

FLIGHT EVALUATION OF THE INTEGRATED INERTIAL

SENSOR ASSEMBLY (IISA) ON A HELICOPTER

by

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ABSTRACT

After successful flight test evaluation of the Integrated Inertial Sensor Assembly (IISA) on an F-15 aircraft, the system was installed onboard a Blackhawk helicopter at Wilmington Airport in Delaware.

This paper provides an overview of the flight test evaluation conducted on the Blackhawk. All program objectives were successfully demonstrated and proved that the IISA system can be used for helicopter applications.

INTRODUCTION

Successful flight evaluation on an F-15 aircraft demonstrated and proved that the IISA concept - an integrated navigation and flight control reference system - is viable for fixed-wing aircraft. What remained to be proven was that the IISA concept is also feasible for a rotary-wing aircraft or helicopter. This feasibility was demonstrated when IISA was evaluated on a Blackhawk helicopter. This paper discusses the helicopter flight test methodology and results. For completeness sake, the IISA system is described first.

SYSTEM DESCRIPTION

The IISA system consists of five boxes: two Inertial Navigation Assemblies (INAs), two Digital Computer Assemblies (DCAs), and a Control Display Unit (CDU) [1]. Within each INA, sensor axes are orthogonal but skewed relative to the aircraft yaw axis. Figure 1 depicts the orientation of axes when the INAs are installed within the equipment bays of the aircraft. When one INA is installed into the right bay, with 180 degrees rotation about yaw relative to the identical left INA, the six sensor axes are uniformly distributed about a 54.7 degree half-angle cone. No two axes are coincident, nor are three in the same plane. Thus, any three sensors may be used to derive three-axis outputs in aircraft axes after suitable computer transformation.

An INA is divided into three, largely independent channels. Each channel consists of

one gyro and one accelerometer plus related sensor electronics, a flight control electronic pre-processor (FCEP), independent low and high voltage power supplies, and quad-redundant differential outputs. In addition, each INA contains a Navigation Processor with a MIL-STD-1553B I/O which receives power from one of the three sensor pair channels. The three channels are physically separated to eliminate common failure modes. Wiring from the sensors to the sensor electronics is also kept physically separated to avoid short circuit, EMI, etc..

Each INA is itself an Inertial Navigation System (INS), performing attitude and navigation computations. Navigation outputs are available on the INA's 1553B data bus. In addition each INA outputs data, in skewed sensor axes, directly from its FCEPs to the Flight Control Processors (FCPs) contained in the DCAs.

Redundancy Management (RM) of the sensors in IISA is illustrated in Figure 2. The RM operation has been programmed within each of the two FCPs contained in both DCAs. Each FCP receives angular rate and linear acceleration data in skewed sensor axes from all six FCEPs. Each FCP performs identical processing, thereby providing for quad-redundancy. The FCP first examines the built-in test information in the INA's flight control messages to disqualify any failed sensors. It then performs simple reasonableness tests on the INA flight control data. Any sensor channels which fail these tests are not used to determine the flight control outputs in aircraft body axes. Finally, the FCP compares the flight control data from the six INA channels, referenced to the same point in the aircraft, to determine which sensors to use. Fifteen equations provide outputs which represent the mismatch between the sensor data from all groups of four like sensors (i.e. gyros or accels). Each equation is formed by comparing a sensor's output to a linear combination of its output derived from three other like sensors. These equations are termed parity equations. Under ideal conditions, parity equation outputs should be zero under any aircraft dynamic or vibration condition. However, because sensors are in separate isolated units, shelf motion, isolator rocking and unit-to-unit misalignments can cause parity equation outputs (parity equation noise). For this reason, dynamic parity equation thresholds are used to prevent failure detection

logic from erroneously eliminating a healthy sensor. They are allowable noise levels which are functions of the aircraft's dynamic motion.

Once the RM software has determined which combination of inertial instruments to use, the acceleration measurements are then referenced to a common flight control reference point in the aircraft. Depending on the chosen sensor combination, the appropriate direction cosine matrix is used to transform the flight control outputs from sensor to aircraft body axes. Each FCP outputs this data at 80 Hz in a digital serial format.

The CDU is used for displaying IISA data and providing the operator interface for initialization, mode selection, insertion of simulated failures and the execution of simulated failure sequences.

IISA HELICOPTER DEMONSTRATION PROGRAM

IISA was installed onboard a U.S. Army Blackhawk helicopter (JUH-60A) stationed at Boeing Flight Test Center in Wilmington, Delaware. Flight tests were performed to determine the feasibility of using IISA within the high vibration environment of a rotary-wing aircraft. The objectives of this test and evaluation effort were to:

- (i) Verify the signal quality of IISA's outputs.
- (ii) Evaluate IISA's response to induced faults.
- (iii) Evaluate IISA's navigation performance.

The IISA Helicopter Demonstration Program was comprised of two phases. During the first phase, vibration data was obtained to quantify the helicopter's vibration environment. Based on this data, IISA's digital flight control filters were modified for flight control signal quality evaluations during the second phase [2].

Phase I: Identification of Vibration Environment

Hardware Installation. Two different installation configurations were used on the helicopter. One configuration utilized a specially designed equipment rack to which both INAs were mounted (Figure 3). The second configuration, which was considered more representative of a production installation, had both INAs mounted to the helicopter's floor (Figure 4).

The rigid equipment rack was designed to have a resonant frequency above 80 Hz. This was done for two reasons. First, each INA contains a set of vibration isolators that limit environmental

vibration sensed by the inertial instruments. This vibration isolation system has a low Q resonance mode. The resonant frequency of the vibration isolators is 37.5 Hz with a Q of 4. Therefore, any vibration frequencies near 37.5 Hz may have sufficient energy to excite this mode, and cause excessive rocking and coning motions of the sensor triad, resulting in possible sensor damage. Second, structural bending modes may affect the operation of IISA's Redundancy Management (RM) software. The parity equation residuals, which represent the mismatch between various linear combinations of inertial sensors, may exceed the dynamic threshold levels causing excessive sensor switching and erroneous sensor selection.

The second installation considered the effects of aircraft bending modes as well as ballistic protection requirements. The INAs were separated by more than six feet with each INA secured to the helo's floor via aircraft mounts.

During Phase I both configurations were used to collect vibration data in order to characterize the helicopter's environment. Identification of the vibration environment was carried out through the use of triaxis accelerometers that were mounted to each INA. These accelerometer triads were epoxied directly to the top of each box and referenced to the aircraft's body axes. The accelerometers, which normally have a bandwidth between 2 and 4000 hertz, were limited to 100 Hz by a low pass filter contained in the instrumentation package. The Instrumentation Data System (IDS) allowed the transmission of the accelerometer outputs to a ground station for real time display and post flight analysis.

Vibration Data/Filter Design. A standard flight scenario was developed consisting of dynamic maneuvers designed to provide worst case data. The flight regime was representative of the most commonly flown maneuvers in modern helicopters for both Scout/Attack and general purpose missions. The same scenarios were flown during both phases of the flight test program. Vibration data was collected throughout; Table 1 lists some of those conditions.

TABLE 1. IISA EVALUATION TASKS

TEST CONDITION	TEST REQUIREMENTS
o Ground Taxi	Taxi aircraft over a predetermined ground course to include straight-aways and turns into and away from the prevailing winds.
o Vertical Lift-off	Perform a vertical lift-off from the ground to an In the Ground Effect (IGE) hover maintaining X-Y position and heading within ± 5 ft. and $\pm 5^\circ$, respectively.

TABLE 1. IISA EVALUATION TASKS (cont'd)

TEST CONDITION	TEST REQUIREMENTS
o Vertical Landing	From a stabilized hover, descend vertically to a landing on a flat surface maintaining X-Y position and heading within ± 5 ft. and $\pm 5^\circ$, respectively.
o Precision Hover	Hover IGE and Out of the Ground Effect (OGE) while maintaining X-Y position within 5 ft. radius and altitude as appropriate.
o Pedal Turn	From a stabilized hover, acquire a preselected "target" that is 150° - 210° from the initial heading in a minimal time using the Pilot Handbook limits. Maintain X-Y position within 5 ft. and altitude as appropriate.
o Box Pattern	From straight and level flight, turn left or right 90° to acquire the new heading within $\pm 3^\circ$, pause momentarily, then continue to the next 90° heading and repeat. Altitude should remain constant. (AIRSPEED: 80 KIAS, BANK ANGLE: $\pm 45^\circ$).
o Roll Reversals	From a constant altitude stable 45° banked turn, roll the aircraft to a 45° banked turn in the opposite direction, pause momentarily and roll back again. Roll rate should be in excess of 30 deg/sec and the target attitude achieved within $\pm 5^\circ$. (AIRSPEED: 120 KIAS, BANK ANGLE: $\pm 45^\circ$).
o Pull-up/Pushover	From straight and level flight at 80 KIAS perform a 0.5 g pushover to 140 KIAS then a 2.0 g pullup to 80 KIAS. Repeat as required. Attempt to maintain constant g and smooth control inputs. (AIRSPEED: 80/140 KIAS BANK ANGLE: 0°).

Vibration data collected during this phase were in the form of real time stripchart recordings and spectral plots. The gyro parity equation residuals that were available from each FCP were also monitored and recorded. This was to determine whether excessive sensor switching existed and to measure body bending of the fuselage.

Analysis of the data from Phase I indicated that it was not necessary to mount the INAs on a stiff equipment rack. This was an important finding. Adding rigid structures to a helicopter is not desirable, since it can change the helicopter's dynamics and significantly add weight.

A comparison of the acceleration signals from both accel triads showed that low frequency body bending was insignificant at the INA floor locations. This was also made evident from an analysis of the recorded parity equation outputs. The magnitudes of the residuals indicated that IISA's redundancy management would not be susceptible to false alarms as a result of body bending effects. Additionally, concerns over possible sensor damage due to vibration isolator resonances were alleviated. The power spectral data indicated that there was not sufficient energy at each isolator's resonant frequency to cause excessive motion of the sensor triads during maneuvers. The structural vibration frequency levels were well under 0.2 g RMS.

Figures 5 and 6 are a representative sample of the power spectral data collected. Each plot was recorded during a particular flight maneuver. From this data it was determined that the helicopter's main rotor frequency of 17.2 Hz (4/rev) and its first harmonic of 34.4 Hz (8/rev) were the frequencies of concern. Without proper filters, the inertial instruments would pick-up vibration at these frequencies and corrupt the helicopter's Automatic Flight Control System (AFCS). Therefore, filters were designed to attenuate these frequencies. In addition, time delays in the Blackhawk's AFCS were also considered in the filter design (Figure 7). By emulating these delays in the filter, a more accurate comparison between IISA's flight control signals and the standard flight control sensor signals could be made.

Two signal conditioning filters were designed for IISA's inertial sensors. A notch filter, slightly lower than the 4/rev rotor frequency (16.0 Hz), was designed to have greater than 25 dB attenuation. In addition, a moving average filter was implemented to attenuate higher frequency harmonics of the 4/rev rotor frequency. This filter also emulated AFCS processing delays.

The transfer functions of the two filters are shown below. The notch filter was designed to be executed at a rate of 512 Hz.

$$H(z) = 1.09763 \frac{z^2 - 1.947840z + 0.986049}{z^2 - 1.801703z + 0.843642}$$

The moving average filter was designed to have an execution rate of 1024 Hz with a transfer function of:

$$H(z) = \frac{1}{50} \cdot \frac{(z^{50} - 1)}{(z^{50} - z^{49})} \cdot \frac{z^2}{2z^2 - 2z + 1} \cdot \frac{(z + 1)}{2z}$$

The combined response of both filters is shown in Figure 8. The combined filter produces over 25 dB attenuation at frequencies between 16 and 22 Hz, and over 15 dB at 34.4 Hz.

Phase II: Flight Test Evaluation

The second phase of the flight test program involved gathering data from IISA and the standard sensors onboard the helicopter. The IDS was again modified to allow the transmission of linear acceleration and angular rate data from both IISA and the standard sensors. Figure 9 illustrates the manner in which data was gathered via the IDS. A specially built digital-to-analog converter (D/A) was used to convert IISA's digital serial flight control outputs to analog DC voltages. It was also necessary to reference IISA's acceleration outputs to the same flight control reference point as that of the standard sensors, so that an accurate comparison could be made. The lever arm correction factors were measured between both INAs and the common reference point. These correction factors were then inserted into the IISA system software.

Flight Control Signal Quality. Examination of the stripchart recordings (Figure 10) indicates that IISA's flight control signal characteristics were better than the standard signals, because IISA's signals exhibited less noise content than the standard flight control signals. The IISA signals also led the standard signals by 60 to 100 msec, proving IISA has a faster response than the standard sensors. There was also no indication of any low frequency body bending effects on IISA's flight control outputs.

Fault Insertion Evaluation. The final flights of Phase II investigated IISA's response to induced sensor failures. The failures were induced through the Control Display Unit during the most dynamic part of a maneuver. Two types of simulated failures were chosen: hardover and ramp-bias. Hardover failures were simulated by adding a large magnitude fixed bias to a sensor's output (e.g. 255 deg/sec for gyros). Ramp-bias failures were simulated by adding a bias to a sensor's output which grows linearly during each

computational cycle (e.g. 0.5 deg/sec² for gyros).

The nominal parity equation thresholds were entered via the CDU. These thresholds, which are adjusted each computational cycle due to aircraft dynamics, produce temporary and permanent dynamic thresholds. These thresholds are utilized by the RM software to determine whether to fail a sensor temporarily or permanently. The determination of the nominal parity thresholds should be done experimentally, in order to limit sensor switching and system susceptibility to false alarms. For this portion of the flight test program, the nominal threshold were adjusted prior to each flight based on the behavior of IISA's flight control outputs during the last flight.

During hardover failure testing the system work flawlessly. The redundancy management successfully performed reasonableness tests to obtain a rapid determination of the failed sensor. The failed sensor was removed instantly, with no noticeable switching transients. However, the insertion of ramp-bias failures produced an oscillation in the angular rate and acceleration outputs for a short period immediately prior to permanently failing the bad instrument (Figure 11).

In order to determine the cause of this anomaly, the system was removed from the helicopter and brought back to the NAVAIRDEVCEEN's Strapdown Navigation Laboratory for further testing. The problem was duplicated in the laboratory (Figure 12) where it was discovered that one of the RM algorithms used for sensor selection was incorrectly implemented. The algorithm was corrected and then retested. Figure 13 shows the effect of the software change.

Navigation Evaluation. A thorough evaluation of IISA's navigation performance was beyond the scope of this flight test effort. However, during the flight control signal evaluation tests navigation data was collected by the CDU, which has the capability to store up to 10 checkpoints from each INA. During the course of flight, convenient checkpoints were always available.

The Radial Position Error Rate (RPER) of INA 1 and INA 2 was .77 nm/hr and .42 nm/hr, respectively (based on a least squares curve fit).

CONCLUSIONS/RECOMMENDATIONS

During the flight tests, no evidence of body bending modes or local structural vibration degrading IISA's performance was found. No evidence of either of these was noticed even during aggressive maneuvers. Nevertheless, these concerns should not be dismissed. Fuselage bending modes and local structural vibration can cause significant parity equation noise, leading to false alarms or permanent failures of good instruments. Body bending is even more pronounced

in a rotary-wing aircraft than a fixed-wing aircraft, because the structure is softer. Fuselage bending effects become even more significant as the separation between units is increased.

The inertial instruments inside each INA are isolated from the environment by elastomeric vibration isolators that exhibit a resonant frequency of 37.5 Hz. This resonance mode can be changed, should it become necessary for a particular application.

The data accumulated and analyzed during this flight test program indicate that IISA's flight control signals have the necessary dynamic range to satisfy future helicopter fly-by-wire flight control systems. As performance requirements of future rotorcraft become more demanding, a Ring Laser Gyro (RLG) system, like the IISA, can provide improved signal quality for flight control.

IISA's RM software worked flawlessly during the insertion of hardover failures. There was no indication of any switching transients or biases of the flight control signals. The anomaly encountered during the insertion of the ramp-bias failures was the result of a logic error which was corrected.

From the standpoint of flight safety IISA exhibits full QUAD redundancy. However, from the standpoint of ballistic survivability, IISA exhibits dual ballistic redundancy, because the sensors within each INA cannot be physically separated and dispersed throughout the aircraft. Appropriate armor plating and shielding measures may become necessary to protect the INAs from a direct catastrophic ballistic hit. It is recommended that prior to incorporating IISA into a combat helicopter, this issue be thoroughly reviewed.

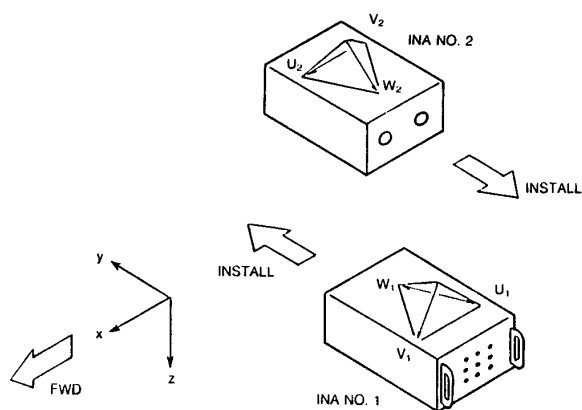


Figure 1. INA Installation Configuration

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1. "Final Report for the Integrated Inertial Sensor Assembly", Doc. No. 403895, Litton Guidance and Control Systems, November 1988.
2. "ADOCS - Integrated Inertial Sensor Assembly (IISA) Flight Demonstration/Evaluation Program", Doc. No. D 358-10055-4, Boeing Helicopters, 15 August 1989.

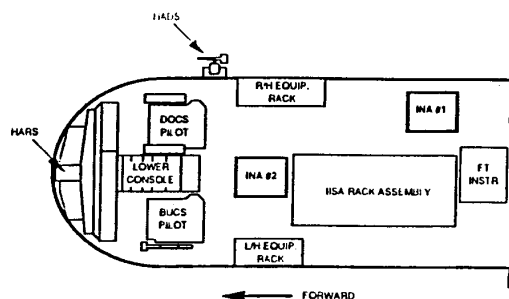


Figure 4. Installation of INAs on A/C Floor

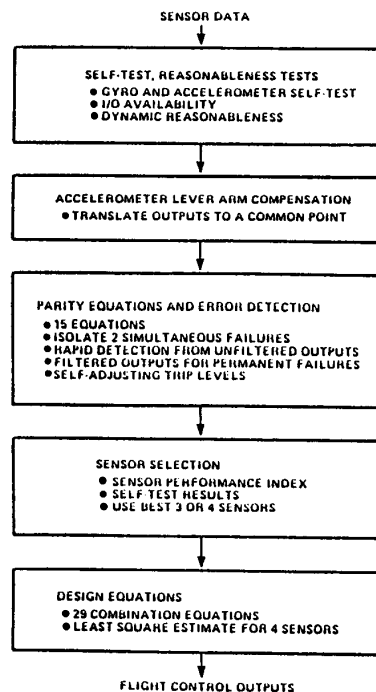


Figure 2. Redundancy Management Operation

Figure 3. Installation of IISA Components on Equipment Rack

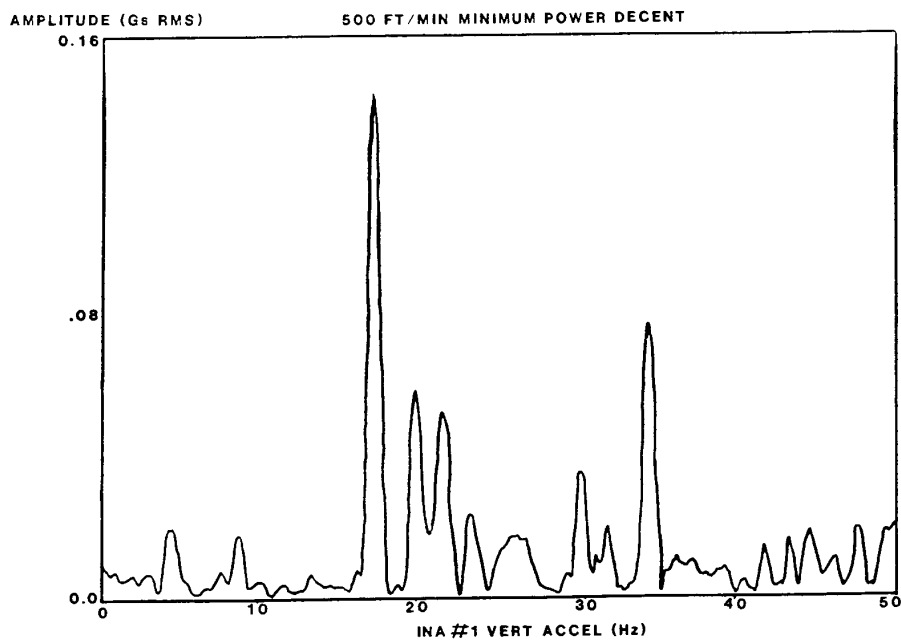
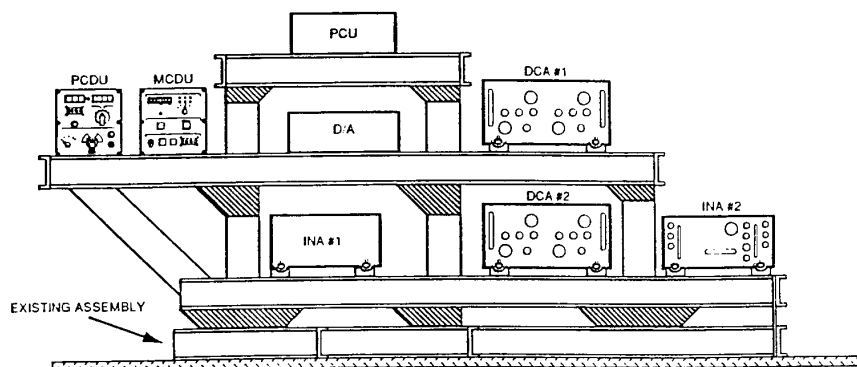


Figure 5. Vibration Power Spectral Data

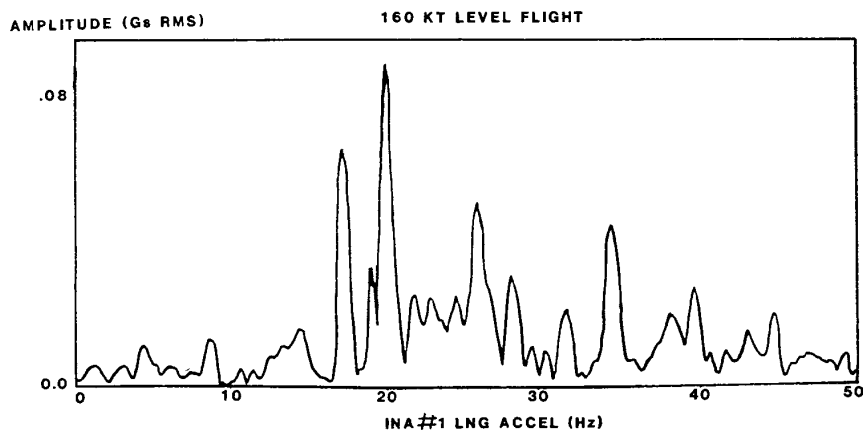


Figure 6. Vibration Power Spectral Data

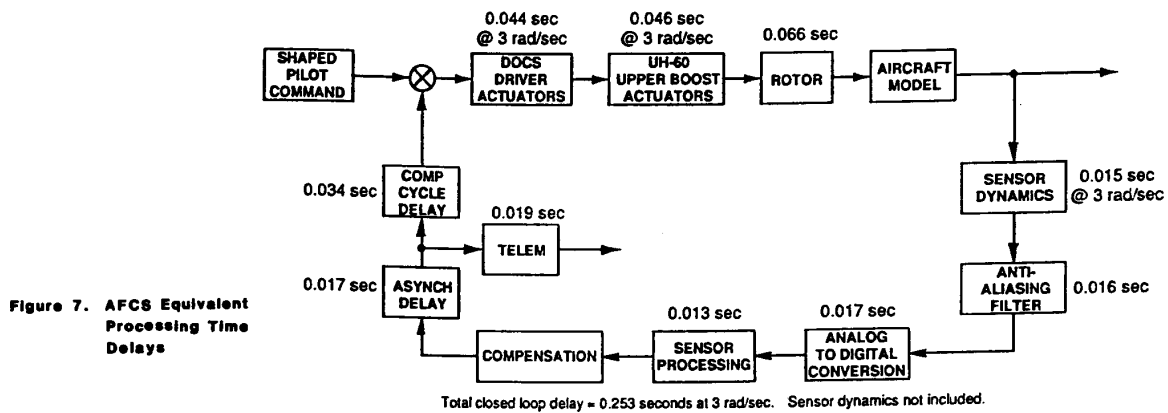
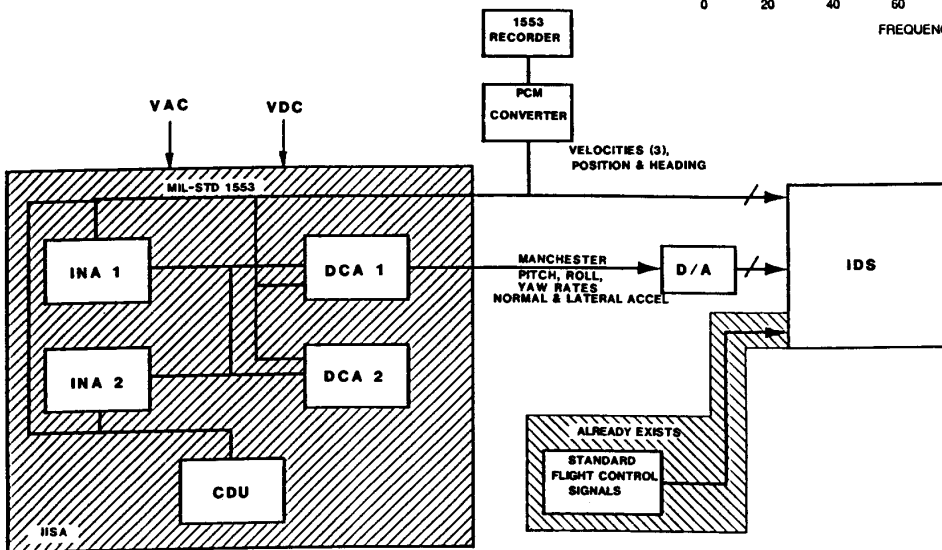
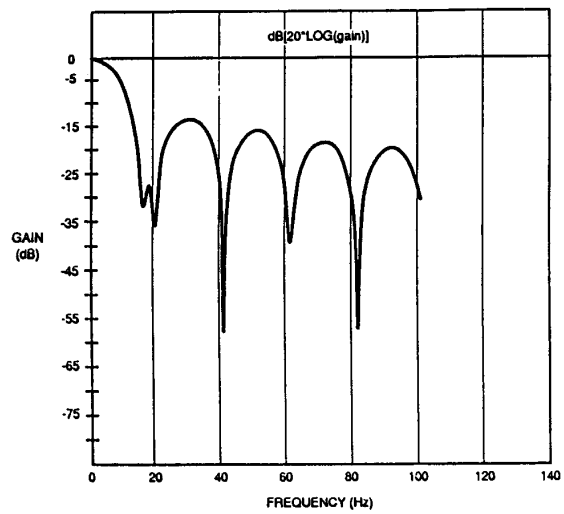


Figure 8. Combined Filter Magnitude Response



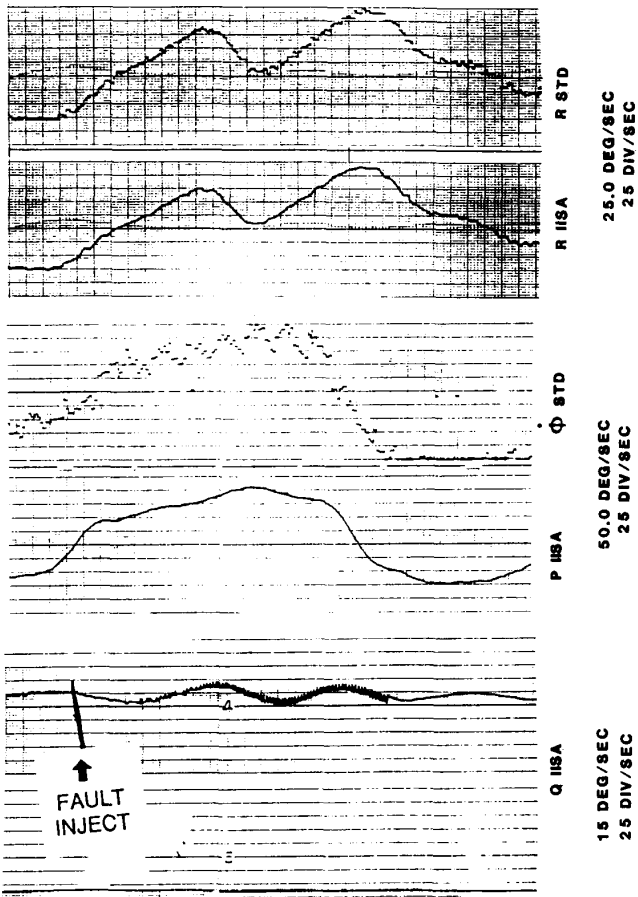


Figure 10. IISA/STD Sensor Response During Roll Reversals

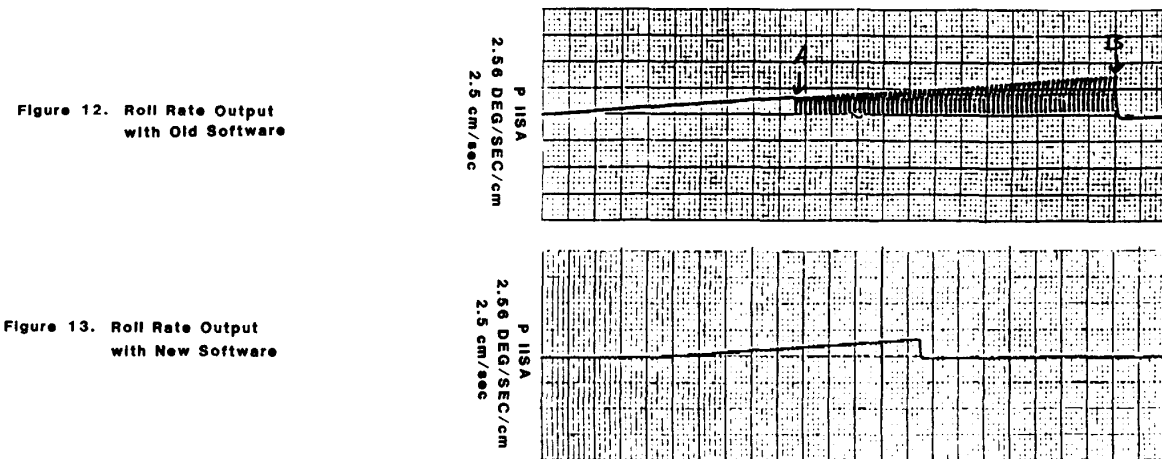


Figure 12. Roll Rate Output with Old Software

Figure 13. Roll Rate Output with New Software