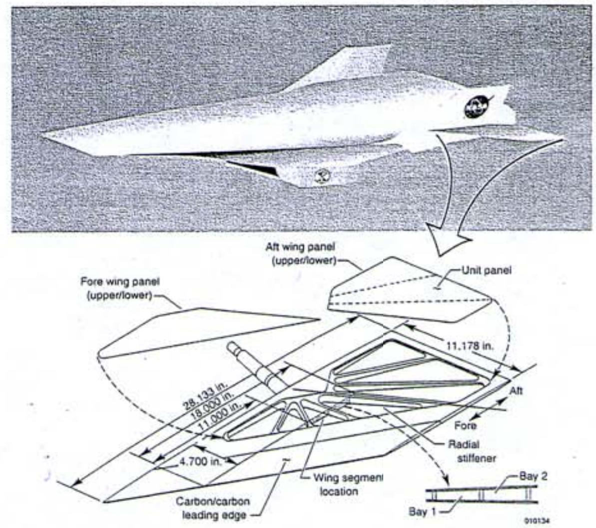


Figure 3. Unconventional wing structures of Hyper-X hypersonic flight research vehicle.

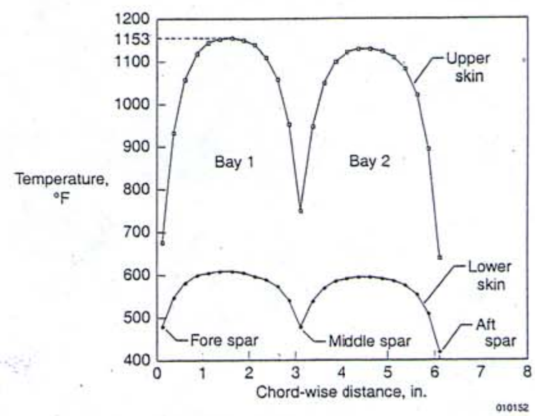
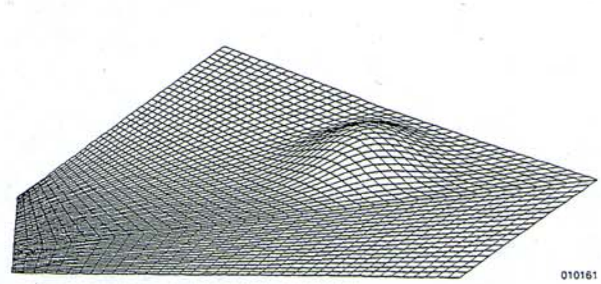


Figure 2. Chord-wise distributiuon of Hyper-X wing segment skin temperatures; t = 89 sec, 0.04 in. web contact.



(d) CL-CL condition.
Figure 3. Buckled shapes of aft panel

Developing Uncertainty Models For Robust Flutter Analysis Using Ground Vibration Test Data

Summary:

A ground vibration test can be used to obtain information about structural dynamics that is important for flutter analysis. Traditionally this information, such as natural frequencies of modes, is used to update analytical models that are used to predict flutter speeds. The ground vibration test can also be used to obtain uncertainty models, such as natural frequencies and their associated variations, that can update analytical models for the purpose of predicting robust flutter speeds. This approach is demonstrated using ground vibration test data for the Aerostructures Test Wing. Different norms are used to formulate uncertainty models and their associated robust flutter speeds in order to evaluate which is least conservative.

Objective:

A structure known as the Aerostructures Test Wing (ATW) is being utilized at NASA Dryden Flight Research Center. This wing serves as a testbed for investigating pre-flight and on-line methods of predicting the onset of flutter. The wing operates in a realistic flight environment through the use of an F-15 aircraft and associated Flight Test Fixture. The Flight Test Fixture is a host carrier that rests under the center of the fuselage behind the engine intakes. The ATW will be mounted horizontally to the side of this carrier for flight tests. The wing is shown in Figure 1.

The objective is to formulate an uncertain value of natural frequency that represents the amount of variation that is observed in the test data.

Results:

It is shown that analyzing the data using an ∞-norm approach generates a model with less uncertainty than corresponding 1-norm or 2-norm approaches. The smaller uncertainty is useful for the flight test community because the associated robust flutter speeds can be computed with less conservatism. The Aerostructures Test Wing is used to demonstrate this procedure. A ground vibration test is performed on the wing and used to generate a range of natural frequency estimates. These estimates are analyzed using the different norm approaches to formulate uncertain models that cover the entire range of observed variations. The main feature of interest in Figure 2 is the variation in value of the robust flutter speed given by V_{rob}. Specifically, the model based on ∞-norm analysis has a robust flutter speed that is 42 KEAS higher than the corresponding speed computed using the 1-norm analysis. This clearly shows that the ∞-norm approach results in the least conservative answer.

Status/Plans:

The Aerostructures Test Wing was successful in demonstrating flutter during flight test, see Figure 3. Initially, the inability of the flutterometer to converge to the correct solution was actually a result of model updating. The flutterometer was programmed to only update the uncertainty description such that the theoretical model was never changed. Follow-on work showed that a model updating procedure improved the flutter predictive capabilities dramatically with reduced conservatism in uncertainty modeling.

Figure1 : Aerostructures Test Wing

Norm	Vnom	Vrob
1	371	317
2	384	343
∞	392	359

Figure 2. Nominal and Robust Flutter Speeds in KEAS

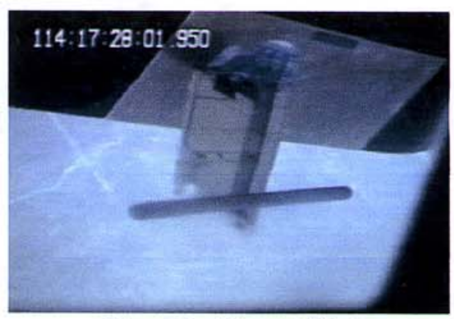


Figure 3.: Planned Flutter

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Nonstationary Dynamics Data Analysis with Wavelet-SVD Filtering

Summary:
Nonstationary time-frequency analysis is used for identification and classification of aeroelastic and aeroservoelastic dynamics. Time-frequency multiscale wavelet processing generates discrete energy density distributions. The distributions are processed using the singular value decomposition (SVD).

Objective:
The primary objective is the automation of time-frequency analysis with modal system identification. The contribution is a more general approach in which distinct analysis tools are merged into a unified procedure for linear and nonlinear data analysis.

Approach:
Discrete density functions derived from the SVD generate moments that detect the principal features in the data. The SVD standard basis vectors are applied and then compared with a transformed-SVD, or TSVD, which reduces the number of features into more compact energy density concentrations. Finally, from the feature extraction, wavelet-based modal parameter estimation is applied.

Results:
Aeroelastic and aeroservoelastic flight data from an F18 aircraft are investigated and comparisons made between the SVD and TSVD results. Input-output data is used to show that this process is an efficient and reliable tool for automated on-line analysis. Figures 1 and 2 show a persistent oscillation of the wing bending mode at 8-9Hz (first fuselage and wing bending modes). The top plot in figure 2 is the original energy density, and the bottom is the log-scaled TSVD-filtered distribution. Time-dependent modal frequency and damping ratios estimated with the TSVD filter are displayed in figure 3.

Benefits:

- Method using a standard SVD demonstrates the ability to discriminate the most significant and other less dominant dynamics.
- Method called the transformed-SVD, or TSVD, optimally rotates the singular vectors to create a more precise filter for finer tracking
- Both methods complement each other and show promise for automatic feature extraction supplemented with noise removal, residual analysis, and uncertainty estimation.

References:

- AIAA SDM Conference, April 16-19, 2001, Seattle, WA
- NASA/TM-2001-210391
- Submitted to *Mechanical Systems and Signal Processing*

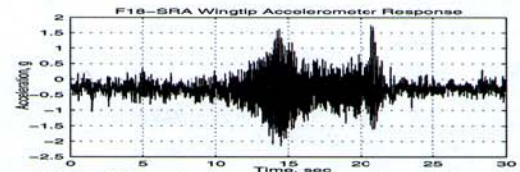

Figure 1: F18-SRA wingtip accelerometer response from a multisine aileron input.

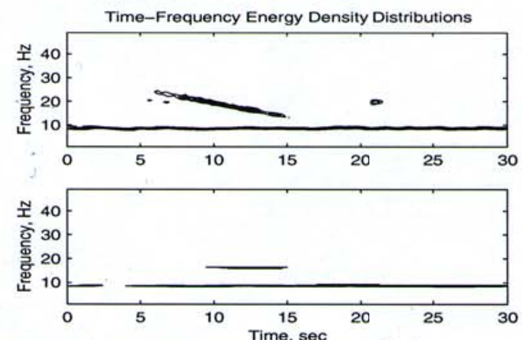

Figure 2: Contour plots from original wavelet decomposition (top) and TSVD-filtered (bottom, log-scale), scalograms of F18-SRA fuselage lateral accelerometer.

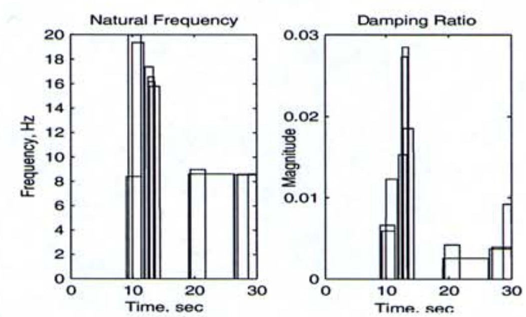

Figure 3: TSVD estimated modal frequencies (left) and damping ratios (right) from F18-SRA fuselage accelerometer.

Contact:
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Aeroservoelastic Robust Model Development from Flight Data

Summary:

Uncertainty modeling is a critical element in the estimation of robust stability margins for stability boundary prediction and robust flight control system development. There has been a serious deficiency to date in aeroservoelastic data analysis with attention to uncertainty modeling. Uncertainty can be estimated from flight data using both parametric and nonparametric identification techniques.

Objective:

The model validation problem is to identify aeroservoelastic models with associated uncertainty structures from a limited amount of controlled excitation inputs over an extensive flight envelope. The challenge is to update analytical models from flight data estimates while also deriving non-conservative uncertainty descriptions consistent with the flight data.

Approach:

Transfer function estimates are incorporated in a robust minimax estimation scheme to update models and get error bounds consistent with the data and model structure. Uncertainty estimates derived from the data in this manner provide an appropriate and relevant representation for model development and robust stability analysis. The method incorporates parametric and nonparamteric uncertainty into various uncertainty structures for quantitative measures of robust stability relating to parameter variations and unmodeled dynamics.

Results:

This model-plus-uncertainty identification procedure is applied to aeroservoelastic flight data from the NASA Dryden Flight Research Center F-18 Systems Research Aircraft (F-18 SRA). Figures 1 and 2 depict structured uncertainty descriptions which are used to define the analysis plots in figure 3. The bottom two plots are the nominal input and output multivariable transfer functions, the middle two are with input-output complex uncertainty, and the upper two are with both complex and real-parametric identified uncertainty. The left set is with multiplicative uncertainty and the right set with additive uncertainty.

Benefits:

- Aeroservoelatic model identification with uncertainty uses robust data-oriented procedures for model development.
- Transfer functions incorporated in a robust minimax estimation scheme to identify input-output parameters and structured error bounds consistent with the data.
- Uncertainty estimates derived from the data in this manner provide appropriate and relevant representations for robust stability analysis useful for model validation and control system design.
- This procedure is an automated, efficient, and reliable approach for analysis of numerous flight data sets for robust stability and model development.

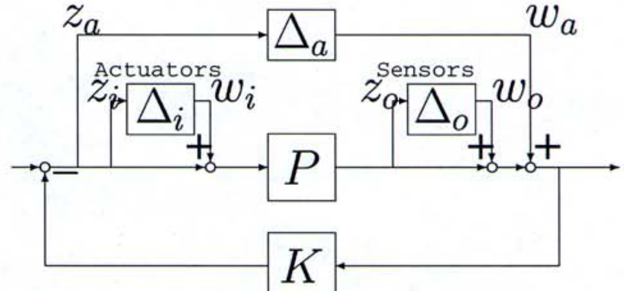

Figure 1: Aeroservoelastic model with additive and input-output multiplicative complex uncertainty.

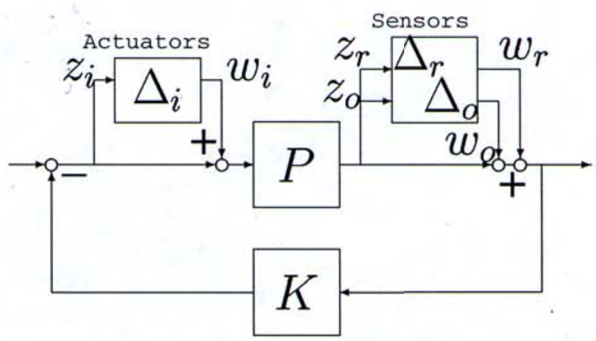

Figure 2: Aeroservoelastic model with input-complex and output-mixed multiplicative complex uncertainty.

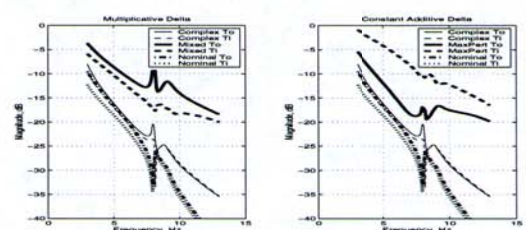

Figure 3: Nominal, complex-uncertainty, and mixed-uncertainty complementary sensitivities at the inputs and outputs.

References:

- International Forum on Aeroelasticity and Structural Dynamics, Madrid, Spain, June 2001
- NASA/TM-2001-210397
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AAW Strain-Gage Loads Calibration Test

Summary

The Active Aeroelastic Wing F/A-18 aircraft was ground test loaded for the purpose of wing strain gage load calibration (See Figure 1). Seventy-two load cases were applied symmetrically to the wings while the applied loads and 200 strain gage bridge outputs were recorded. A total of 20 component load equations, including many back-up equations were derived from this database. Peak applied net vertical load reached 120,000 pounds – enough to lift four F/A-18's off the floor. The required data were produced without damage to the test aircraft.

Objectives

The primary goal of this test was to produce a database suitable for deriving load equations for left and right wing root and fold shear, bending moment and torque and all eight wing control surface hinge moments. A second goal was to strain-gage data through both the laboratory data acquisition system and the onboard aircraft data system as a check of the aircraft system.

Approach

Thirty-two hydraulic jacks were used to apply load through whiffletrees to 104 tension-compression load pads bonded to the lower wing surface. Sixteen of these load columns are shown in Figure 2. The load pads covered 60% of the lower wing surface (including control surfaces). A series of 72 load cases were performed, including single-point, double-point, and distributed load cases. Applied loads reached 70% of the flight limit load. Load and deflection data and the airframe were extensively monitored throughout the testing.

Results

Maximum wingtip deflection reached nearly 16 inches. A sample of some of the strain and load database is shown in Figure 3. The data were very good quality and have been used to produce flight load equations exhibiting very good accuracy. The data produced through the 12 bit aircraft data system compared well with corresponding data from the 14 bit laboratory data system.

For more information see NASA/TM-2002-210726

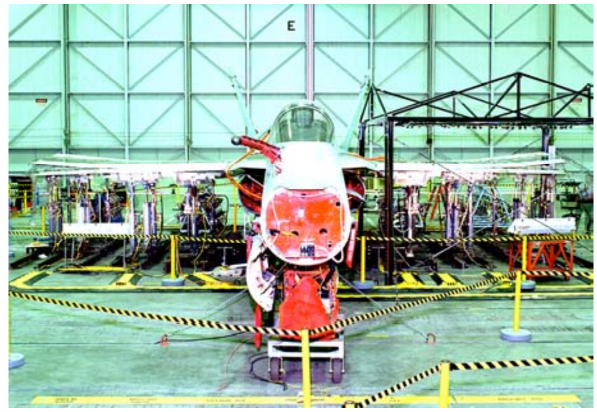

Figure 1. AAW Under 0% and 100% Load (Composite Photo)

Figure 2. Right Wing Loading Hardware

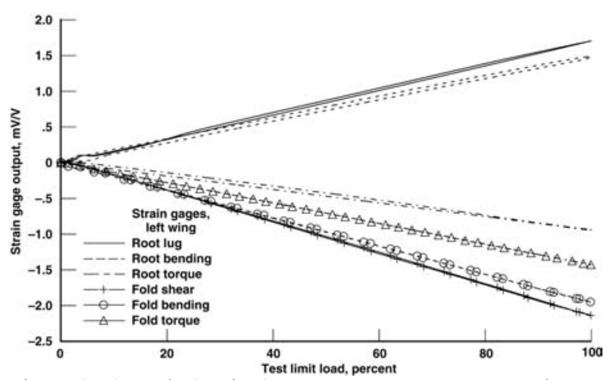

Figure 3. Sample Strain Gage Responses to Test Load

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AAW Wing Torsional Stiffness Testing

Summary

The left wing of the Active Aeroelastic Wing (AAW) aircraft was ground load tested to quantify its torsional stiffness. The test was performed in the DFRC Flight Loads Laboratory in November 1996 and again in April 2001 after a wing skin modification was performed. The modified wing was shown to have more torsional flexibility and less structural hysteresis than the unmodified wing.

Objectives

The primary objectives of these tests were to characterize the wing behavior before first flight, and provide a before-and-after (modification) measurement of wing torsional stiffness.

Approach

Two streamwise load couples were applied in-phase (see Figure 1) using two load columns at the wing tip and two load columns at a mid-span station. Wing deflections were measured by 48 digital dial gauges and string potentiometers arrayed under the test wing. Any rigid-body motion of the overall airframe was monitored through additional string potentiometers. Various wing rigging configurations were tested.

Results

Figure 2 shows a sample of the wing deflections measured during one load application. Figure 3 shows a summary of the streamwise twist results from the 1996 test, the 2001 test, and the finite element predictions. These data indicate that the modified AAW wing is slightly more flexible than the un-modified wing. The characteristic structural hysteresis of the modified wing is slightly less than that of the un-modified wing. This is attributed to the fastener rework that was done. These data were also used to tune the finite element stiffness model for future use.

For more information see NASA/TM-2002-210723.

Figure 1. AAW Left Wing Under Negative Torque Load

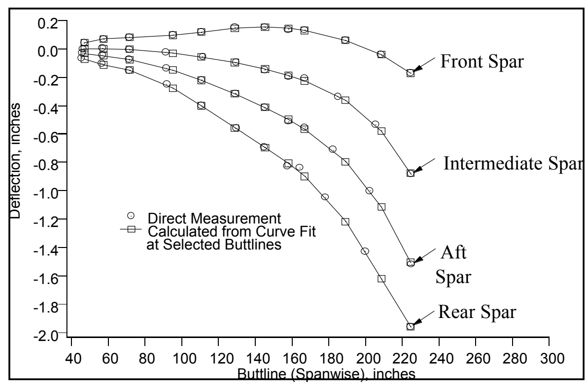

Figure 2 . Sample of Deflection Data

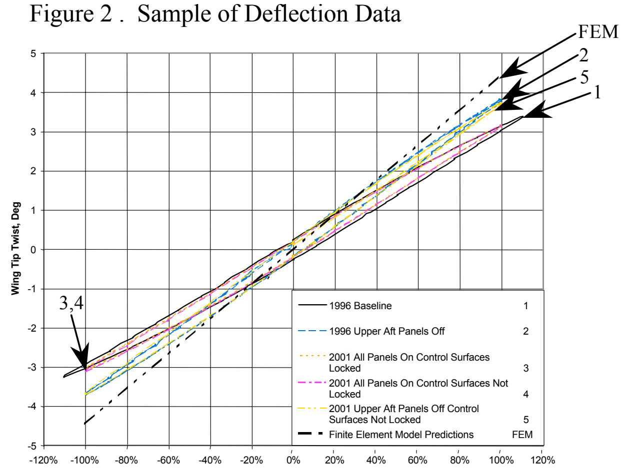

Figure 3: Summary of Wing Tip Twist vs % Test Limit Load

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Realtime Flight Data Remote Site Node Development

Summary:
NASA Dryden Flight Research Center has developed and deployed the capability to enable buffered access to live test data for remote users over IP networks. This network service gives users simultaneous access to current and historical data necessary for time constrained distributed computing and process automation tasks. This report describes efforts underway to scale capabilities of a centralized content distribution server in anticipation of growing demand.

Background:
As part of a project called AeroSAPIENT* funded by NASA Ames, a network of Ring Buffered Network Bus (RBNB) middleware servers for live data streams was deployed at Dryden, Glenn, Ames, and Langley Research Centers. These servers managed live data feeds to and from an aircraft-based server via satellite. These servers remained in place after the experiment for flight projects interested in using the new data sharing capabilities. Opportunities for reducing control room staffing and operating costs while increasing customer productivity were recognized. However, use of the Java virtual machine in embedded or realtime applications was noted as a concern given that the virtual machine is evolving and that sufficient deterministic behavior in dynamic environments is not yet guaranteed. Given that the RBNB is Java software implementing dynamic random access to both realtime and buffered data, it was determined that performance limitations of the server application and the underlying Java virtual machine needed to be investigated.

Approach:
A measurement and telemetry networks laboratory was established to conduct network-centric measurement process automation research. A second generation remote site node was designed and procured. The chief engineer for Dryden's Western Aeronautical Test Range (WATR) flight support organization defined the following benchmark problem to study: a 1 Msps input source telemetry stream (32Mbit/s) across 5000 parameters, with some realtime value-added calculation (EU, derived parameters) support and the ability to sustain 100 simultaneous users. The target maximum latency is 100msec. The flexibility to scale realtime processing requirements and total user load guarantees that this approach is challenging enough to locate bottlenecks in software and underlying hardware infrastructure. Conversely, this benchmark problem is mostly a one-way information flow and does not focus on collaborative measurement processiong environments with many distributed data sources and the problems that arise in that environment.

Status:
Initial tests showed that a simple data sink could sustain 100Mbit Ethernet hardware at 80 Mbits per second (2.5 Msps) and that as single client can pull 5000 parameters out at 1.7 Msps using the then-current V1.1.2 server. Practical issues such as underlying network topology, load segregation for parallel servers, and specific client demands become important here. In addition, the use of firewalls and network address translation as might be used in realistic deployments were observed to throttle the performance as viewed from the downstream client.

* *AeroSAPIENT: Aeronautical Satellite-Assisted Process for Information Exchange using Network Technology*

Fig. 1. Front view of prototype second generation remote site node containing multiple servers, network firewall and switches, and uninterruptible power supplies.

Latency metrics from within the RBNB were used to study the effects of process preemption by the virtual machine garbage collector and the underlying operating system. While our target latencies appear achievable, these metrics reveal differences and sensitivities across platforms and implementations.

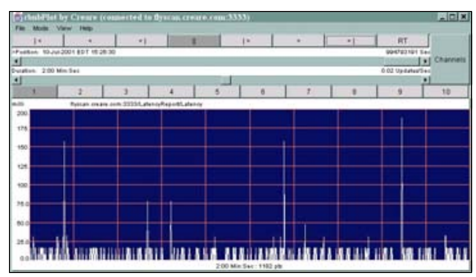

Fig. 2. Typical Latency Metric: IBM JVM 1.3 on Linux/dual 500Mhz CPU, 4 clients with 5000 parameter / 1Msps source: average latency ~30 msec, max latency ~300msec.

Plans:
System configuration design using gigabit Ethernet front end and increased RAM storage necessary to support deeper random access caches and 100 users is being implemented and tested in 2002. This work will be compared with similar activities being funded by the DoD in support of flight test and evaluation.

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Realtime Telemetry Networks Initiative

Summary:
Adapting network technology for trustworthy and high performance operation in harsh wireless environments common to the aerospace industry is a challenge that warrants attention now. A joint NASA/DOD program is envisioned focused on developing the technology necessary for transitioning the telemetry industry to a network-based paradigm.

Background:
Network communication hides from applications many of the ugly details of communicating dynamically changing data between groups of physical 'things'. Standard transport protocols and services decouple applications from hardware dependencies, enabling tremendous advances in productivity and progress. Wireless networking technology has matured such that it is now feasible to consider networking solutions problems in telemetry, which at this time is still a point-to-point or link-layer communication strategy.

Since 1998, NASA's Dryden Flight Research Center, Edwards, Calif., has been part of an informal group of government visionaries that include representatives of DOD range, DOD space, NASA space, and NASA range interests. In August 2000, this group recognized similarities across a range of problems and defined a common set of needs with the following statement:

Considering that:
1. Both NASA and DOD operate and maintain critical space and range communications infrastructure that is often based on technologies that are not currently well integrated with exploding terrestrial "Internet" capabilities.
2. Users are increasingly demanding that the interfaces and communications services provided to them by the NASA and DOD space and range infrastructure should be identical to familiar and well-supported Internet capabilities.
3. There are strong economic pressures for government agencies to share space and range infrastructure by developing interoperable systems based on open standards.

And recognizing that:
1. There is growing and relentless pressure to make more effective use of space and range wireless communications capacity.
2. The next stage of Internet evolution will be the rapid development of commercial "untethered" wireless communications services.
3. Existing space and range infrastructure needs to be re-engineered and modernized to provide increased levels of performance while staying within relatively fixed operations and maintenance budgets.
4. The space and range infrastructure may benefit significantly from the assimilation and extension of these emerging commercial wireless Internet technologies.

It is recommended that:
1. NASA and DOD should jointly mount an initiative with the objective of rapidly enabling the re-engineering of current space and range infrastructure to more fully and effectively exploit commercial Internet technologies.
2. This initiative should work closely with the emerging wireless Internet community to jointly develop open standards that are capable of being adopted (and extended as necessary) for use within space and range infrastructure.

Objectives:
The objectives of the realtime telemetry networks (RTTN) initiative focus on common solutions to similar problems. Specifically, RTTN seeks to to identify and mitigate the risks associated with networking in the management of aerospace systems, and to accelerate adoption of useful and productive network oriented standards and solutions to test and measurement applications. There are a number of risk areas in individual components, layers, and systems that need to be mitigated in order to provide long term solutions for the industry.

Approach:
The problem is viewed as separable into three layers subject to constraints not typically addressed on mainstream terrestrial networking applications.

- Constrained Applications
- Constrained Networking Services
- Constrained Links and Wireless Devices

For each of these layers there exist three stages to developing new capabilities

- Requirements and Architecture definition
- Standards and Prototype Demonstrations
- Field Demonstration and Rollout

Complex systems-oriented applications are expected to include demonstration of full end-to-end connectivity between sensors and decision makers, thereby prototyping by example the RTTN concept of operations.

Status/Plans:
DOD Department of Operational test and Evaluation will be funding a Joint Improvement and Modernization Program called 3rd Generation Range and Space Wireless Networks, starting with risk mitigation projects (Test Technology Demonstration and Development) in 2003. The full program, referred to as "3G RSWN" is expected to begin in 2004.

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Research Environment for Vehicle-Embedded Analysis

Summary:
NASA's Dryden Flight Research Center, Edwards, Calif., in cooperation with Air Force, Navy, and NASA Ames Research Center is researching advanced network-based and distributed data acquisition tools and methodologies applicable to mobile aerospace environments. The REVEAL system (Research Environment for Vehicle-Embedded Analysis using Linux) is a small generic data acquisition node that will allow researchers to explore the feasibility of the Linux operating system and emerging interoperability tools such as Java and Extensible Markup Language (XML) as a platform for distributed computing, sensor webs, and intelligent distributed control applications. Successful demonstration of this risk-mitigating research can be leveraged to benefit a variety of terrestrial fixed, terrestrial mobile, and airborne wired and wireless applications.

Background:
Today we see an evolving landscape of aerospace technology that is increasingly influenced by information technology. These days, dominant themes of research objectives include intelligent systems, autonomous systems, reconfigurable systems, nondeterministic systems, integrated prognostic health management systems, high-speed digital instrumentation systems, smart sensor systems, sensor "webs", wireless sensor systems, and cooperative or collaborative systems of systems. The distinction between aerospace technology and information technology is blurring and we are starting to see an evolution toward distributed information architectures on and across aerospace vehicles. In the future, vehicles and vehicle subsystems will be viewed as nodes collaborating on networks. In consideration of latency and determinism constraints placed on many applications in this arena, the notion of the *realtime telemetry and measurement network* arises as the general area of research requiring attention.

Meanwhile, the evolution of the Linux operating system, Java programming language, and Extensible Markup Language (XML) is observed to have potentially significant advantages for aerospace and data acquisition applications. Advantages include increased productivity, reduced development and life cycle costs, and interoperability with systems, applications, and infrastructure common to terrestrial applications.

Noting that Linux and Java are new to high performance real-time applications involving data acquisition and also noting that the state of the art in networks of embedded systems is a very immature technology area, it was decided that a systems approach in applying these technologies would be useful for revealing and in some cases mitigating the risks that currently prevent broader use of these technologies in the aerospace industry. The REVEAL project was identified as a systems-oriented, cost-effective, and leading edge approach for researching realtime measurement and telemetry networks problems.

Objectives:
The primary objective of the REVEAL project was to design and build a number of small, rugged, low power, inexpensive, generic, extensible data acquisition systems that would have both wired and wireless network connectivity. These systems must be configurable over the network and in general be a useful tool for measurement networks research in the lab with application in land mobile and airborne environments.

Approach:
A number of internal sensors provides variability in sensor rates and data volume while providing useful information. In addition to having the ability to acquire external analog and digital data sources, the baseline embedded sensor configuration provides knowledge about its external environment (where it is, its velocity and acceleration, which way is up, which way is north, video, audio) and something about its own health and internal environment (temperature, pressure, line and bus voltages, operating system metrics). Other features include

- HardHat Linux with RTLinux realtime core.
- Weathertight PC104/PC104+ hardware
- VME hardware rehosting straightforward
- Wired and wireless network connections
- Generic "user job" software architecture
- Self-configuring and self-documenting via XML
- Dynamically configurable user-processing
- Ability to acquire data from other nodes on network
- Use of RBNB/DataTurbine for caching of at-rate data
- Web browser interface
- Embedded sensors and external wired acquisition

Status/Plans:
Hardware for six units have been purchased. Software implementation is in progress. Applications as risk mitigation for future NASA/DOD telemetry networks and space access programs, on-board computing networks, intelligent vehicle systems, realtime emergency response and remote access applications are being advocated and funding is being sought.

Baseline REVEAL configuration (S/N 001)

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Strategic Computing Initiative for Situational Awareness

Summary:
NASA Dryden Flight Research Center has initiated efforts to proactively align information technology tools and techniques with strategic needs and interests of itself and its stakeholders. Technical and business process automation is needed to continually improve value and return on investment in flight research. Automation, integration, and interoperability increase productivity, efficiency, and quality. The long-term objective is to increase situational awareness for Dryden team members and reduce risk by connecting people with the information they need.

Background:
Today we see an evolving landscape of aerospace technology that is increasingly influenced by information technology. These days, dominant research topics include intelligent systems, autonomous systems, reconfigurable systems, nondeterministic systems, integrated prognostic health management systems, high-speed digital instrumentation systems, smart sensor systems, sensor "webs", wireless sensor systems, and cooperative or collaborative systems of systems. The distinction between aerospace technology and information technology is blurring and we are starting to see an evolution toward distributed information architectures on and across aerospace vehicles. It is NASA's job to understand and influence the evolution of intelligent complex adaptive systems.

On the ground, rapid advances in network services and interoperability technologies feed a software industry eager to provide solutions for business process automation for increasingly distributed teams and organizations. Group collaboration, shared workspaces, data mining on vast digital warehouses, middleware services, etc., are promoting the web browser to be the knowledge worker's portal into cyberspace. Pursuit of NASA's mission demands that we understand and leverage this revolution.

Teams with nontraditional multidisciplinary skills will be necessary to solve these problems. Whether terrestrial or airborne, the creation of situational awareness infrastructure in a manner that is trustworthy, cost effective, and measurably reduces risk over time is a challenging but important systems problem.

Approach:
The strategic computing initiative is organized into three focus areas, each with three subelements. This approach is a strategy for adding new functionality to information systems infrastructure.

1 Strategic Technologies Research and Development addresses risk mitigating network systems research and development

1.1 Documentation Networks. *Lack of access* to the right information at the right time is a fundamental source of risk that often gets accepted without realizing it. This subelement advocates, adopts, adapts, and deploys on-line web-portal document management and shared workspace tools for individuals and groups.

1.2 Content Distribution Networks. Underneath applications are network protocols and physical links that actually move data around. Security and performance constraints often demand ad hoc solutions or altering of existing physical network infrastructure. This subelement works to eliminate performance bottlenecks under increasingly challenging security constraints.

1.3 Measurement and Telemetry Networks Technical process automation involving live data and integration of those data with larger enterprise information management infrastructure is an common problem without common solutions. This subelement prototypes tools and techniques for sensor-to-user network connectivity with emphasis on unique performance constraints imposed by data acquisition on underlying transport.

2. Strategic Operations
2.1 Core Administrative Process Automation This subelement encompasses financial and human resource management and is currently addressed by the Agency-wide Integrated Financial Management Program (IFMP).

2.2 Core Management Process Automation Executive management increasingly recognizes that IT's ability to create value through intangible assets is profound but are frustrated with their inability to effectively measure and manage these assets. This subelement encompasses project management metrics and forecasting focused on observing organizational performance relevant to strategy.

2.3 Core Technical Process Automation migrates engineering ground and flight test operations to efficient network operations and on-line collaboration, enabling a form of virtual flight research center, or *e*FlightTest. Value derived by minimizing the time between data acquisition and decisions based on data analysis.

3. Strategic Aerospace Systems element assesses aerospace flight projects from an intelligent complex adaptive system-of-systems perspective.

3.1 Intra-vehicular Technologies. On-board networked computing systems, integrated vehicle health management; cockpit situational awareness, vehicle autonomy; intelligent/adaptive flight control.

3.2 Inter-vehicular Technologies. Trustworthy inter-vehicle networking, telemetry networks, remotely piloted vehicles, and secure wireless links.

3.3 Multivehicular systems. Formation flying; swarming; goal-seeking aerospace platforms.

Status:
A number of accomplishments have occurred under strategic technologies research and development banner. Alignment with center strategy in progress

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Vehicle Health Monitoring Toolkit

Summary:
NASA Dryden Flight Research Center is pursuing object-oriented software tools to aid the design, analysis, implementation and use of machinery health management systems. An open, scaleable health management software system is targeted that will enable the configuration and initiation of remote algorithms that will reduce raw sensor data into relevant health information for both novice and sophisticated designers and users. The system is intended to be useful for a variety of airborne and terrestrial prognostic and diagnostic automation applications.

Background:
Widespread use of on-line health management has the potential to deliver just-in-time maintenance information and safety related situational awareness while reducing cost. Vehicle health management algorithms and components are maturing and warrant implementation, but the lack of software toolkits for designers, systems integrators, and users of health management systems represent a major obstacle to progress. Current applications are typically custom written for specific systems. Such solutions are costly in development and training resources and rarely get integrated into subsequent enterprise information management schemes.

There is an increasing emphasis on health management systems and condition-based maintenance systems in engines, vehicles, factories, and even homes. A great deal of research is being conducted in academic and government establishments on system modeling and techniques for fault identification. However, these systems will not be widely implemented until accessible software tools are available to the developers and users in the field. As the design of each health monitoring system is dependent on the system being monitored, the toolkit must provide generic tools that can be scaled and structured to match the functionality of the system. Emerging and established standards exist that can contribute to useful solutions. XML for metadata management and system interoperability, IEEE 1451, and Fieldbus Foundation for sensor and device description management are standards that need to be considered in the solution approach.

Approach:
The goal of this project is to develop a suite of object-oriented software tools to aid the design, analysis, implementation and use of health management systems. Object-oriented paradigms with graphical user interfaces have revolutionized many fields. A key innovation required for health management software is the ability to analyze both current and historical data from a range of sources in real time. This project builds on middleware designed for managing measurements on-line, over local and wide area networks. The underlying technology is extended here with an intuitive and flexible graphical user interface, demonstrating state-of-the- art health monitoring algorithms in an object-oriented toolkit. Requirements and capabilities for a general-purpose system include the ability to:

- Combine information and data from many sources over local/wide area networks.
- Analyze both current and historical data in real time.
- Implement system-specific decision-making models
- Report alerts to a range of users, both on- and off-site.
- Allow scaleable investigation of the prognoses/diagnoses and underlying data by a range of users, including potential simultaneous access by multiple distributed users assimilating the analyses in a collaborative network environment
- Reduce system development time through easy to use graphical interfaces

Status:

C-17 aircraft propulsion system health monitoring was chosen as a demonstration application. This project is funded through the Small Business Innovation Program. The Phase I effort (Contract No. NAS4-02005, Creare, Inc.) started in November, 2001. Projected completion of Phase II, if funded, is December 2003.

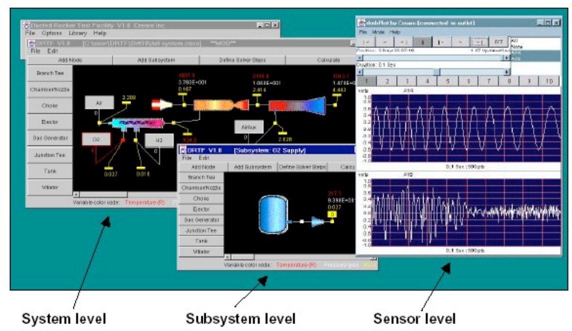

Fig. 1. Object Hierarchy in object-oriented network-based health management system

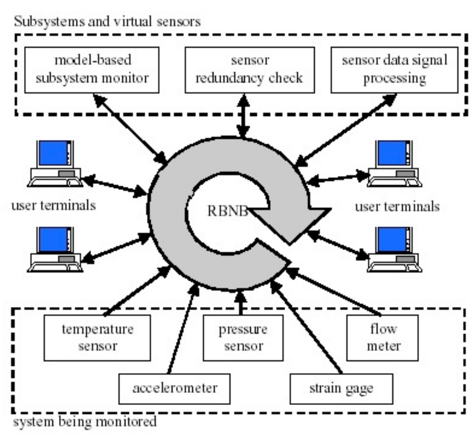

Fig. 2. High-level data flow to and from the RBNB caching network

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Ring Buffered Network Bus Server Development

Summary:
NASA Dryden Flight Research Center has been instrumental in conceiving, developing, and commercializing data caching and distribution middleware useful for high performance network distributed process automation, complex systems, data acquisition networks, and other peer-to-peer computing applications. The production version supports a variety of research and test activities. A second version in development addresses scalability limitations and extends capabilities well beyond the originally envisioned role in vehicle health monitoring.

Background:
While network protocols provide useful and cost effective transport of any data, additional services are needed to use these networks for realtime data integration and distribution to and from geographically distributed nodes. The Ring Buffered Network Bus (RBNB) is a *network caching service* conceived for developing network distributed signal processing applications and is useful for new approaches to automating online aircraft flight test data analysis. It is currently in use at NASA Dryden and elsewhere for test automation, content distribution, measurement networks, and other high performance network applications.

The current production version (V1.2) has an internal data structure that can be optimized either for performance or flexibility, but not both. A new, recursive data structure enables inherent adaptive performance optimization. A second-generation server is currently being implemented. This new architecture extends features either envisioned for the original architecture or discovered to be necessary after adapting the server to various applications. The new architecture enables powerful hierarchical peer-to-peer distributed computing environments for high performance and large-scale collaborative process automation.

Approach:
The elementary data structure in the new architecture is called an *Rmap* (RBNB Map) consisting of a name, a time range, data blocks, and child Rmaps. All other data structures are derived from this simple object. Features of the redesigned server and associated applications include the following:

Simple API. A simplified application programmer interface satisfies the needs of the majority of applications in just six commands. This simple API is available as a JavaBean, ActiveX, or C interface. A full "DeepCVT" API automates commonly needed functions for more sophisticated applications.

Routing Hierarchy. The designation of a server as a leaf, branch, or trunk provides users with the ability to create parent-child relationships amongst servers, improving scalability in large server networks. Peer-to-peer routing is available for branch and trunk servers.

Buffered Routing. A particularly challenging problem being addressed is to enable random access for remote users while simultaneously being bandwidth efficient. Bandwidth efficiency implies repeated requests for remote data are satisfied with locally cached data wherever possible. Optimum transport efficiency is achieved when only the requested data is sent across the network and is sent the minimum number of times. A design that accomplishes this while merging multiple distributed sources for clients with minimum latency requirements has been the focus here.

Virtual RMaps. The equivalent of a set of symbolic links or shortcuts, virtual Rmaps allow channel regrouping/encapsulation and a mechanism to remember and manage client requests.

Plugin Servlets. Application modules that plug into the server to provide value-added processing on data streams have evolved toward synergy with the Java Servlet concept, enabling if necessary configuration through web browsers. Existing applications are being rethought and in some cases re-implemented as servlets.

Metadata and Metadata Search. The ability to extend data channels with defined attributes and the ability to search for data channels with specific attributes has been designed. Extensible Markup Language (XML) can be used to describe metadata.

Web Integration. The desire to use the web browser as a client has resulted in exploitation of the RBNB servers as a web cache proxy server. The value of this application is to provide a user with accelerated response of Internet browsing, a browsable memory of visited sites, and opportunities for user control of web content. In addition, the ability to designate a filesystem folder through which a user can share information with other RBNB network users has been demonstrated.

Enterprise-Scale Services. Concerns over the current reliability of the Java virtual machine (with nondeterministic garbage collection) when used in large scale deployments and in applications requiring determinism guarantees are being addressed. At the network layer, the ability to route data natively via ATM was demonstrated. A benchmark scenario of one million samples per sec distributed over 5000 channels with 100 clients is used to characterize latencies induced by various garbage collection strategies and underlying operating system activities. Preliminary results with average latencies under 100msec using commodity-class computers indicate our objectives are achievable.

Status/Plans:
The RBNB solves certain integration problems for network application developers interested in serving engineering and science communities with dynamic, compute-intensive, or otherwise complex application integration needs. The production server is a stable and supported commercial product. The second generation server is available as a beta release. The RBNB Data Management System was awarded U.S. Patent No. 6,212,568 B1 on April 3, 2001. Current plans include pilot deployments within the center for solving several data-to-the desktop and workgroup collaboration problems. It continues to be used for measurement networks and knowledge engineering research. The production server remains in use at Dryden for supporting flight projects. Funding is sought for continued development of new capabilities.

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