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Flight Demonstration of X-33 Vehicle Health Management System Components on the F/A-18 Systems Research Aircraft

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FLIGHT DEMONSTRATION OF X-33 VEHICLE HEALTH MANAGEMENT SYSTEM COMPONENTS ON THE F/A-18 SYSTEMS RESEARCH AIRCRAFT

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Abstract

The X-33 reusable launch vehicle demonstrator has identified the need to implement a vehicle health monitoring system that can acquire data that monitors system health and performance. Sanders, a Lockheed Martin Company, has designed and developed a COTS-based open architecture system that implements a number of technologies that have not been previously used in a flight environment. NASA Dryden Flight Research Center and Sanders teamed to demonstrate that the distributed remote health nodes, fiber optic distributed strain sensor, and fiber distributed data interface communications components of the X-33 vehicle health management (VHM) system could be successfully integrated and flown on a NASA F-18 aircraft. This paper briefly describes components of X-33 VHM architecture flown at Dryden and summarizes the integration and flight demonstration of these X-33 VHM components. Finally, it presents early results from the integration and flight efforts.

Acronyms and Symbols

BIT	built-in test
COTS	commercial off-the-shelf
DAS	data acquisition system
DHS	distributed hydrogen sensor
DLL	design limit load
DSS	distributed strain system
DTS	distributed temperature sensor
FDDI	fiber distributed data interface
FTF	flight test fixture
GEN II	generation II architecture
HOB	health optical bus
IVHM	integrated vehicle health management
MHz	megahertz (million Hertz)
NASA	National Aeronautics and Space Administration
RHN	remote health node
RS-232	Recommended Standard 232 (serial interface, IEEE)
RS-422	Differential Serial Interface Standard
SRA	Systems Research Aircraft
STE	system test equipment
VHM	vehicle health management
VME	versa module eurocard
β	angle of sideslip
μm	micrometers (10^{-6} meters)

Introduction

Integrated vehicle health management (IVHM) is becoming an essential part of aircraft systems for commercial and military vehicles as well as for space applications, such as reusable launch vehicles, space stations, and satellites. The development of an IVHM involves the integration of new or existing sensor technology with acquisition and processing devices, implementation of prognostic algorithms, and dissemination of results to the correct user in a timely manner. The development of such systems can improve vehicle

safety, reduce maintenance costs, and improve vehicle readiness by identifying potential faults and failures, taking the proper corrective actions, and informing the responsible crew member of the health and condition of the vehicle.

The X-33 Advanced Technology Demonstrator program has taken a first step towards IVHM by implementing a Generation I vehicle health management (VHM) system that monitors and records all data on the X-33. This VHM system transmits critical X-33 data by means of a radio frequency downlink [1]. Sanders, a Lockheed Martin Company (Nashua, New Hampshire), was contracted to develop and integrate the VHM system on the X-33. This system consists of a commercial-off-the-shelf-based (COTS), distributed, open architecture, data acquisition system, with a ground-based download storage component. This design implemented a number of new concepts that have been demonstrated in the laboratory, but have not previously been used in critical flight applications. In an effort to reduce the risk to the X-33 program it was decided that high-risk components of the X-33 VHM system should be demonstrated in a flight environment. NASA Dryden and Sanders teamed to demonstrate this VHM system modified for installation on the NASA Dryden Flight Research Center F/A-18B Systems Research Aircraft (SRA).

This paper gives an overview of the X-33 VHM architecture, defines the VHM architecture that was integrated and flown on the SRA, and summarizes the VHM system integration efforts on the aircraft, with an overview of the results of the flight demonstration phase of the experiment. Finally, it summarizes the lessons learned from the experiment.

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Scope and Approach

There were three main purposes for conducting this experiment; to (1) provide a risk reduction effort for the X-33 program, to (2) demonstrate enabling technologies for future reusable launch vehicle efforts, and to (3) develop generic tools and platforms that could be used for future VHM research efforts.

The research goals for the X-33 risk reduction effort included demonstrating that a fiber distributed data interface (FDDI) architecture could be used to distribute data between the remote health nodes (RHN) and the VHM computer. Proving the accuracy and capabilities of the RHN was also important in providing risk reduction for the X-33 program. The final area of risk reduction that was performed involved the integration and application of a limited but representative X-33 VHM system architecture on the aircraft. Data collected using the VHM system could be compared to existing data that sensors had collected using a separate onboard data acquisition system (DAS).

Enabling technologies investigated during the experiment included implementations of fiber optic cable plants that were unique for this experiment. The enabling technologies were (1) a generation II (GEN II) multi-mode fiber optic cable plant, (2) a single-mode fiber optic cable plant design, and (3) a fiber optic demodulation scheme and distributed strain sensors. None of these had been tested in a flight environment.

The final objective of the flight experiment was the development of a generic platform for VHM research. The two areas where generic platforms were implemented during this experiment were the open architecture VHM system design, which allow COTS hardware to be directly integrated with the VHM system; and the development of a generic structural flight test fixture (FTF) which was designed to generate repeatable static loads in both a laboratory and flight environment that could be attached to a standard centerline pylon on a fighter class aircraft. Proper design of the fixture would allow rapid, repeatable comparisons of the distributed strain system (DSS) strain gage measurements with conventional foil strain in the laboratory as well as in flight.

X-33 System Architecture

Vehicle Architecture

The X-33 VHM architecture (fig. 1) acquires data from a wide variety of sensors and systems. The system monitors and records all data on the vehicle, including all the flight bus data, all vehicle health data (structural, mechanical and system built-in test (BIT) status), and all the flight test instrumentation sensor data. This system interfaces to a NASA Jet Propulsion Laboratory avionics flight experiment that communicates over a MIL-STD-1773 fiber optic interface [2].

The VHM system uses a distributed sensing architecture to collect data from conventional sensors. The data are concentrated using 50 RHNs to gather structural, mechanical and environmental sensor data. Each RHN communicates with the VHM computer over a fiber optic health optical bus (HOB). The VHM computers (1) acquire data from the HOB, (2) monitor and record all MIL-STD-1553B data [3], and (3) gather fiber optic distributed temperature, hydrogen, and strain sensor data from digital signal processors which reside in the VHM computer. Optical sensors are mounted directly on the X-33 propellant tanks, while the driving optics and processing are housed in the VHM computers.

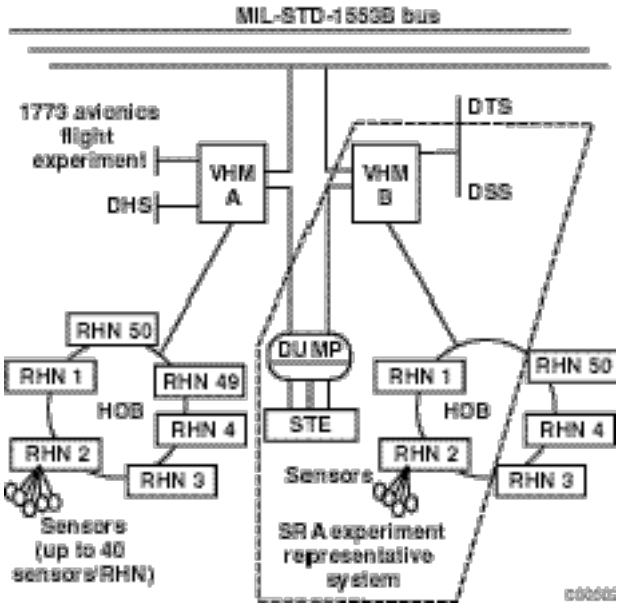

Figure 1. X-33/SRA configuration.

The X-33 vehicle health manager-A (VHM-A) shown in figure 2 is one of two nearly identical versa-module-eurocard (VME)-based computers housed in a modified full air transportation rack chassis used to monitor and record data. VHM-A contains a commercial off-the-shelf or military off-the-shelf standard processor interfaced to a 4.5 gigabyte PC-based hard disk drive storage module and various input/output modules. VHM-A connects to 25 RHNs by means of the fiber optic HOB implementing an FDDI. The distributed fiber optic temperature sensors and MIL-STD-1553B bus monitors interface directly to the VHM-A computer. The VHM-A computer acts as a remote terminal by using a MIL-STD-1553B interface for VHM command and control information. VHM-B is nearly identical to VHM-A; the differences are that the VHM-B line replaceable unit contains the distributed fiber optic strain and hydrogen sensors. The VHM computer also handles telemetry downlink, recorder download, and system diagnostic functions.

The RHN shown in figure 3 acts as a data concentrator for analog sensor information. Each RHN communicates with its respective VHM computer over two independent HOBs using FDDI as the network protocol. The FDDI uses a full duplex, dual counter-rotating token ring topology to provide reliable communications in the event of a failure.

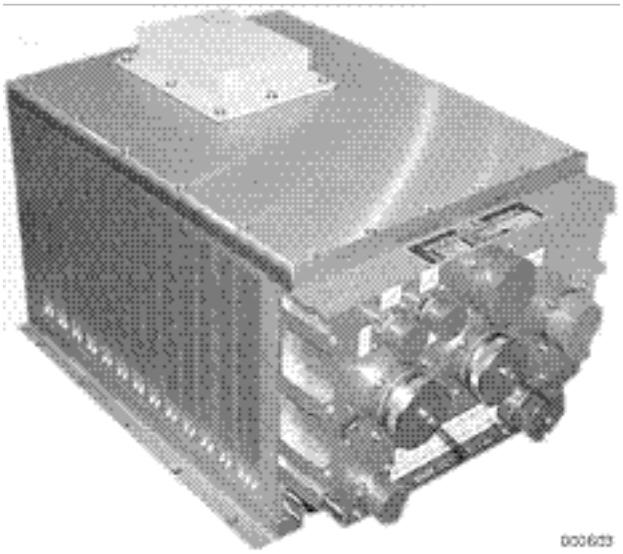

Figure 2. VHM-A/B.

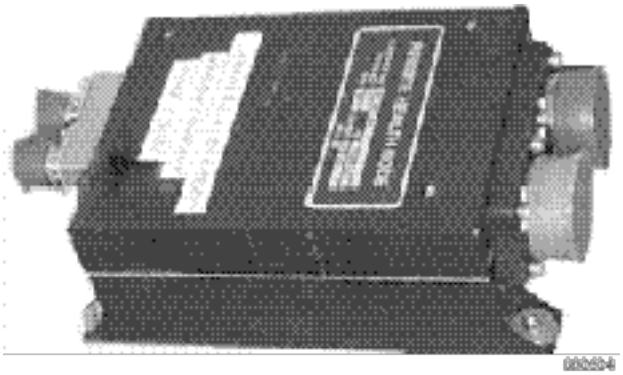

Figure 3. Figure 3 RHN.

Each individual RHN can collect data from up to 40 sensors located in close proximity of the RHN mounting location. This configuration reduces sensor wire weight and the chance of lost data that could result from electromagnetic interference and other outside phenomena. The RHN interfaces with several types of sensors including accelerometers, strain gages, thermocouples, resistive temperature devices, and sensors that measure pressure, rate, angular position, voltage, current, angular rate, and linear position. The RHN also provides the excitation to the sensor, if any is required. The sensor data are collected, conditioned, digitized, and time-tagged in the RHN. The data are sent to the associated VHM computer by means of the HOB to be recorded on a mass storage device in the VHM computer. After the vehicle lands, the data are downloaded by an optical connection from the VHM-A and VHM-B computers to the ground-based processing and storage facilities.

An optical network of distributed strain, temperature, and hydrogen sensors make up the fiber optic distributed sensor system on the X-33. The goal of this network of fiber sensors is to create a global strain, temperature, and hydrogen leak map for monitoring the health of the tank structure and cryogenic insulation. The fiber optic sensors employ two unique sensing techniques. The first technique used for strain and hydrogen leak detection utilizes a wavelength tunable narrow linewidth laser, a fiber optic network containing a sensing fiber with Bragg grating sensors, light detection photodiodes, signal conditioning electronics, and a digital signal processor. The second technique used a multi-mode fiber with a broadband laser source as a temperature sensor.

Distributed Strain System

The DSS is a fiber-optic-based sensing network that uses a NASA Langley patented demodulation technique to infer local information (strain, temperature, gas concentration, etc.) at multiple vehicle locations [4]. Sanders modified the Langley demodulation architecture [5]. Through X-33 and internal research and development funding the design further evolved over the course of the flight demonstration experiment. The DSS is designed to collect information from eight DSS fibers with up to 20 discrete fiber Bragg gratings multiplexed along a single fiber. Bragg gratings are localized, photo-induced, periodic perturbations of the refractive index in the core of single-mode optical fiber. These gratings have the unique characteristic of reflecting back a narrowband wavelength, called the Bragg wavelength, when illuminated by either a broadband light source, or in the case of the DSS, a tunable laser diode. The demodulation technique begins by first transforming the reflecting signals from each of the fiber Bragg gratings from the wavelength to the length domain by performing fast Fourier transforms. This yields the location of each fiber Bragg grating from the reference reflector along the sensing fiber. Each grating is then windowed, an inverse fast Fourier transform is performed, and the center wavelength of each grating is calculated during loading, and then subtracted from the initial wavelength at zero load [6].

SRA Experiment Architecture

SRA Aircraft Configuration

The SRA is a multi-purpose facility with the primary goals of identification and flight test of technologies beneficial to subsonic, supersonic, hypersonic, and space applications [7]. The aircraft is configured to fly multiple independent experiments during any given flight block, allowing for maximum utilization of vehicle assets and flight time by sharing flight time and costs for any given mission. Experiments flown with the VHM experiment included: a parameter identification experiment, a conformal ultrahigh frequency and very high frequency antenna experiment, and a flush air data system.

SRA VHM System Architecture

High-risk components of the X-33 VHM system were identified for the SRA flight demonstration. These components comprised a representative subset of the overall X-33 VHM system architecture (fig. 1). Slight modifications were made to the X-33 VHM components to either validate technologies that would eventually be fielded in a VHM system or that were necessary to interface into the F/A-18 SRA. A diagram of the SRA experiment configuration is shown in figure 4.

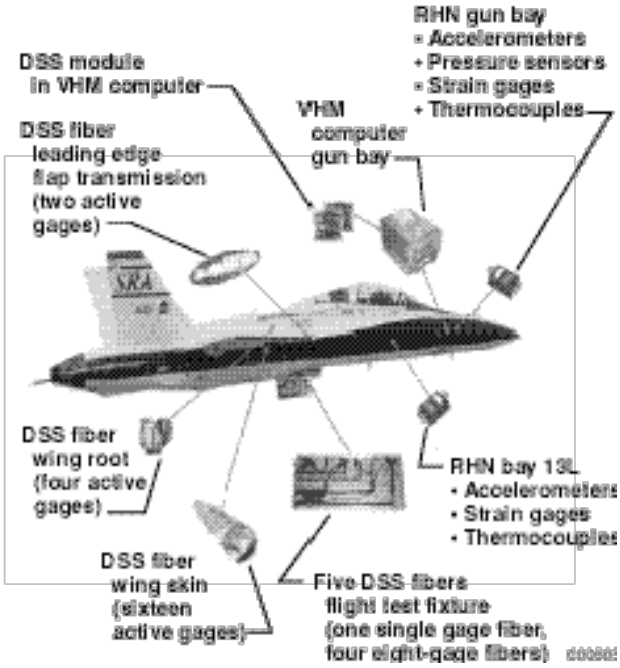

Figure 4. VHM experiment configuration.

Modifications to the SRA VHM experiment included: a single VHM computer with solid-state memory replacing the disk drive, a single FDDI module, a DSS module, and GEN II optical interconnects. Two RHNs were configured in a half-FDDI ring configuration.

HOB communication was accomplished using a prototype GEN II fiber optic cable plant consisting of a five-fiber, 62.5 μm multi-mode ribbon cable that had five fibers interwoven with the strength member and clear rubberized jacket and a multi-fiber termination on each end. The termination of the fiber could only be accomplished at the manufacturer facility. As a result, the multi-mode cable plant was delivered cut to length with termini and connectors on both ends as described above.

The ground-based system test equipment (STE) was based on a standard PC architecture. This equipment consists of a 166 MHz PC, a 2-gigabyte hard drive, an Ethernet module, and the FDDI module. The STE software was developed by Sanders using a PC-based operating system and C++.

Flight Test Fixture

The flight test fixture was designed to address three specific issues. First, eliminating errors resulting from composite-based materials used in the F-18 fuselage panels. A large percentage of F-18 wing and fuselage panels are made of quasi-isotropic graphite-epoxy composite material. The fixture was fabricated out of 2024 aluminum with the goal of separating sensor from substrate phenomenon during thermal and mechanical in-flight loading. If the aircraft composite structural components were relied upon to validate the fiber optic sensors, then significant strain measurement errors, such as apparent strain, would be much more difficult to eliminate. Second, it is difficult to fly repeatable steady-state maneuvers which load the primary aircraft structure sufficiently to generate levels of strain necessary to validate the DSS. The ability to generate well-controlled, highly loaded and repeatable maneuvers was important in achieving both research quality and flight safety goals. Finally, the development of a “loads” fixture provides a configuration that could be easily tested in both a flight environment and a laboratory environment.

The FTF provided an extremely useful configuration for verifying the accuracy of the DSS sensors. The FTF was designed to maximize the applied strains within a limited flight envelope, therefore, it was to be installed and flown within its designed flight envelope for a limited number of flights. Instrumentation attached to the FTF would be active when the FTF was installed on the aircraft.

Installation and Integration of the System on the SRA

The installation and integration of the VHM system on the aircraft posed a number of challenges. The entire system contained new components or configurations. This forced some

unique aircraft installations. Both X-33 flight hardware and the configuration of the SRA aircraft were modified to support this experiment.

VHM Computer

The modified chassis design for the VHM computer (fig. 2) resulted in a box that would not fit in a standard avionics rack. As a result the computer was installed vertically on an instrumentation rack located in the gun bay on the SRA. This installation subjected the VHM computer to "off axis" loading for a significant portion of the test program.

VHM computer modifications included incorporation of GEN II fiber optic connectors and a multi-channel single-mode fiber optic connector for the DSS. Software modifications were made to integrate the system onto the SRA. These modifications changed the computer operating commands and time synchronization from a MIL-STD-1553B interface to an analog switch to command the recorder and an RS-422 interface to provide time synchronization. Additional changes included; changing to a single FDDI module for both the RHN network and the recorder download, and the use of solid-state memory replacing the hard disk drive.

The original DSS laser module integrated into the VHM computer exhibited thermal stability problems. A redesigned optics module was developed and installed in the computer before the first dedicated DSS flights. Part of the redesign included software modifications that disabled automatic thermal protection logic in the DSS laser module. This resulted in a requirement for ground cooling to be supplied to the gun bay during aircraft ground operations to prevent the DSS laser from overheating while operating the system on the ground. Airflow through the gun bay was sufficient to cool the VHM computer during flight. The other major DSS software modification involved changing the digital signal processor sampling frequency in an attempt to improve the coherency of the sampled data. This modification effectively decreased the sampling frequency of the DSS module to approximately one sample every 90 seconds.

Remote Health Nodes and Fiber Optic Interconnects

The RHNs were distributed in two locations in the aircraft. These components were designed for installation in locations in the X-33 that are exposed to extreme temperature ranges and high vibro-acoustic levels [8]. Their compact size and large thermal envelope made the choice of RHN location dependent on sensor location, not the environment of the RHN module. The RHNs were modified slightly from the X-33 configuration. The major change to both RHNs was the change to GEN II optical interconnects. One RHN had the outer chassis changed to a composite chassis.

The main obstacle to integrating the RHN was connector placement on both ends of the RHN, as seen in figure 3. Fiber optic and power connectors were located on one end of each RHN. The sensor inputs on two connectors were located on the opposite end of the RHN. This design made it difficult to install and remove RHNs from the aircraft because it either required access to both ends of the RHN, or that RHN connectors be mated prior to mounting the RHN to the aircraft structure.

Routing of the HOB required that the backshell and part of the connector be removed from the GEN II cable plant and a routing and protection device supplied by the manufacturer be installed over the exposed multi-fiber termini. The cable plant was then routed as part of a larger cable harness using a convoluted tubing cable guide. A fiber optic termini was damaged during this routing process. Routing of the cable plant continued after the termini was damaged. The damaged termini and approximately 10 ft of excess ribbon cable were removed from the end of the cable plant and a new section of cable plant, with an undamaged termini, was mechanically spliced to the installed cable plant. Excess fiber optic cable was coiled into service loops and stowed near the RHN or the VHM computer. A second multi-fiber termini was damaged during routine connector mate and demate operations. The probable cause of the damage was the result of a misalignment between the connector and multi-fiber termini when it was being mated to the RHN. The connector with the damaged termini required that a "blind" mate of the cable plant to the RHN be accomplished. This damaged cable plant was removed from the aircraft and returned to the manufacturer for repair. The cable plant was

reinstalled in the aircraft upon its return from the manufacturer.

The single-mode fiber optic cable plant was routed from the VHM computer to the aircraft sensors and the FTF using convoluted tubing cable guides. The fiber was fusion-spliced after the fiber, sensors, and FTF were installed on the aircraft.

DSS Sensor Location

Conventional and fiber optic strain sensors needed to support the DSS experiment were routed to four separate aircraft locations: at the wing root, wing leading edge, wing skin, and FTF (fig. 4). Fiber optic sensors were installed at the wing root to compare with strain gages already existing as part of the conventional F-18 health monitoring system. Both fiber optic strain sensors and strain gages were installed on the outboard leading edge transmission to demonstrate that fiber could be applied on geometrically challenging aircraft components that were normally instrumented with strain gages. The underside of the right wing was instrumented with 16 fiber optic strain sensors spaced 1 ft apart on a single fiber. The fiber was routed from the inboard wing root area for 8 sensors in the outboard direction, and returned inboard with the remaining 8 sensors bonded at the same locations of the first 8. The goal for this configuration was to address surface attachment issues on realistic aircraft composite materials, as well as to establish what sort of precision colocated fiber optic sensors had when compared with each other at the same location. These three locations were permanently attached to the aircraft. The fourth location was the removable FTF mounted on the centerline pylon. The design and instrumentation philosophy of the fixture was described in the SRA Experiment Architecture section of this paper.

DSS Sensor Attachment

The fiber optic DSS sensors were bonded directly to the aircraft (and FTF) structure using methods and materials developed specifically for this experiment. Limited information was available in the literature for bonding fiber optic sensors. Procedures and techniques were developed based on strain gage technology and extensive flight and ground test experience. Lessons learned from the attachment of the fiber optic sensors for this program are being documented.

RHN Sensor Installation

Integration of analog sensors with the RHN was relatively straightforward. A combination of strain gages, accelerometers, thermocouples, and pressure sensors were integrated with the RHNs. Sensor selection was based primarily on using the same sensors specified in the X-33 master measurement list. However, in some cases, COTS transducers that were not specified in the X-33 master measurement list were used. The RHN signal conditioning, sample rate, and gain configurations were loaded into each RHN prior to the installation on the aircraft. These configurations were modified periodically during the flight program as needed for data quality purposes. There were no serious problems integrating the selected sensors with either RHN.

Once the sensors connected to the RHN bridge completion, their circuits were zeroed in order to remove bridge offsets from each channel. This was accomplished using ground system test equipment. Care had to be taken during this process because an invalid system command issued to the RHN would result in the loss of sensor load information.

Post Flight Data Management System

The final part of the VHM system to be integrated was the interface between the VHM computer and the ground STE. This interface was used after each flight to download data recorded on the VHM computer and to initialize the disk after the data were downloaded. The interface proved to be a challenge for the program. The FDDI recorder dump capability between the VHM computer and the ground STE worked well for most of the program. There was one incident of the system failing to work properly. This problem was later traced to a faulty processor in the VHM computer, not the FDDI link. Transfer of data between the VHM computer and the ground STE exhibited communications problems related to timing and data buffering. As a result of problems with the ground STE about 20 percent of the data recorded by the VHM computer was lost. This problem was traced to software problems with the ground STE. The flight program was completed before that problem could be fully identified and corrected.

Flight Experiment Phase

Flight Summary

The VHM experiment began flying on the SRA at the end of September 1999. Flight tests concluded in early April 2000. The experiment was installed on the aircraft for 53 flights and flew for over 46 hours. These missions covered a large portion of the SRA flight envelope. Figure 5 shows the flight conditions accomplished during the test program. Thirteen of the 53 flights were dedicated to testing the VHM system. Eleven of the thirteen dedicated VHM flights were used to characterize the DSS using the FTF. The remaining two dedicated flights were used to gather RHN data in regions of the aircraft flight envelope not covered during the nondedicated flights. The remaining 40 flights were dedicated to testing other experiments being flown on the aircraft. Data were collected during these 40 missions, but there were no test maneuvers specifically dedicated to the VHM experiment.

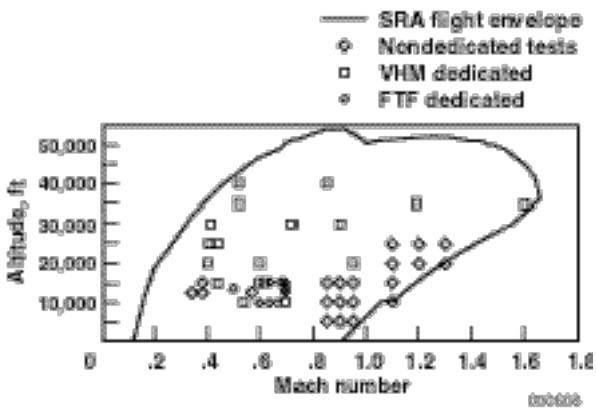

Figure 5. Flight envelope.

Two of the fifteen dedicated flights were used to gather RHN data in regions of the flight envelope that were not reached during nondedicated flight test, nor during FTF flights. Constant speed climbs and descents, and dynamic maneuvers were performed at eighteen flight conditions highlighted in figure 5. The aircraft was dynamically loaded to 6 *g* and subjected to high pitch- and roll-rate maneuvers during these flights. The aircraft reached altitudes of over 40,000 ft and speeds of Mach 1.6 during these test maneuvers.

Eleven of the dedicated flights were flown with the FTF installed on the aircraft. Of these eleven flights, seven were dedicated to validating the DSS sensor measurements. The other four flights were used to gather analog instrumentation data from the conventional strain gages being recorded by the RHNs and the NASA DAS. The FTF flights consisted of flying to a specific airspeed and altitude.

Flight Loading Approach—FTF Flights

The fixture was loaded by applying an angle of sideslip (β) to the fixture. The use of rudder trim and a fixed offset angle set at the FTF allowed β to be added to flight loads. This loading condition could be held until a representative data sample could be taken.

The FTF test points were flown using a build up approach starting from 10 percent of the DLL of the flight test fixture and continuing in an incremental fashion until 100 percent DLL was achieved. The percentage of DLL was determined prior to flight through a series of ground tests on the fixture. These tests were performed to gather a baseline strain gage survey based on loading conditions up to 115 percent of the DLL. These same strain gages were monitored in flight to determine the actual loading condition in real-time.

The FTF flights showed that repeatable loading conditions could be obtained consistently using the FTF offset angle and rudder trim. Both the DSS fibers and analog RHN strain gages were flight tested using the FTF configuration and flight approach.

Results and Discussion

The VHM system gathered data during the entire flight demonstration phase. The overall system performance suffered from a combination of faulty hardware, system integration, and system programming problems. The system performance is now discussed in detail.

The RHNs performed without failure during the entire flight program. The data correlated well with the NASA DAS but exhibited higher levels of noise on the individual channels. Figure 6 shows elevated noise levels for colocated thermocouples recorded using the VHM System and the NASA DAS. RHN-B data were lost for the first 21 flights

as a result of integration problems with the FDDI ring communications between RHN-A and RHN-B. This problem was tracked to a problem with the multi-mode cable plant between the two RHNs. The cable plant was repaired and the system performed well for the remaining 32 flights. RHN-A data were lost for nine flights because of a corrupt sensor system load in the RHN. The RHN continued to report valid BIT status to the VHM during these flights, but did not produce the requested data. This was traced to the BIT logic in the RHN which checks the health and status of the individual RHN channels, but does not check for valid system load.

This problem was repeated during ground operations when the sensor system loads for both RHNs were inadvertently corrupted during a calibration procedure. Both RHNs reported valid BIT status without valid sensor system loads being present.

Validation of the DSS experiment was only partially successful. There were a number of factors that contributed to this lack of success. The first problem was attributed to poor performance of the DSS laser and overall system. The DSS did not generate meaningful fiber optic strain data as a result of the high technological risk associated with fielding a flight-rugged tunable laser source. The experiment objective to validate fiber optic sensing technology in flight was, therefore, not achieved. Thermal stability issues with the laser also resulted in a very low sample rate (~0.01 Hz) during the flight experiment, which required that the test condition be maintained for a longer duration. This increased the total number of flights required to validate the DSS. The second factor that contributed to the lack of success in validating the DSS was a combination of a compressed flight schedule and intermittent operation of the DSS module. This

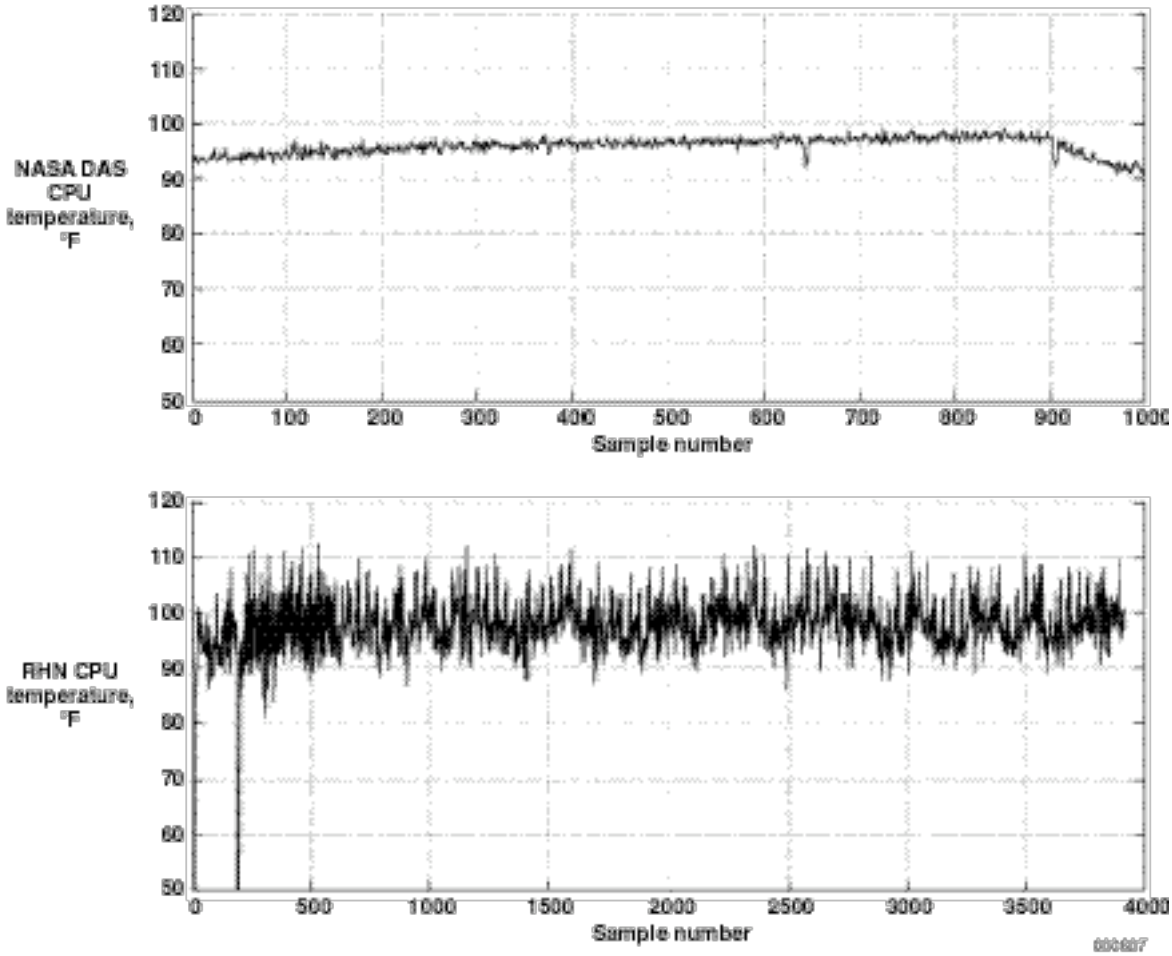

Figure 6. Flight data.

intermittent operation was not identified through analysis until flight testing of the DSS was complete. As a result, DSS data were only collected for 25–60 percent of the DLL cases.

The second series of four FTF flights were dedicated to validating analog sensor data. The DSS fibers mounted on the FTF were not connected during the flight phase. These four flights were dedicated to validating analog strain sensor data, after changes were made to sensor gains and bridge offsets in the RHN. The DSS was not connected, and the sample rates of the analog gages were much faster, therefore the duration of each test point could be reduced. Data were gathered for FTF loading cases up to 100 percent of the DLL.

The flight demonstration program succeeded in collecting over 25 gigabytes of data from 53 test missions. Data from these flights are being analyzed to fully assess the system performance and capabilities.

Conclusions

The basic VHM system architecture was well designed and implemented. Experiment integration and flight test showed the flexibility of the COTS-based open architecture design. A redesigned DSS module was integrated into the VHM computer after system integration began on the aircraft. The COTS-based VME boards, FDDI interface, and RHNs performed nominally after the initial integration problems were solved. Acquisition of analog sensor data was successful after the RHN channels were correctly configured with the correct system gains and bridge offsets. Data from the RHN channels show a tendency for levels of noise that are higher than noise levels of data recorded using the onboard DAS. Likely causes for this noise include differences in sample rates, gains and signal conditioning. As analysis of the flight data continues the accuracy and repeatability of individual sensor data will be more fully assessed.

There were some areas where design improvements would have aided the system integration and operations efforts. Hardware modifications that should be considered include implementation of a standard size air transportation rack chassis for the VHM computer and placement of RHN connectors at only one end of the chassis. Software issues that need to be addressed include modifications to the RHN BIT logic to test the validity of sensor system loads. A rework of the data transfer software in the ground STE is needed to avoid losing critical flight data during ground data transfers.

The integration of the fiber optic cable plants and DSS sensors on the aircraft presented their own challenges that included routing, splicing, termination, and attachment issues. The cable plants worked well after they were installed on the aircraft, demonstrating that both single- and multi-mode fiber can be used to gather and transmit information in a flight environment. Follow-on work is needed to address the installation and integration issues that can make the technology commercially feasible.

The DSS module did not meet expectations. Problems integrating the tunable laser into a flightworthy piece of hardware resulted in a system that performed intermittently. As a result, it was not possible to fully assess the performance of the DSS in a flight environment. Follow-on research is needed to develop a flightworthy tunable laser module that will meet the accuracy and sample rate requirements needed to implement structural health monitoring functions for future airframe and cryogenic tank applications.

Implementation and demonstration of the FTF showed that the fixture could be integrated with the VHM experiment. It could also produce predictable, stable, repeatable loading conditions in a flight environment. The FTF can be used in the future to aid in the development and demonstration of new structural sensors in a flight environment.

In general, the integration and flight program reduced the risk to the X-33 program through demonstrated system capabilities and identified areas where improvements or follow-on research are needed to meet system requirements. The program also demonstrated enabling technologies through the flight demonstration of the single- and multi-mode fiber optic cable plants. Finally it

developed tools, such as the FTF, that can be used for follow-on VHM research.

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The X-33 reusable launch vehicle demonstrator has identified the need to implement a vehicle health monitoring system that can acquire data that monitors system health and performance. Sanders, a Lockheed Martin Company, has designed and developed a COTS-based open architecture system that implements a number of technologies that have not been previously used in a flight environment. NASA Dryden Flight Research Center and Sanders teamed to demonstrate that the distributed remote health nodes, fiber optic distributed strain sensor, and fiber distributed data interface communications components of the X-33 vehicle health management (VHM) system could be successfully integrated and flown on a NASA F-18 aircraft. This paper briefly describes components of X-33 VHM architecture flown at Dryden and summarizes the integration and flight demonstration of these X-33 VHM components. Finally, it presents early results from the integration and flight efforts.

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