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LOCKHEED AERONAUTICAL SYSTEMS COMPANY

LTN-72 INERTIAL NAVIGATION SYSTEM

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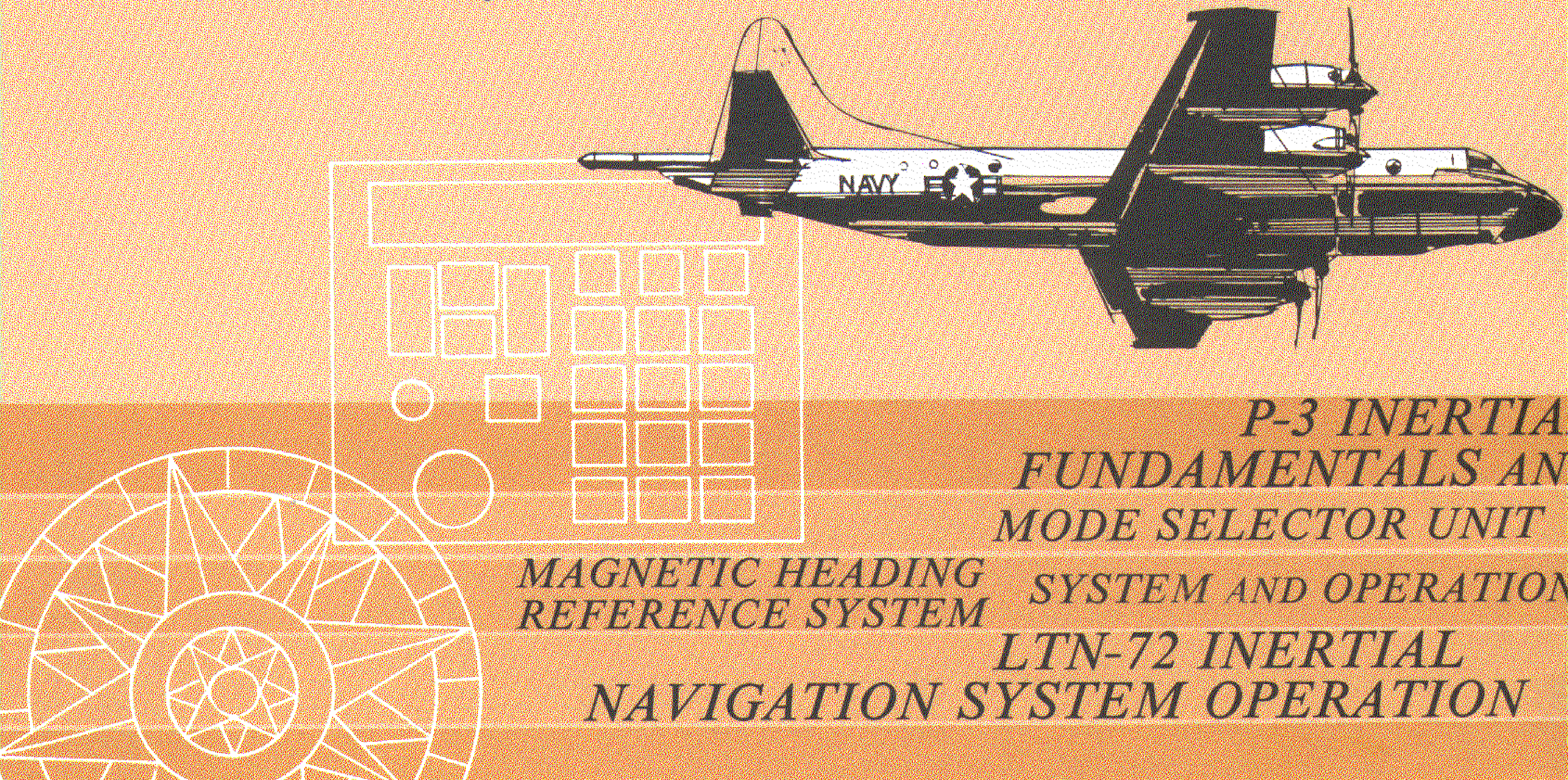
FRONT AND BACK COVERS

Patrol Squadron Ninety-Four was commissioned on 1 November 1970, in conjunction with the implementation of the Naval Air Reserve concept. Personnel were drawn from NAS New Orleans and NAS Dallas to man the newly-formed squadron. Since its formation, the squadron's home port has been NAS New Orleans, Alvin Callender Field, at Belle Chasse LA. Today, selected reserve personnel commute from Florida, Texas, Alabama, Mississippi, and Louisiana. Known as the "Crawfishers," VP-94's logo is featured on the back cover of this issue.

VP-94's mission is to train for mobilization. To maintain and enhance its operational ASW expertise, the squadron regularly flies operational and training missions in support of fleet operations in the Gulf of Mexico, Caribbean Sea, Mid-Atlantic, and in the Mediterranean Sea. Antisubmarine warfare, aerial mining, surface surveillance, and search annd rescue missions comprise the squadron's scope of operation.

From its commissioning through August 1976, Patrol Squadron Ninety-Four operated the Lockheed SP-2H Neptune aircraft. The squadron has since operated Lockheed P-3A Orions, and now flies the Lockheed P-3B TAC/NAV MOD aircraft.



The Crawfishers have performed Active Duty-for-Training (ACDUTRA) cruises at NAS Bermuda; NAF Lajes, Azores; NAS Barbers Point, HI; and NS Rota, Spain. During ACDUTRA in 1983, the squadron flew operational flights from five sites on the same day — NAS Keflavik, Iceland; NAF Lajes, Azores; NAS Bermuda; NS Rota, Spain; and NAS Sigonella, Sicily. The following year, the squadron made three separate ACDUTRA cruises to NAF Lajes.

In January 1986, VP-94 was a major participant in the U.S. Coast Guard's Drug Interdiction "Operation Hat Trick." For its outstanding performance, the squadron received the Coast Guard Meritorious Unit Commendation. During their 1986 ACDUTRA at NAF Lajes, Azores, the squadron logged more contact time with non-allied submarines than any other VP Reserve Squadron in history. Following their 1986 cruise, PATRON Ninety-Four received high marks in a unit NATOPS evaluation, and was commended by COMRESPATWINGLANT for their impressive standard of performance.

ACTDUTRA 1987 proved to be the most challenging to date. Transitioning to updated acoustic equipment only three months before their cruise, VP-94 performed flawlessly during ASW operations staged in the Mediterranean from NS Rota, Spain, and in the North Atlantic from NAF Lajes, Azores. VADM Moranville, Commander, Sixth Fleet, wrote to the Commanding Officer, VP-94: "Your total dedication and professionalism during MED operations contributed significantly to sustained readiness of the Sixth Fleet. I extend my personal apprciation, and commend your for the superb performance you displayed during your Mediterranean deployment."

The squadron is now at its highest level of readiness since it was commissioned in 1971. VP-94's Combat Aircrew 15 was selected by COMRESPACWINGLANT as Wing "ASW Crew-of-the-Quarter" for the first quarter of FY 1987. To date, the squadron has flown more than 58,000 accident-free hours.

Lockheed ORION Service Digest

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editor **WAYNE CRADDUCK**

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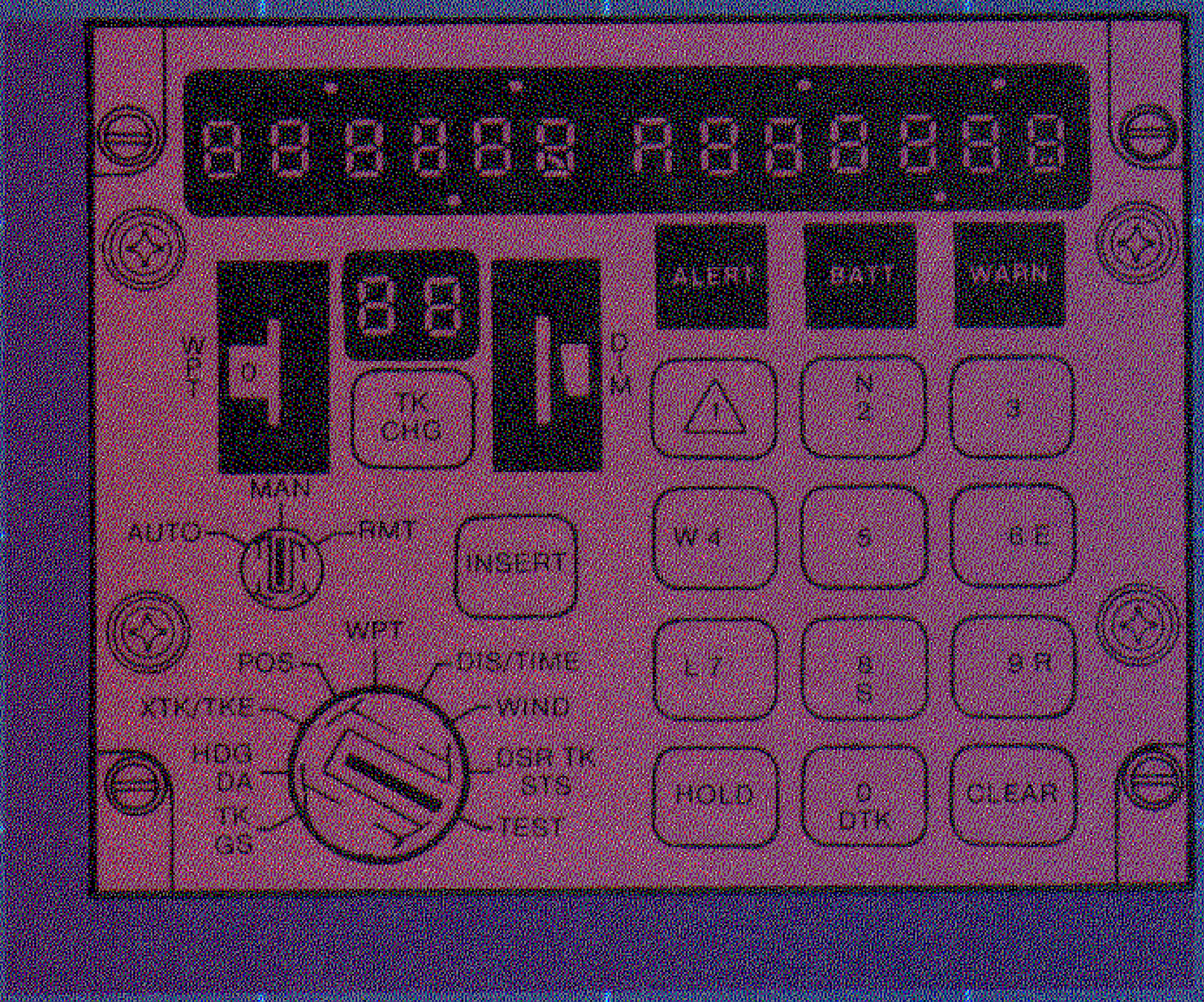
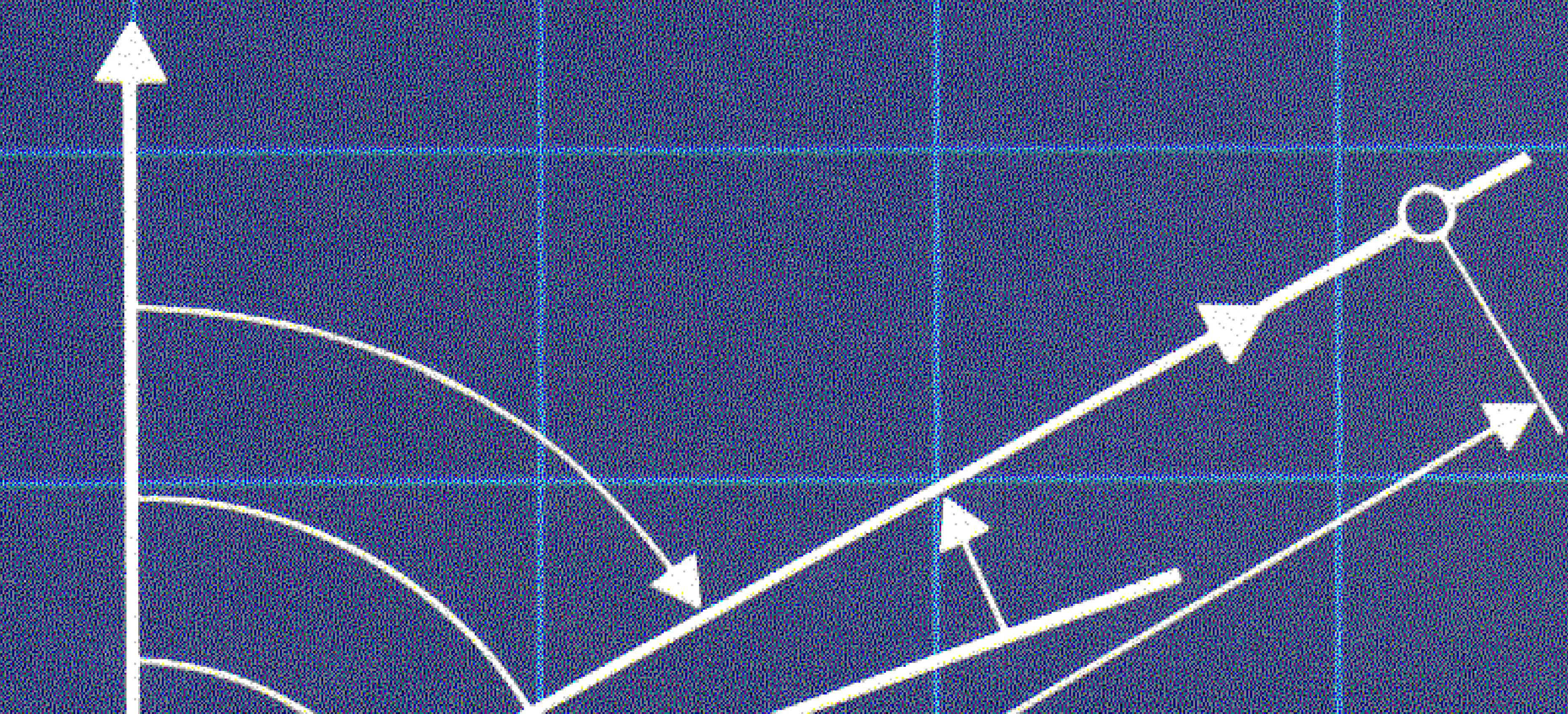
NAVIGATION DEVELOPMENT

NAVIGATION POWER ALARM

CP-901 COMPUTER INTERFACE

GEOGRAPHIC NAVIGATION





LTN-72 INERTIAL NAVIGATION SYSTEM

by Tom Blakely
Engineering Manager
U.S. Navy P-3

INTRODUCTION

The long-range patrol mission of the P-3 Orion aircraft imposes demanding requirements on the aircraft's navigation systems. Navigation accuracy, particularly during extended over-water flights, is critical to the safety and success of P-3 operations. The navigation systems enable the crew to determine the aircraft's present position, plus the direction, distance, and time to intermediate waypoints or to the final destination. Current-generation airborne navigation systems can consistently establish position with accuracy better than 0.5-nautical-mile-per-hour traveled. However, the navigator still must evaluate the various navigation solutions and make a judgment of the system position and steering accuracy. Modern navigation equipment has given the navigator more accurate sensors to aid in performing this task.

There are two basic classifications of navigation technique: dead reckoning and position-fixing. Dead-reckoning navigation measures vehicle speed and direction over a period of time to establish distance traveled and aircraft position. Self-contained sensors are used to determine speed and direction. This is one of the earliest forms of navigation and is the method employed by inertial navigation systems. In contrast to dead-reckoning systems, position-fix navigation relies primarily on transmission of an R.F. signal from an external navigation aid[1] to determine the aircraft's position.

The purpose of this article is to provide P-3 aircraft operators and maintenance technicians with a description of the interface and operation of the LTN-72 Inertial Navigation System (INS) as it is installed in the P-3C Orion. This system has been introduced to the fleet as an upgrade of the aircraft's long-range navigation equipment. The LTN-72 system is off-the-shelf commercial equipment that was selected based on proven performance and reliability in airline service. This article is presented in two parts. Part One discusses the development of long-range inertial navigation for the P-3 aircraft, and the basic theory of inertial navigation. Part Two provides a brief discussion of the interface between the mission avionics and the inertial navigation system, and describes the installation, interface and operation of the LTN-72 Inertial Navigation System in the P-3C Orion.

PART 1 — P-3 INERTIAL NAVIGATION FUNDAMENTALS AND DEVELOPMENT

INERTIAL NAVIGATION DEVELOPMENT

Since the introduction of the P-3 Orion, its inertial navigation system has been one of the crew's primary navigation aids. The P-3A and P-3B aircraft were equipped in production with the AN/ASN-42 Inertial Navigation System. This was a first-generation airborne inertial navigation system designed by Litton Systems Inc., that, with the APN-153 Doppler Radar, formed the core of the P-3A and P-3B navigation system. During the development of the P-3A in the late 1950s, airborne inertial navigation was in the early stages of development. At that time, most inertial navigation applications had been developed for submarines and ships. Both commercial and military avionics designers acknowledged the advantages of a self-contained navigation system that could dead-reckon aircraft position without reference to external navigation aids. The additional military advantages of operating an aircraft with its radar navigation systems secured prompted the U.S. Navy to incorporate this relatively new technology into the P-3A.

The ASN-42 Inertial Navigation System was an analog system that used electro-mechanical integrators to perform navigation computations. The system displayed the aircraft's present latitude and longitude to the navigator. In addition, it provided pitch, roll and heading angles to other avionics systems and to the flight station instruments. The system also provided north/south and east/west distance-traveled data to the crew sta-

[1]Celestial navigation is a type of position fixing that does not use R. F. transmissions.

TABLE I. P-3 Inertial Navigation System Development

TYPE	ASN-42 SEMI-ANALYTIC	ASN-84 SEMI-ANALYTIC	LTN-72 SEMI-ANALYTIC
HEADING REFERENCE	NORTH POINTING	NORTH POINTING AT ALIGNMENT – UNIPOLAR DURING NAVIGATE	WANDER AZIMUTH
COMPUTER	ANALOG	DIGITAL	DIGITAL
NUMBER OF WRAS PER SYSTEM	10	6	6
MAGNETIC HEADING COUPLING	INTEGRAL FUNCTION (SDC)	INTEGRAL FUNCTION (GYROSCOPE ASSY CONTROL)	MAGNETIC HEADING REFERENCE SYSTEM (SPERRY COMPASS COUPLER)
BATTERY OPERATION	10 SECONDS (VIA AIRCRAFT BATTERY)	2 SECONDS	15 TO 30 MINUTES
DOPPLER AIDED ACCURACY	YES	YES	NO
MTBF (APPROX)	39 HOURS	83 HOURS	1000 HOURS
MANUFACTURER	LITTON SYSTEMS	SINGER KEARFOTT	LITTON AERO PRODUCTS
WAYPOINTS	0	0	9
ALIGNMENT TIME (AVG)	20 - 30 MINUTES	25 MINUTES	17 MINUTES
SPECIFICATION	ER 421648	MIL-N-81497A-2	ARINC 561

tion navigation displays. Due to the complexity of this first-generation system and the lack of experience with airborne inertial navigation equipment, P-3A and P-3B aircraft were equipped with a single ASN-42 system installation. This installation was backed-up by an ASN-50 Attitude Heading Reference System to provide aircraft pitch and roll attitude and heading signals, and with the ASA-47 Doppler/Air Mass Computer to calculate a dead-reckoned position. The characteristics of the ASN-42 and subsequent P-3 inertial navigation systems are compared in Table I.

P-3C INERTIAL NAVIGATION SYSTEMS

Experience with inertial navigation in P-3A and P-3B aircraft demonstrated that it could be used as an effective independent system for long-range patrol navigation. As a result, the P-3C Orion was developed with *two* inertial navigation systems. The second inertial system was installed in lieu of the Attitude Heading Reference System. This second system is a backup source for true heading and velocity (for the mission avionics), aircraft position, and for analog attitude and heading reference. The P-3C's inertial navigation systems calculate and display aircraft position independent of its other navigation systems. The aircraft's CP-901/ASQ-114 Tactical Computer calculates the aircraft's dead-reckoned position based on velocity, distance and heading inputs from various on-board navigation sensors, including the inertial navigation system.[2] This provides the crew with *three* calculations of dead-reckoned position, which results in a greater probability of mission success and increased mission safety. The CP-901/ASQ-114 Computer operates as the central computer for the P-3C tactical avionics.

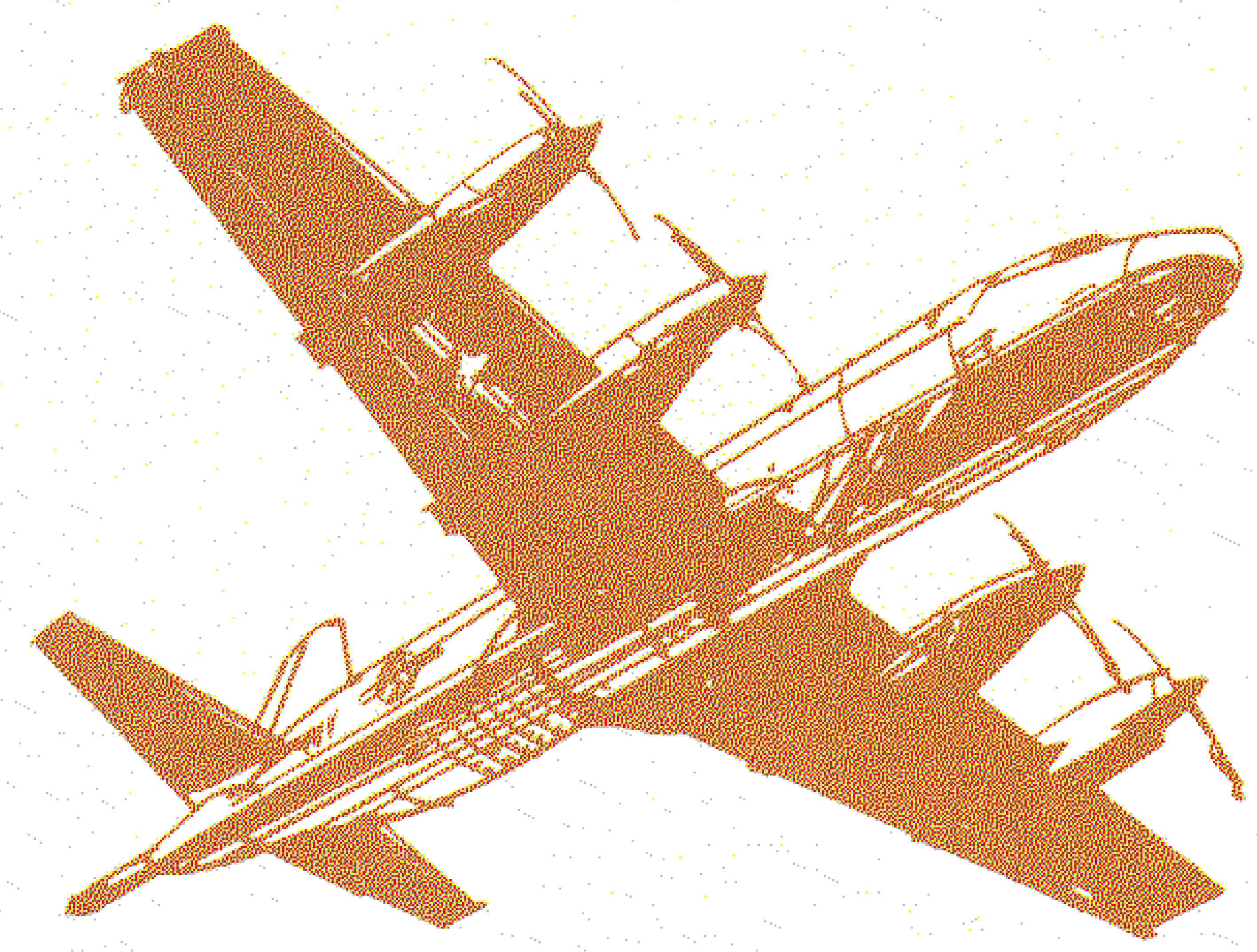
Production P-3C aircraft were initially equipped with two AN/ASN-84 Inertial Navigation Systems. The AN/ASN-84 was a second-generation system produced by the Kearfott Division of the Singer Company. This system provided im-

[2]The CP-901/ASQ114 Computer operates as the central computer for the P-3C tactical avionics.

proved accuracy and reliability over the AN/ASN-42 system. The AN/ASN-84 system also provided the digital north/south and east/west velocity and true heading data required for interface with the CP-901/ASQ-114 Tactical Computer. The AN/ASN-84 system used a digital computer to perform these navigation calculations in lieu of the electro-mechanical integrators that were used by the earlier AN/ASN-42 system. The AN/ASN-84 system was installed in P-3C production aircraft through BUNO 161004, except BUNO 161001.

The in-service reliability expectations for the AN/ASN-84 Inertial Navigation System were never realized. In early 1977, the U.S. Navy decided to incorporate in P-3C production aircraft a third-generation commercial inertial navigation system that had exhibited proven performance and reliability. Several manufacturers had inertial navigation systems, then in current use with airlines, that were compatible with ARINC[3] Characteristic 561 for Air Transport Inertial Navigation Systems.

The basic ARINC 561 Characteristic was approved in December 1966. This characteristic established a standard configuration for design parameters such as connector pin-out, cooling air, and power requirements. Thus, the inertial navigation systems that meet this characteristic are, to a certain degree, interchangeable. The design requirements for the installation of a commercial-type inertial navigation system in the P-3C aircraft were established to ensure interchangeability between the two ARINC 561-type inertial navigation systems that are in the DOD inventory. These two inertial navigation systems are the LTN-72/LTN-51 series system manufactured by



Litton Aero Products, and the Carousel IV system manufactured by Delco. There are minor differences in the operation of these two systems, but the P-3C installation (with some relatively minor modifications) will accommodate either one. The LTN-72/LTN-51 system is used in the A-3, C-130, and P-3B aircraft. Consequently, the Navy elected to specify the Litton system for the P-3C aircraft.[4]

The LTN-72 Inertial Navigation System is a semi-analytic gimbaled-platform system. It uses an internal digital computer to perform the navigation calculations and to control system operation. Additional features include automatic self-calibration of gyro bias, and a comprehensive program for alignment, malfunction detection and fault isolation. One design premise for the installation of the LTN-72 system in P-3C aircraft was to minimize its impact on the aircraft's existing avionics systems. This required some additional equipment to: (1) couple the magnetic heading to the aircraft compass system, (2) reformat the ARINC 561 digital data output to P-3C digital system format, and (3) condition the LTN-72 system's validity logic to function with the existing P-3C systems.

This article is a detailed description of the LTN-72 Inertial Navigation System installation in the P-3C aircraft. A brief review of the fundamentals of inertial navigation has been included to aid the reader in understanding how the LTN-72 system operates. We have also included descriptions of LTN-72 system functions that may be of special interest to the operator or maintenance technician.

[3]ARINC is an acronym that refers to Aeronautical Radio, Inc. This corporation aids the commercial air transportation industry by coordinating development of the design standards for aeronautical communication systems. ARINC also sponsors the Airlines Electronic Engineering Committee (AEEC), which develops standards for air transport electronics systems. The scheduled airlines are ARINC's principal stockholders.

[4]The U.S. Air Force uses the Delco Carousel IV Inertial Navigation System in C-5 and C-141 aircraft.

Airborne inertial navigation systems are dead-reckoning systems that measure the motion of the aircraft relative to the earth's gravitational field. Other types of airborne dead-reckoning navigation systems measure aircraft movement relative to a moving air-mass. Although the techniques for dead reckoning have been used and refined for many years, an inherent error can be introduced when this measurement is converted to earth-reference (geographical) coordinates. The advantage of inertial navigation is that the measurement is made relative to the earth's coordinates. This eliminates the process of converting the aircraft motion measured in one reference frame to the distance-traveled in another.

FUNDAMENTAL SYSTEM COMPONENTS All inertial navigation systems require the following fundamental components:

- Accelerometer — to measure accelerations of the vehicle.
- Stable Platform — the device that the accelerometers are mounted on.
- Gyroscopes — used to maintain the platform in a stable position.
- Computer — (a) to calculate velocity and position based on measured accelerations; (b) to develop correction signals to bias the gyroscopes in order to maintain the desired orientation and control of the stable platform; and (c) to provide corrections to accelerometer outputs.

Specific inertial navigation systems have additional hardware requirements that vary, depending on the application of the particular system. At this point, let us briefly review the function and operation of the fundamental inertial navigation system components.

Accelerometers The basic measurement made by an inertial navigation system is vehicle acceleration. The accelerometer detects movement by employing the basic physics of forces that result from the acceleration of a mass or weight in a gravitational field. It should be kept in mind that acceleration is a change (increase or decrease) of velocity. If the accelerometer's sensitive axis is level, the device should produce no output when the aircraft is stationary or when it is traveling at a constant velocity (no acceleration). The operating principles of an accelerometer are depicted in Figure 1. As the vehicle accelerates, the inertia of the mass resists the change in motion in proportion to the rate of acceleration.

There are several different approaches to mechanization for an accelerometer, depending on the

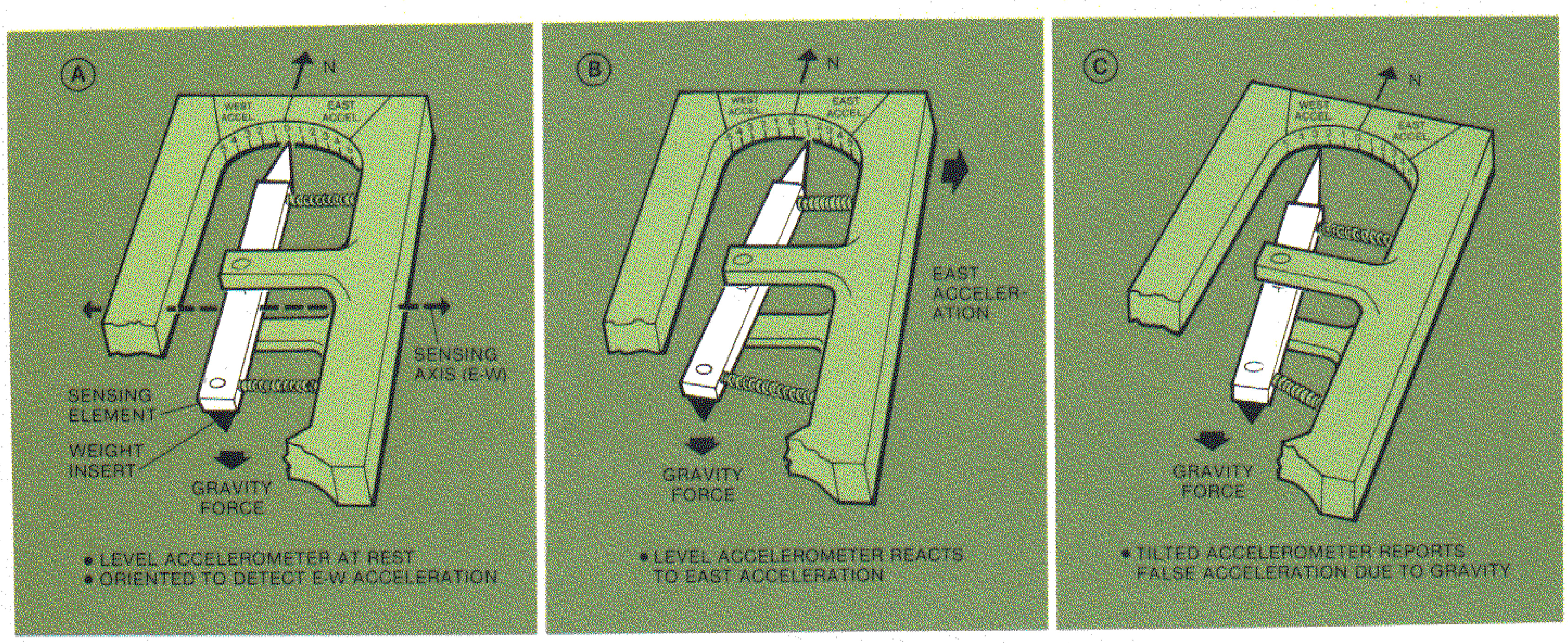


Figure 1. Operating Principles of a Simple Accelerometer

sensitivity, linearity and accuracy requirements. Inertial system-grade accelerometers must be designed to perform the most demanding and complex of these requirements. Other applications of accelerometers (automatic flight controls, flight loads counting, etc.) are less demanding in accuracy, and can be achieved with simpler mechanization and less costly fabrication techniques. The accelerometers in the LTN-72 system are mechanized as torque-rebalance units as shown in Figure 2. This means that during acceleration, the accelerometer circuitry applies electrical restoring torque to the unit's mass to maintain it in a centered or non-displaced position. The electrical current required to restrain the mass is proportional to the acceleration.

Most inertial navigation systems use an arrangement of three accelerometers, one to sense motion in each axis (longitudinal, lateral, and vertical). Figure 3 shows the arrangement of accelerometers on a stable inertial platform. Several different approaches have been used to orient the accelerometer array relative to the earth reference coordinates. The basic requirement is to mechanically drive the platform to orient the accelerometer array while maintaining the lowest possible rotation rate of the inertial platform about its vertical (azimuth) axis.

Some inertial navigation systems have accelerometers that measure acceleration in the north/south, east/west, and vertical directions. The inertial platform is then mechanized to maintain this directional orientation as the aircraft transits its flight path and/or maneuvers. An inertial system of this type is defined as a "north pointing" system.

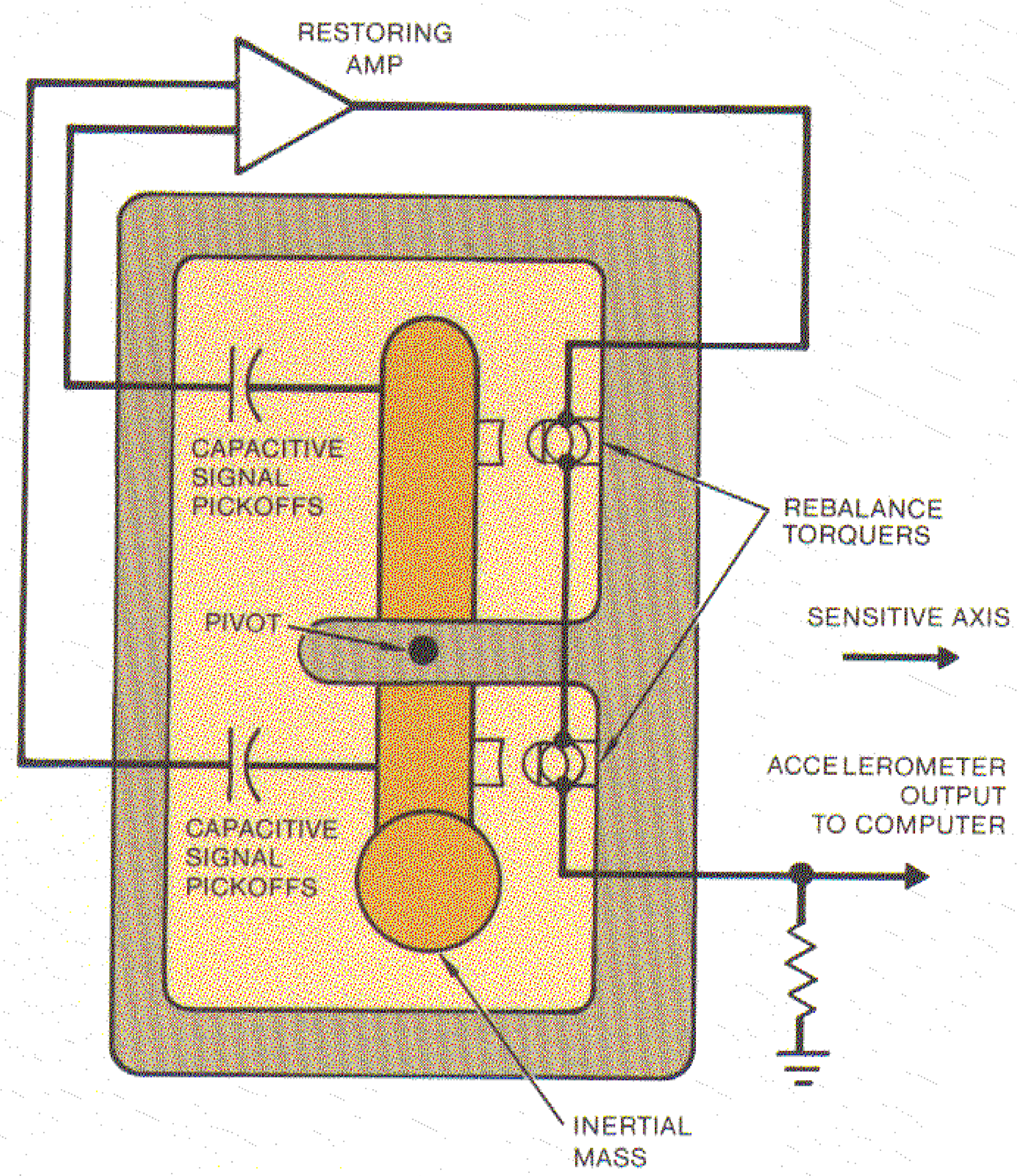


Figure 2. Torque-Rebalancing Accelerometer

Another type of inertial navigation system has its accelerometers mounted in a similar configuration (with the sensitive axes at 90 degrees to each

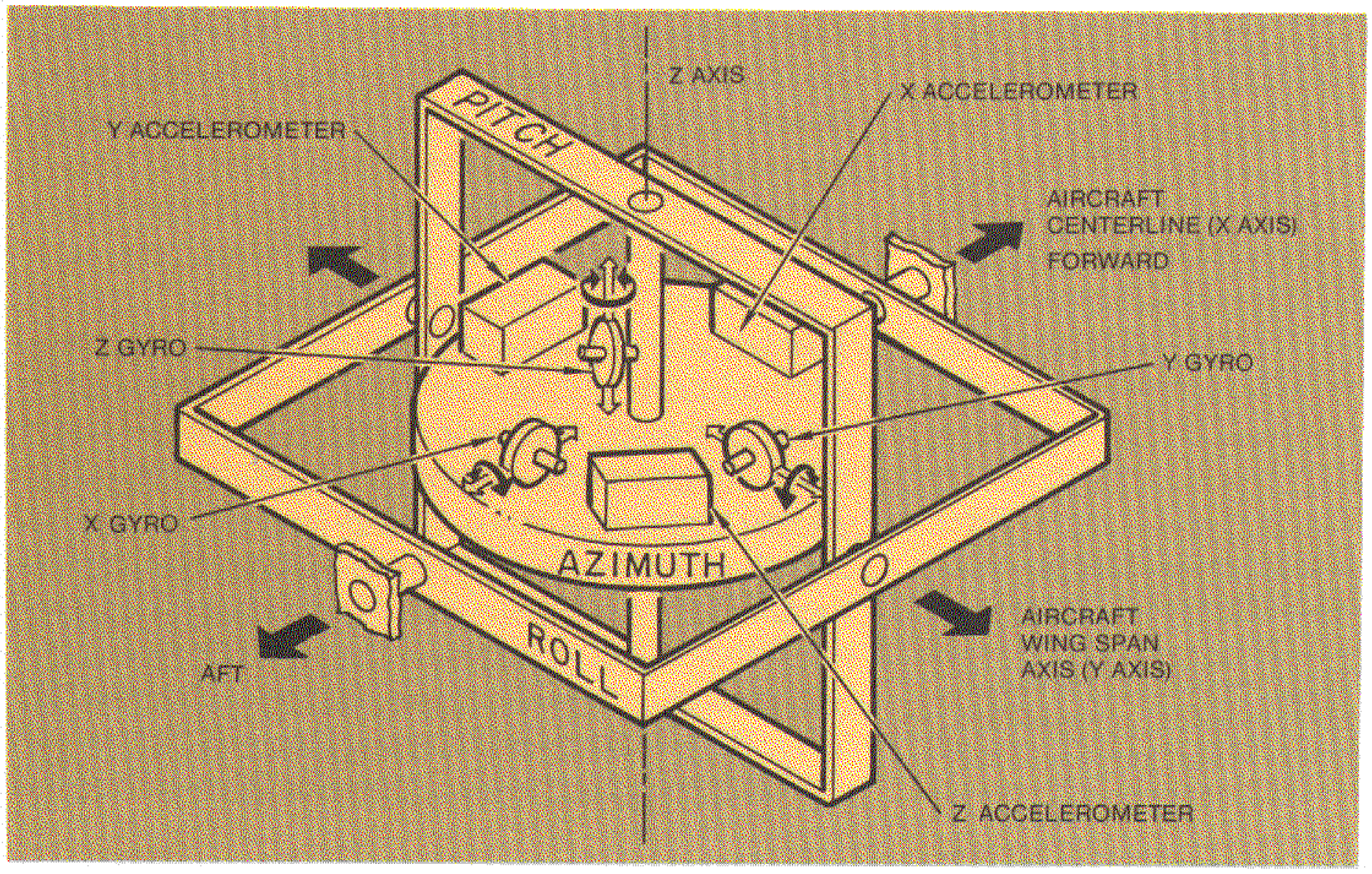


Figure 3. Stable Inertial Platform

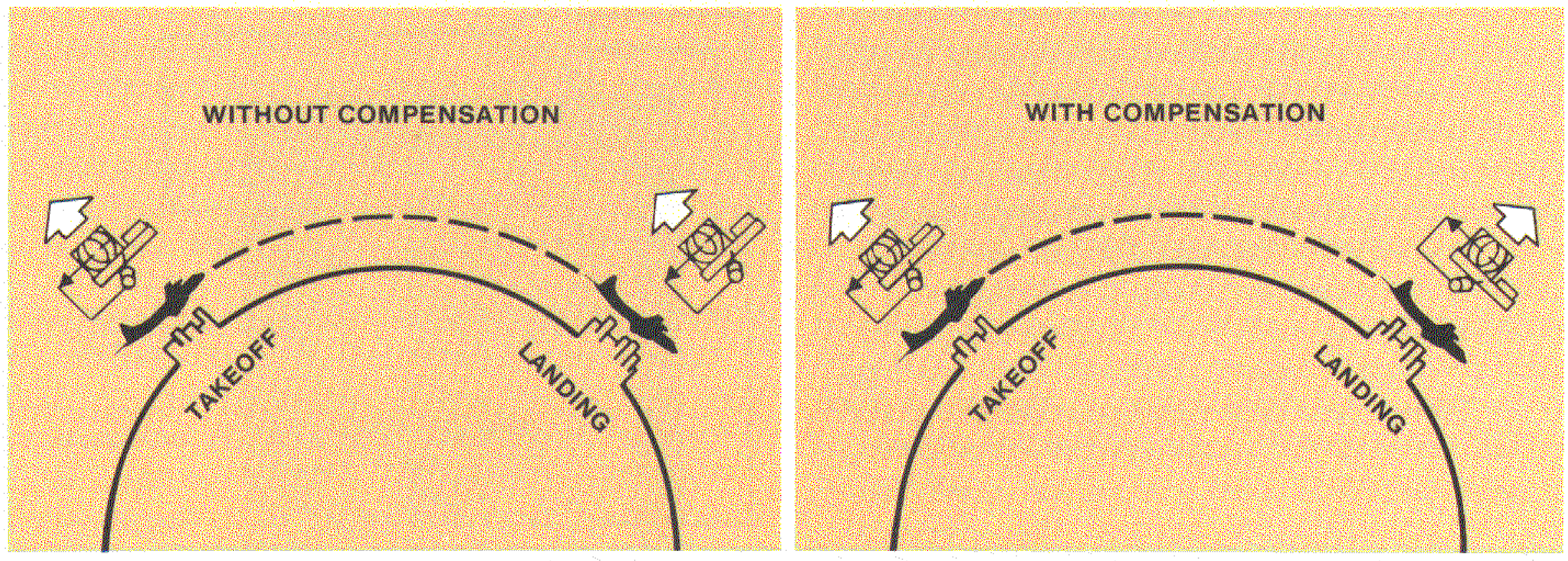


Figure 4. Inertial Platform Compensation

other), but the inertial platform is allowed to assume an arbitrary but defined angle relative to the earth's latitude/longitude coordinate system. The three accelerometers in this type system are designated X, Y, or Z, depending on the direction of sensitivity. The LTN-72 Inertial Navigation System, with its wander-azimuth mechanization, is an example of this latter type of system.[5]

Stable Platform The mounting surface for the accelerometers must provide a reference plane so that the effects of gravitational acceleration can either be isolated from the accelerometers, or be calculated and subtracted from the acceleration output signal. Also, the accelerometer's orientation in azimuth must be measured and controlled to determine the flight path direction. The accelerometer mounting surface is called a *stable platform.*

In a semi-analytic inertial system (like the LTN-72), the stable platform is mounted in a gimbal set (see Figure 3) and it is controlled to define a level and an azimuth position relative to the earth. The gimbals are hard-mounted to the aircraft, through the inertial navigation unit housing. As a result, the pitch, roll and azimuth angles of the aircraft — relative to the earth — can be measured between the attitude of the stable platform and the gimbal mounts.

In order for a semi-analytic inertial navigation system to maintain its stable platform in an earth-referenced level position as the aircraft moves over the spherical earth (see Figure 4), the platform must be torqued or controlled mechanically. This is called *platform compensation.*[6] Other types of inertial navigation systems (full analytic, geometric, strap-down) establish a platform reference that is other than earth-level. As such, the accelerometers in these other systems are subject to gravitational acceleration, in addition to vehicle acceleration. However, in each case the navigation computer determines the effects of the gravitation for the existing platform angle, and it corrects the accelerometer outputs in order to develop an accurate dead-reckoned position.

The attitude of a semi-analytic inertial navigation system stable platform is controlled with corrections that are developed by the navigation computer. The computer uses the inertial navigation system's gyroscopes to define and maintain the correct attitude of the system's stable platform by signaling the gyroscope torquers to apply a precession force to the gyros. The individual gyros, once positioned by the computer at its definition of level, define the attitude corrections required by the platform, and command the gimbal torque

[5]The LTN-72 Inertial Navigation Units procured for U.S. Navy aircraft do not have Z-axis accelerometers installed.

[6]The actual control of the platform to accomplish compensation requires consideration of the total inertial navigator system function, and will be discussed in that portion of this article.

motors (Figure 5) to move the platform to the desired position in the gimbals for both the level and azimuth axes. When a change in aircraft course or attitude causes the platform to deviate from computer-defined level, the gyroscopes detect the off-level condition and drive the platform to correct the error. This is called *platform stabilization.* Operation of the individual gyroscopes will be discussed in the next section of this article.

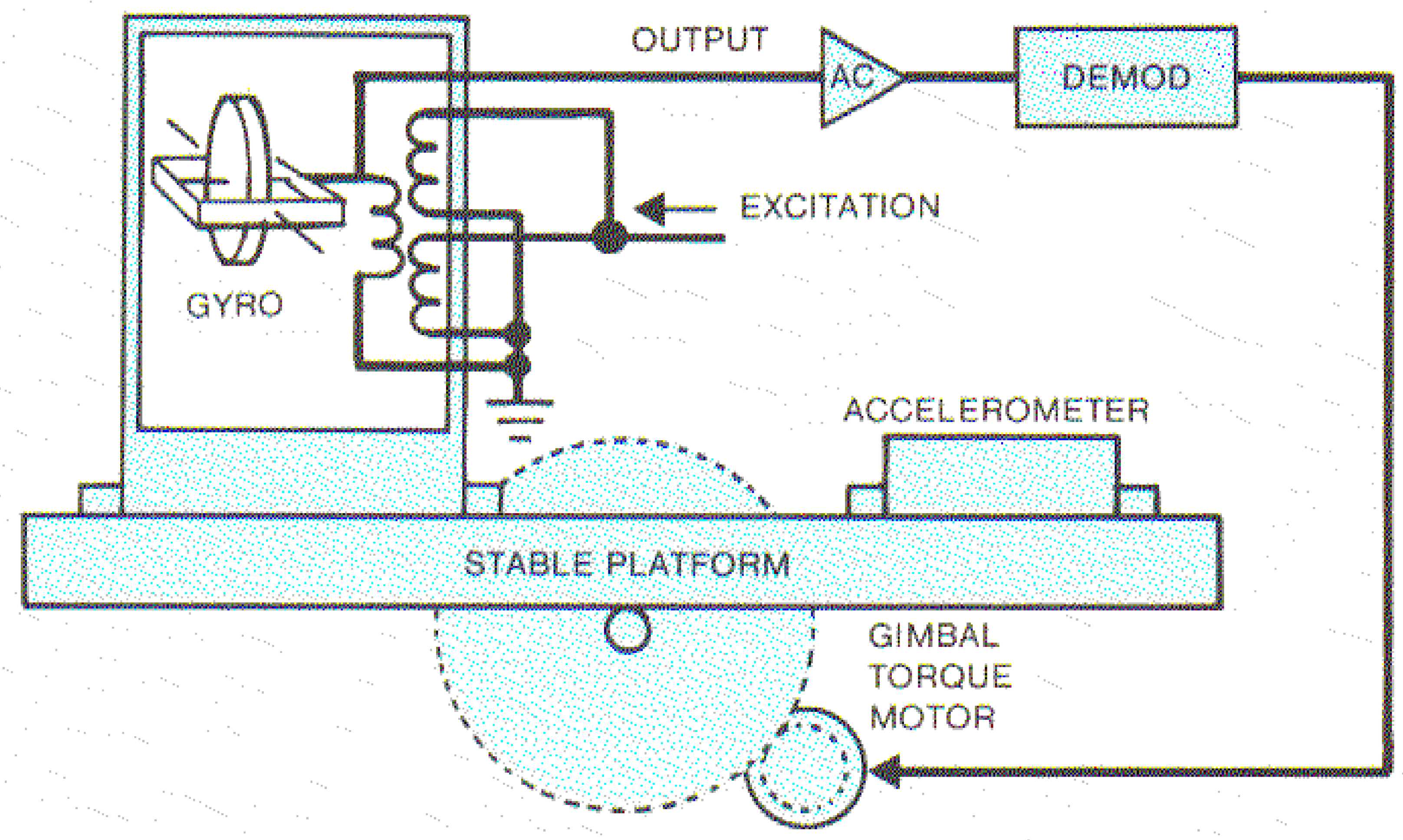


Figure 5. Inertial Platform Stabilization

Inertial systems designed with a gimbaled stable platform are subject to errors that are produced by the acceleration of the platform. A natural oscillation develops as a result of the "pendulum" effects of the platform's gimbal joints and the stable element. A pendulum breaks into oscillation about its rotation axis when it is accelerated as a result of the inertia of its mass resisting the movement of its axis.

We can view the stable platform as a pendulum, with the platform, accelerometers, and gyroscopes acting as the mass and the gimbals acting as the rotation axis. As the host vehicle is accelerated, the platform "mass" will initially resist the motion imparted by the gimbals. This produces an "off-level" condition of the platform, and causes the accelerometers to sense a "false" (grav-

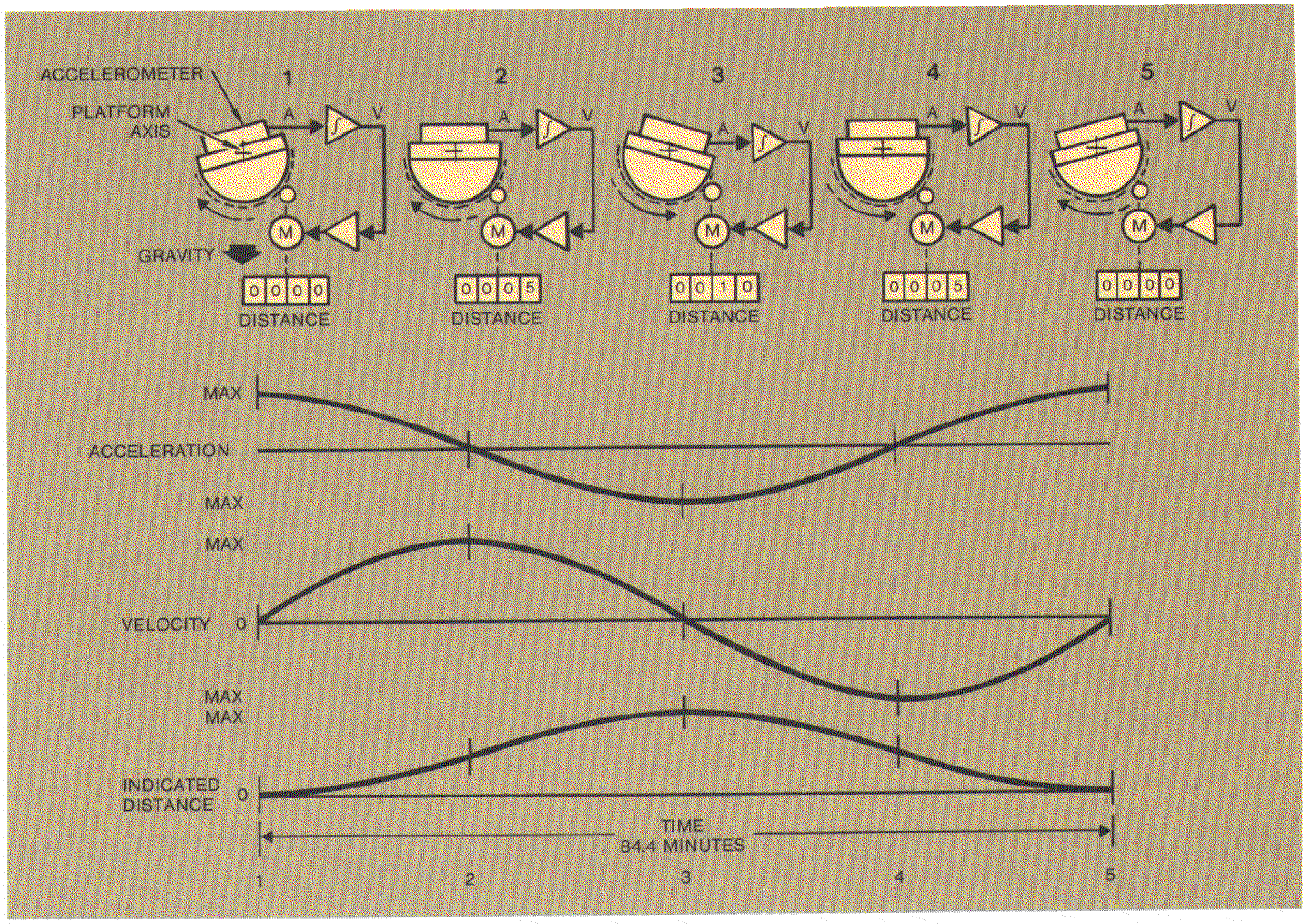


Figure 6. Inertial Platform Schuler Operation

itational) acceleration. Once set into motion, the platform will oscillate at its natural frequency until the motion is damped and the platform is brought back to rest. If compensation is not provided for this condition, the relatively small damping of the the inertial platform and the frequency of changes in velocity during flight will result in an almost constant random error in dead-reckoned position.

Schuler Tuning In the early 1900s, a German engineer named Max Schuler proved that the pendulous oscillation of gyroscopes could be eliminated if the device was designed with mechanical characteristics that simulated a "pendulum" whose length is equivalent to the radius of the earth. This would ensure that the "pendulum's" center of mass would effectively be at the center of the earth, rendering the gyroscope insensitive to acceleration. The errors produced by the natural oscillation of the inertial platform also can be eliminated by designing the platform control response using this technique.

Platforms designed with this characteristic, known as Schuler Tuning, will *not* oscillate due to acceleration. Any errors that develop as a result of accelerometer bias or platform offset will cause the platform to be driven (by the platform compensation function) at the natural frequency of the system, referred to as the Schuler Cycle. The cyclic platform movement commands are developed by the gyroscopes in response to the gyro torque control signals from the INS computer. The Schuler operation cycle of the platform, shown in Figure 6, has a period of 84.4 minutes, which is equivalent to that of an earth-radius pendulum. Once the inertial navigation system is in a navigate mode of operation, the Schuler Cycle operates continuously. The resulting improvement in system position accuracy is shown in Figure 7.

Gyroscopes The position of the stable inertial platform is controlled by the gyroscopes. The inertial stability of the gyroscopes, when combined with the appropriate platform correction commands generated by the computer, establishes the reference level condition that is desired for the platform.

The gyroscope is a device that is sensitive to angular motion about its input axis. The gyro consists of a rotating wheel mounted in its own gimbal structure. The gyro's gimbal is mounted on (and referenced to) the stable platform (see Figure 5). Electrical signal pickoffs, mounted between the gimbal and the gyro's support structure, measure the difference or error between the position (rotation plane) of the rotating wheel and the stable platform. Error signals are generated and applied to the platform torque motors to drive the platform and correct its off-level condition.

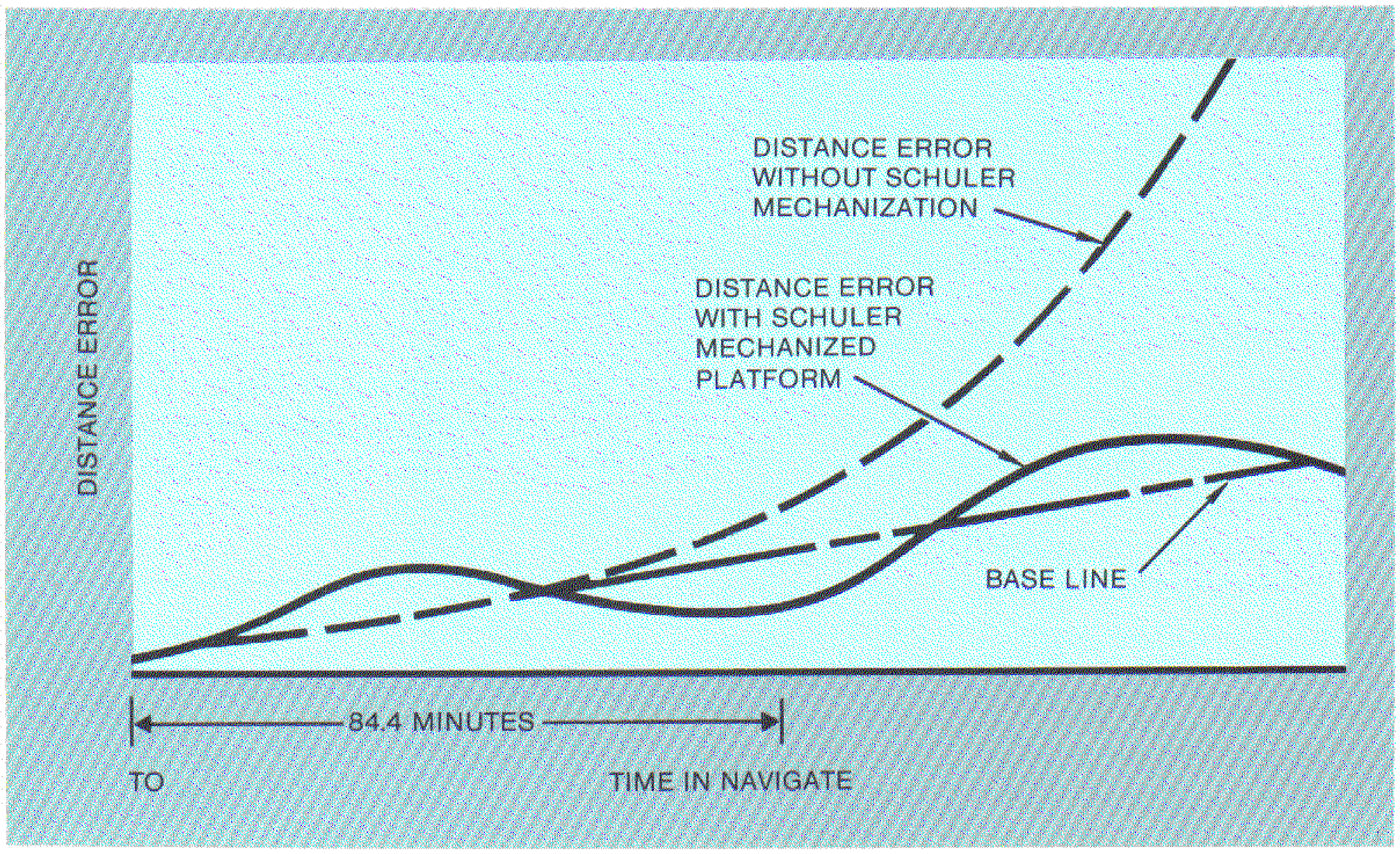
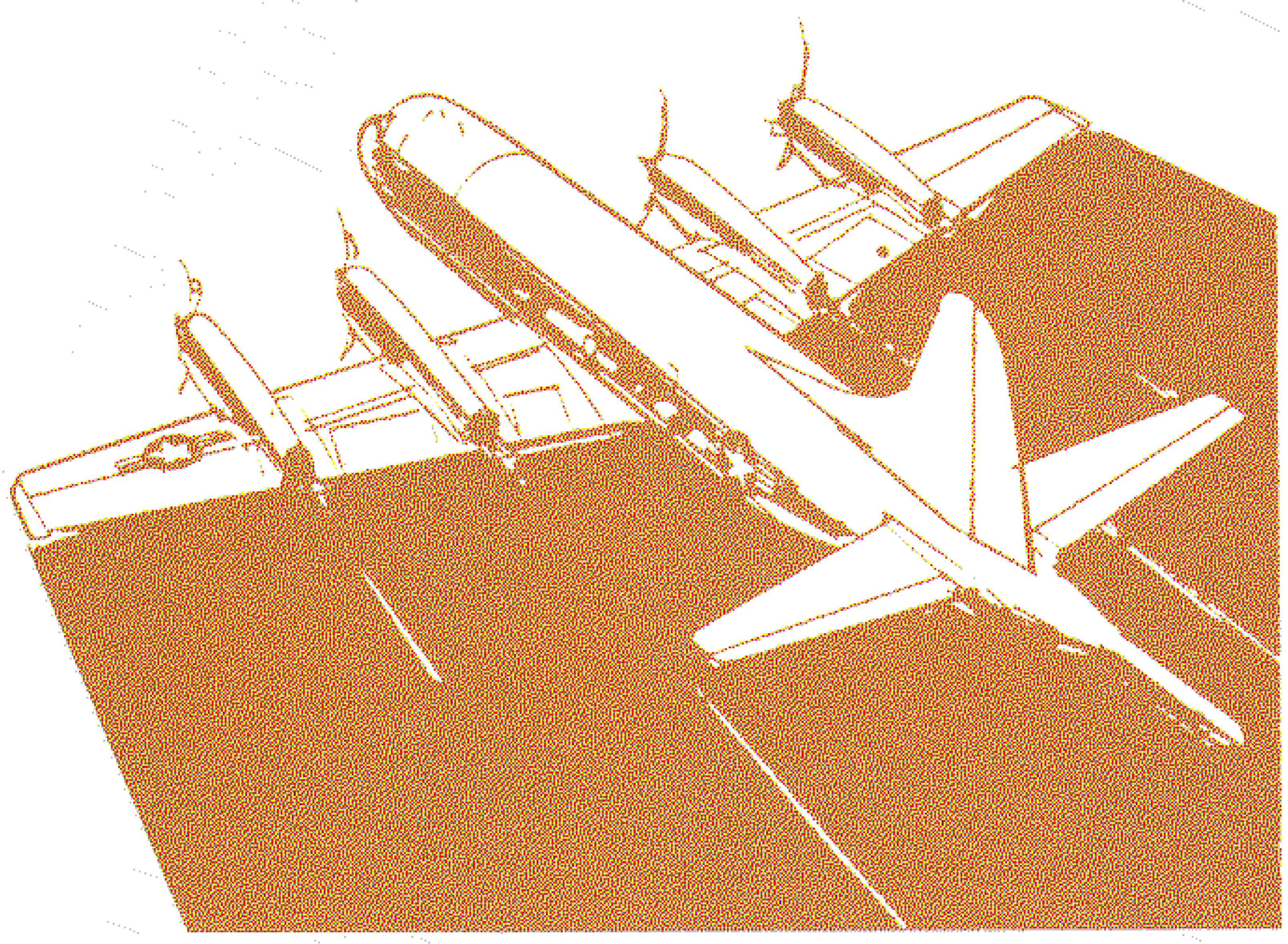


Figure 7. Improving Inertial Navigation System Accuracy with Schuler Tuning



The physical laws that describe how the stability of a gyroscope is produced by the rotation of its wheel are complex. For our purpose, suffice it to say that the gyroscope acts as a rigid body in inertial space by virtue of the angular momentum developed due to the high rotation speed of the wheel. Once the gyroscope's wheel is positioned in space, it will maintain its plane of rotation relative to space. Since the earth is rotating (on its axis) in space, the rotation plane of the gyroscope's wheel will *appear* to be moving relative to the earth-level reference. (Actually, it is the rotating earth that is changing position relative to the stable gyro; more on this later.)

The preceding discussion of gyroscopic stability is simplified to represent the theory. In practice, some minimal imbalance is introduced to most gyroscopes when they are manufactured. This can cause the gyro to "wander" or move while it is operating — an undesirable phenomenon that is called "gyro drift." The accuracy of the INS would be adversely affected if gyro drift was allowed to generate a false command to the stable platform control motors and drive the platform off-level. This would cause the system's accelerometers to sense gravitational acceleration. Fortunately, the designers of inertial navigation systems know why gyro drift occurs and how to compensate for it.

After fabrication, the gyroscope manufacturer runs each production gyro to determine its unique drift characteristics. From this functional check, the manufacturer calculates a correction for the individual gyro. This correction is called the *gyro bias*. The bias value for the individual gyro is loaded into the INS computer. The INS computer then uses this bias value to apply a continuous electrical torque to the gyro to counter any inherent drift.

During the operational life of an inertial navigation system, the mechanical tolerances of its gyroscopes can change. When this occurs, the gyro bias must be revised in order to maintain system accuracy. Gyro bias update techniques have been refined and automated since the introduction of the first generation ASN-42 Inertial Navigation System in the P-3A aircraft.

Navigation Computer The fourth element required for an inertial navigation system is a computer. Previously, we have alluded to the need for a computer to: (1) calculate velocity and position (dead-reckon navigate), based on acceleration data sensed over a period of time; (2) monitor movement of the stable platform over the rotating earth; (3) generate the control signals that are required to maintain the stable platform level and properly oriented in azimuth, and to correct for apparent accelerations; and (4), to store the gyro bias value and combine it with the platform control requirements to generate a gyro torque command. These are the most fundamental requirements for the computer. Other computer functions may include:

- Waypoint Storage — Implemented for the LTN-72 Inertial Navigation System.
- True Heading Calculation — Required for systems that are not mechanized as north-pointing (LTN-72 Wander Azimuth and ASN-84 Unipolar Navigation).
- Steering and Track Error — Calculates steering commands for the autopilot, and angle and distance errors relative to desired track and course.

Official Photograph U.S. Navy

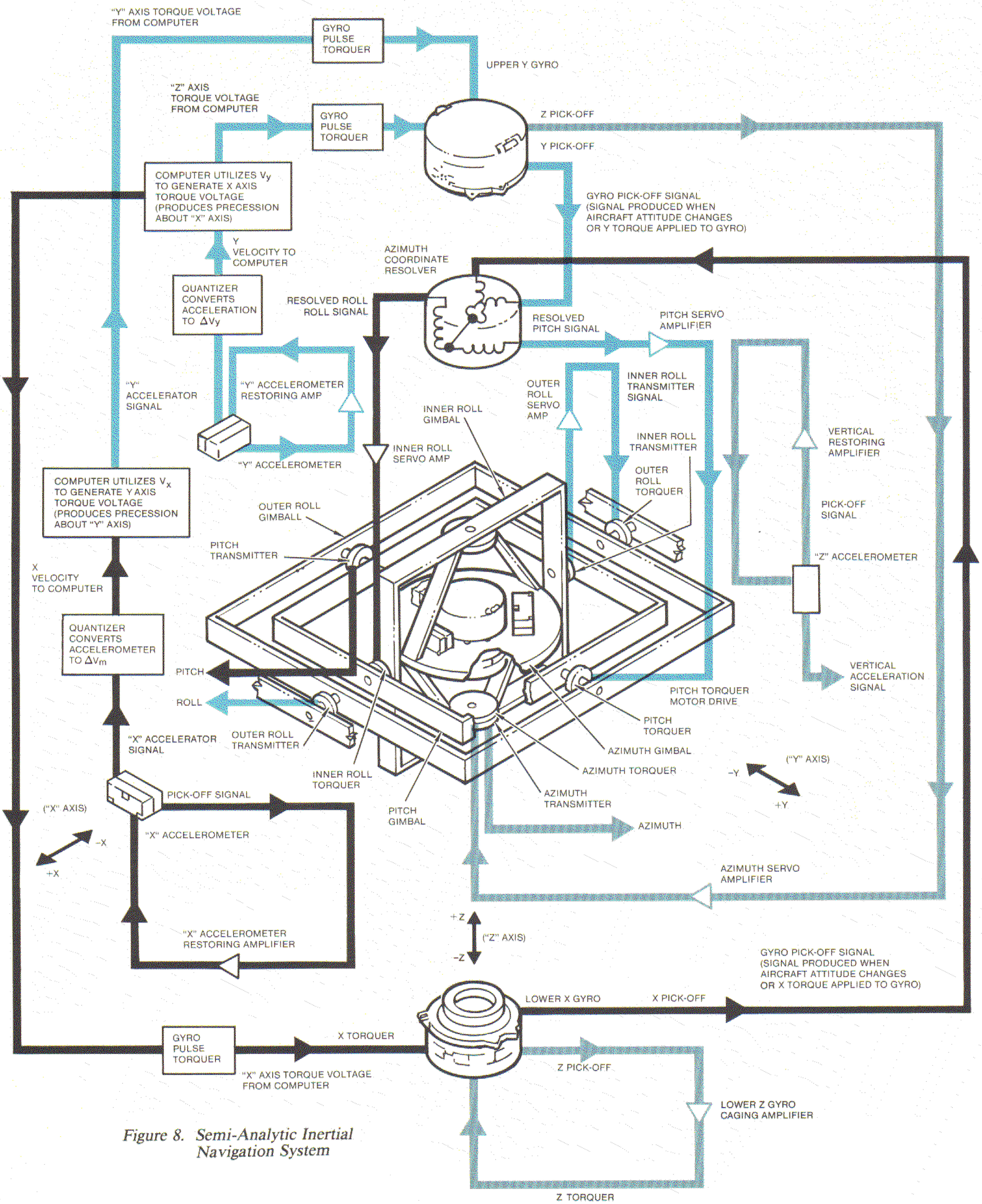
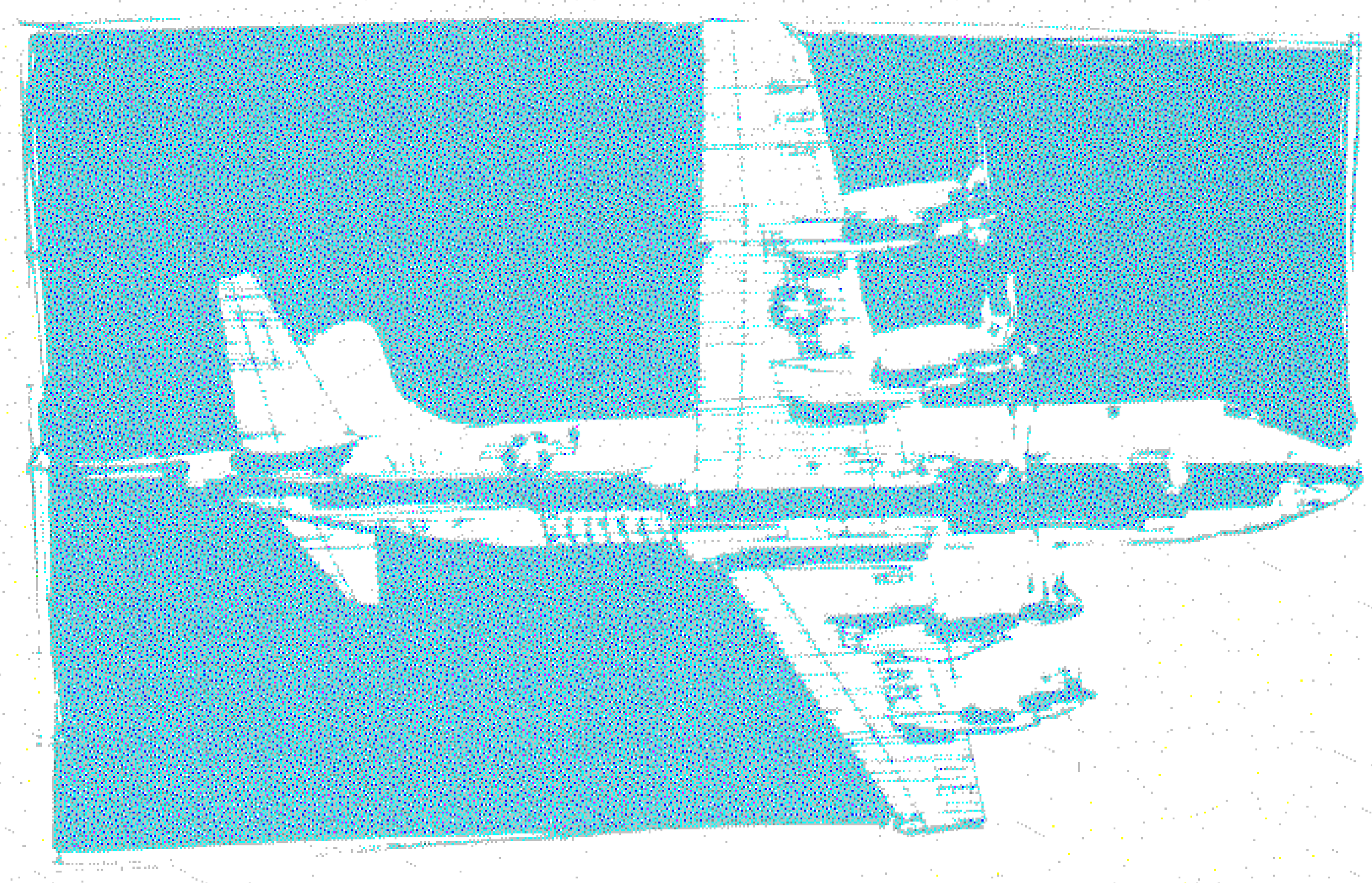


Figure 8. Semi-Analytic Inertial Navigation System

SYSTEM FUNCTION AND OPERATION An inertial navigator is a completely self-contained, non-radiating system that provides attitude, true heading, ground speed and course, and dead-reckoned position. Depending upon their mechanization, these systems may be categorized as follows:

- Analytic Inertial System — A system in which the gyroscopes and accelerometers are oriented to a fixed reference in inertial space.
- Semi-Analytic Inertial System — A system in which the gyroscopes and accelerometers are oriented to a local vertical (perpendicular to the earth's gravitational force).
- Geometric Inertial System — A system in which gyroscopes are oriented in inertial space, and the accelerometers are oriented to the local vertical.
- Strap-Down Inertial System — A system in which the gyroscopes do not maintain any set orientation, and the accelerometers follow the orientation of the vehicle.

The mechanization for a semi-analytic inertial navigation system is illustrated in Figure 8. The gyroscopes establish a level reference for the inertial navigation system. They also generate the control signals that command the platform torque motors to position the platform stable element in a level attitude and maintain it in that condition.



This, in turn, isolates the accelerometers from the effects of gravity, enabling measurement of north/south and east/west acceleration of the aircraft. The computer integrates the acceleration measurements over time to calculate the aircraft's velocity and accumulated distance traveled.

Our discussion to this point has described the requirements for navigation on a constant heading over a flat, non-rotating surface. Some additional controls must be imposed on the system to provide navigation on various headings over a rotating, spherical earth. In our discussion of the requirements for the navigation computer, we identified the need to make platform attitude corrections to maintain a level platform condition (relative to the local earth-definition of level) and to make corrections to the accelerometer outputs. These correction signals are defined and summarized in Table II.

TABLE II. Inertial Navigation System Correction Signals

INERTIAL DEVICE	CORRECTION	DEFINITION
GYROSCOPE	EARTH RATE	COMMAND ROTATES PLATFORM IN SPACE TO REMAIN LEVEL RELATIVE TO ROTATING EARTH
	TRANSPORT RATE	COMMAND ROTATES PLATFORM IN SPACE TO REMAIN LEVEL AS AIRCRAFT TRANSITS THE SPHERICAL EARTH
	BIAS	SUPPLIES A TORQUE TO THE GYRO TO COUNTER MECHANICAL DRIFT
ACCELEROMETER	CENTRIPETAL	CORRECTS NORTH/SOUTH ACCELERATION ERROR THAT EXISTS WITH FLIGHT ALONG LATITUDES OTHER THAN THE EQUATOR
	CORIOLIS	CORRECTS APPARENT EAST/WEST ACCELERATION ERRORS THAT OCCUR WHEN THE AIRCRAFT TRAVELS NORTH OR SOUTH

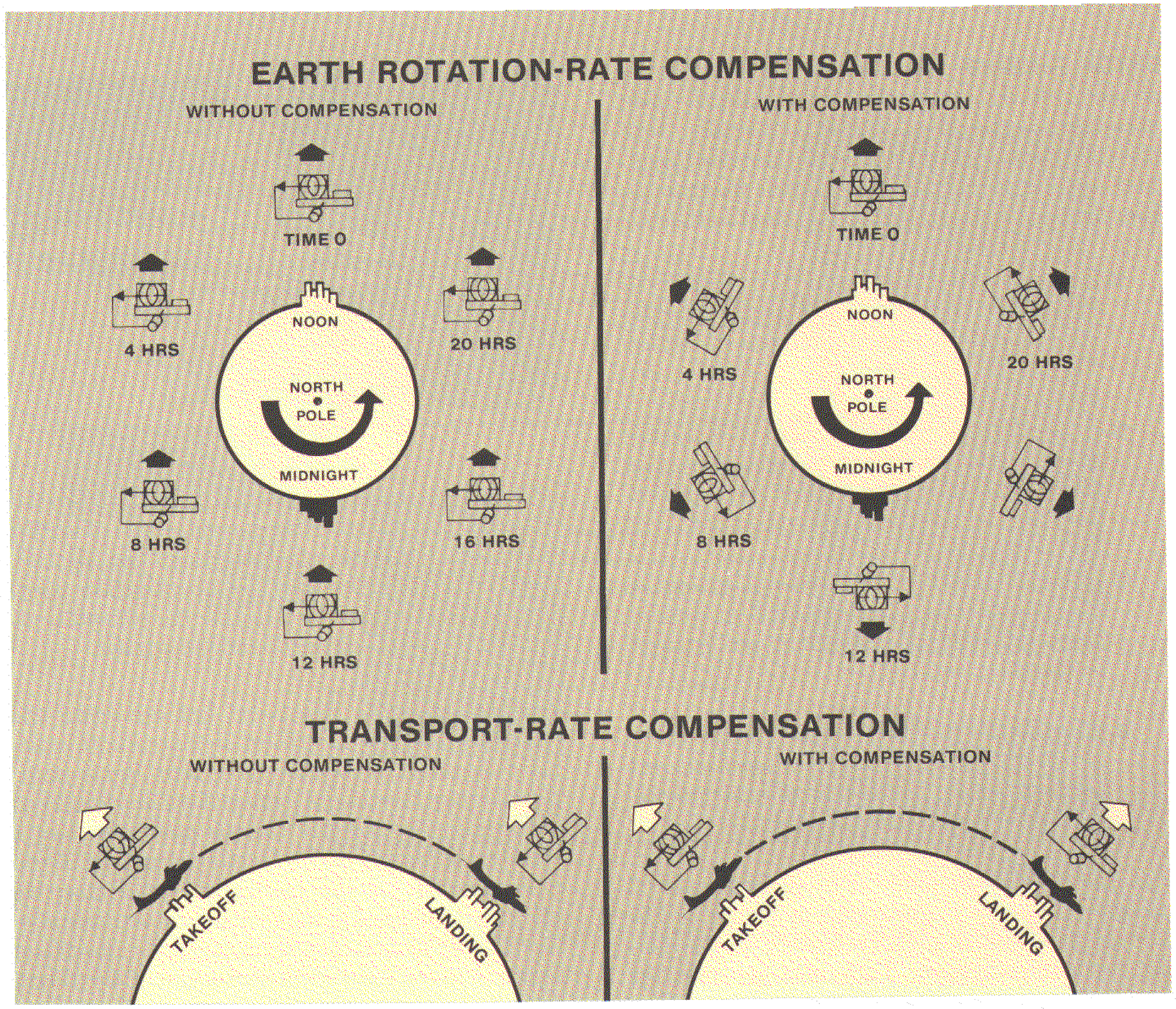


Figure 9. Inertial Platform Earth Rotation-Rate and Transport-Rate Compensation

Gyroscope Corrections When we discussed the basic operation of the gyroscope, we established that during on-speed operation the gyro will maintain a stable position in a gravitational field (ignoring, for the moment, mechanical drift). This is illustrated in Figure 9. Notice that as the earth rotates or the platform is moved, the gyro axis maintains its initial alignment, and is unaffected by the earth's gravitational field. Since the level reference defined by the stable platform is controlled electrically by the gyro output signals, the platform position will be stable relative to the gyro rather than to the earth. This condition will cause the platform to tilt relative to local earth-level, introducing the effects of gravity to the platform-mounted accelerometers, resulting in the measurement of "false" acceleration. Proper operation of a semi-analytic inertial navigation system requires that the accelerometers be isolated from this "false" gravitational acceleration.

If we want to maintain an *earth-referenced level platform*, the navigation system computer must calculate the correction signals required to torque or precess the *gyros* to the earth-referenced position in space. This will produce gyro electrical commands for the platform torque motors (see Figure 10). The proper correction command for each of the three platform-mounted gyroscopes varies,[7] depending on the effects of the earth's constant rotation rate at the latitude where the platform is located.

Additional correction commands are sent to the gyros, commanding them to torque the platform at a rate that corresponds to the movement of the aircraft over the earth's spherical surface. This additional correction term is called the *transport-rate correction*, and is additive to the previously-described *earth-rate correction*. The process of precessing the gyros to control the orientation of the platform (compensation) is called *platform control*. The two rate-correction signals, when combined with the Schuler tuning requirements, produce the platform control signal. The detection/correction of errors in the platform position (off-level condition) and the application of gyro bias to eliminate drift produce the *platform stabilization signal*. The relationship of these signals is shown in Figure 10.

At this point, we need to expand on our discussion of the requirements for *azimuth control of the platform*. In order for the navigation computer to calculate aircraft position, it must receive both velocity and direction information from the inertial platform. Velocity is calculated by integrating the outputs of the accelerometers.

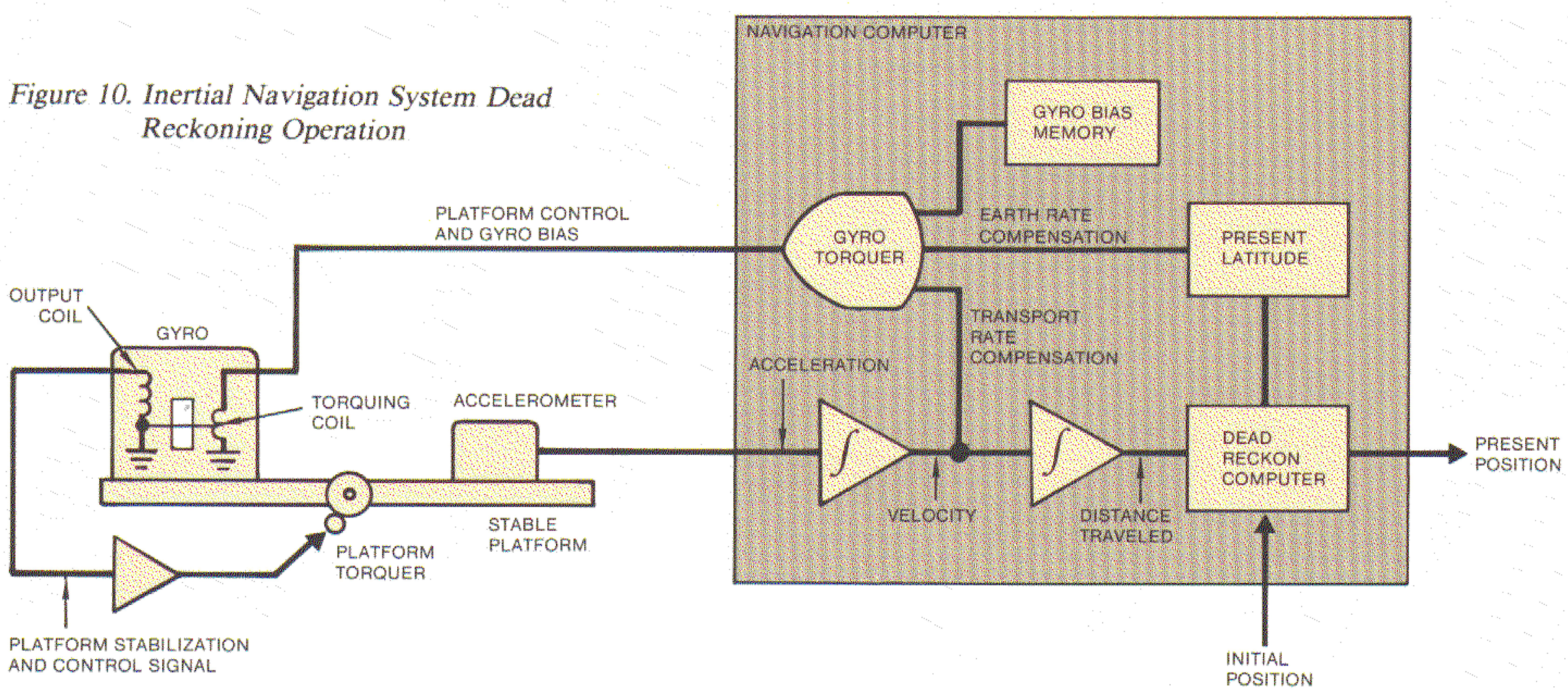
[7]Most inertial platforms use two Two-Degree-of-Freedom (TDF) gyros, as opposed to three Single-Degree-of-Freedom (SDF) instruments. A TDF gyro is sensitive to motion in two axes. As a result, two TDF gyros will provide measurement in all three axes (X, Y, Z).

Aircraft heading is required to establish the flight path direction. As shown in Figure 3, a semi-analytic inertial navigation system has its gyroscopes arranged to provide a stable reference for three directions — the horizontal X- and Y-axes, and the vertical Z-axis. The platform leveling function is managed by the horizontal (X- and Y-axes) gyros; the vertical (Z-axis) gyro controls the orientation of the platform in azimuth. The Z-axis gyro detects motion of the platform in azimuth, and generates a command signal to the azimuth drive torque motors. The response of the stable platform to these attitude and azimuth correction commands causes the platform to maintain a defined reference for the X- and Y-accelerometers relative to the earth's poles.

The angle between the reference position of the stable platform and the aircraft heading (aircraft navigation centerline) is measured by a synchro transmitter that is mounted on the azimuth gimbal. In effect, the aircraft rotates around the platform during maneuvering. The goal is to maintain the *defined* accelerometer orientation. This platform heading synchro output is a directional gyro-stabilized signal that represents the *change* in aircraft heading that occurs during a turn.

First-generation inertial navigation systems, like the ASN-42 system, were mechanized so that one axis of the platform was always pointing to true

Figure 10. Inertial Navigation System Dead Reckoning Operation



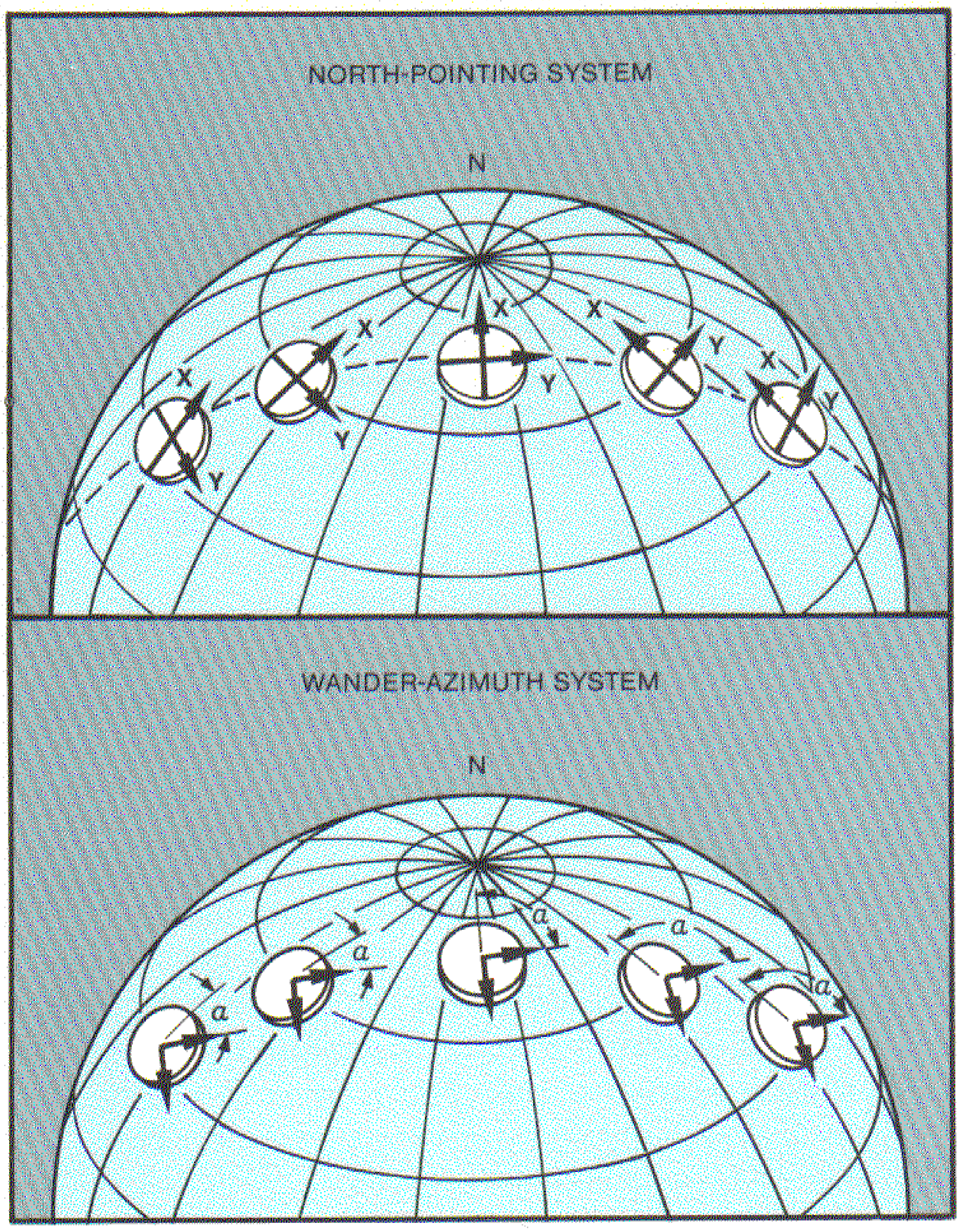


Figure 11. Inertial Platform Azimuth Orientation — North-Pointing System and Wander-Azimuth System

north (see Figure 11). As a result, the platform heading output from these systems provided a true north compass reference. Unfortunately, maintaining a north-pointing reference resulted in unacceptably high platform-rotation rates when the aircraft was operated at high latitudes.

Later model inertial navigation systems have been developed with alternate azimuth control schemes to circumvent this problem. These systems are called *wander azimuth* systems, and they are mechanized to rotate the INS computer "axes" to track actual true north. This tracking technique results in the definition of an angle between true north and the platform's horizontal (X, Y) axes. This angle, designated the *alpha angle*, is calculated by the INS computer during system alignment. The alpha angle is used to resolve the X- and Y-accelerometer outputs to a north/south and or east/west reference. We will discuss how the alpha angle is established later in this article when we deal with the LTN-72 inertial navigation system alignment sequence.

Once the alpha angle is defined, the inertial platform holds a *constant platform heading* as the aircraft moves along the flight path. It is not necessary to apply mechanical torque to make the platform track true north because the computer is "rotating its reference axes" to continually update the changing alpha (wander) angle. If the aircraft changes heading (rhumb line), the Z-gyro senses the motion and applies torque to the stable platform to maintain the alpha angle (angle between true north and platform horizontal axis) that existed at the start of the turn. As before, the inertial navigation system computer continues to track the alpha angle rotation in order to track true north along the flight path. This results in zero platform rotation for flight along a constant rhumb line, which reduces the occurrence of false accelerometer signals that were inherent in the earlier-generation inertial navigation systems.

Accelerometer Corrections When an inertial navigation system navigates over the rotating, spherical earth, corrections must be applied to the accelerometer outputs. These corrections are applied to eliminate errors in velocity and accumulated distance-traveled that occur when the accelerometers sense "false" accelerations. As discussed previously, an accelerometer will detect any accelerations along its sensitive axis, and it is incapable of differentiating between the sources of acceleration.

When the inertial navigation system platform moves over the earth, its position is tracked relative to coordinates on the surface. False accelerations result from the inertial system tracking the motion of the moving, earth-bound coordinate system. Consequently, it is necessary to combine the correction terms (identified in Table II) with the accelerometer outputs in the computer in order to "bias-out" false accelerations. *Coriolis correction* and *centripetal correction* are the two terms for which compensation is made.

PART II – THE P-3C NAVIGATION SYSTEM

TACTICAL/GEOGRAPHIC NAVIGATION

In the P-3C aircraft, the tactical avionics systems are integrated with the CP-901/ASQ-114 Digital Computer. This computer currently performs all of the tactical system calculations, record-keeping and system management. One of its subroutines is a continuous, dead-reckoning calculation based on data from one of three sources: the Doppler Radar Navigation System, the Inertial Navigation System, or the Air Data System. Figure 12 shows the basic P-3C navigation interface with the tactical avionics. The CP-901/ASQ-114 Digital Computer performs the dead-reckoning (DR) calculations in either the Tactical Navigation (TACNAV) mode or the Geographic Navigation (GEONAV) mode. During operation in either of these navigation modes, the computer can use data from any of the three sources mentioned above to perform DR calculations.

The TACCO selects the TACNAV mode to establish the aircraft position relative to the ASW search stores that have been deployed in the water. The details of tactical navigation in the P-3C aircraft are discussed in *Orion Service Digest Issue 37*, and the reader is referred to that article if a more in-depth description is desired. In the GEONAV Mode, the CP-901 Computer tracks and displays the aircraft's present position relative to latitude and longitude.

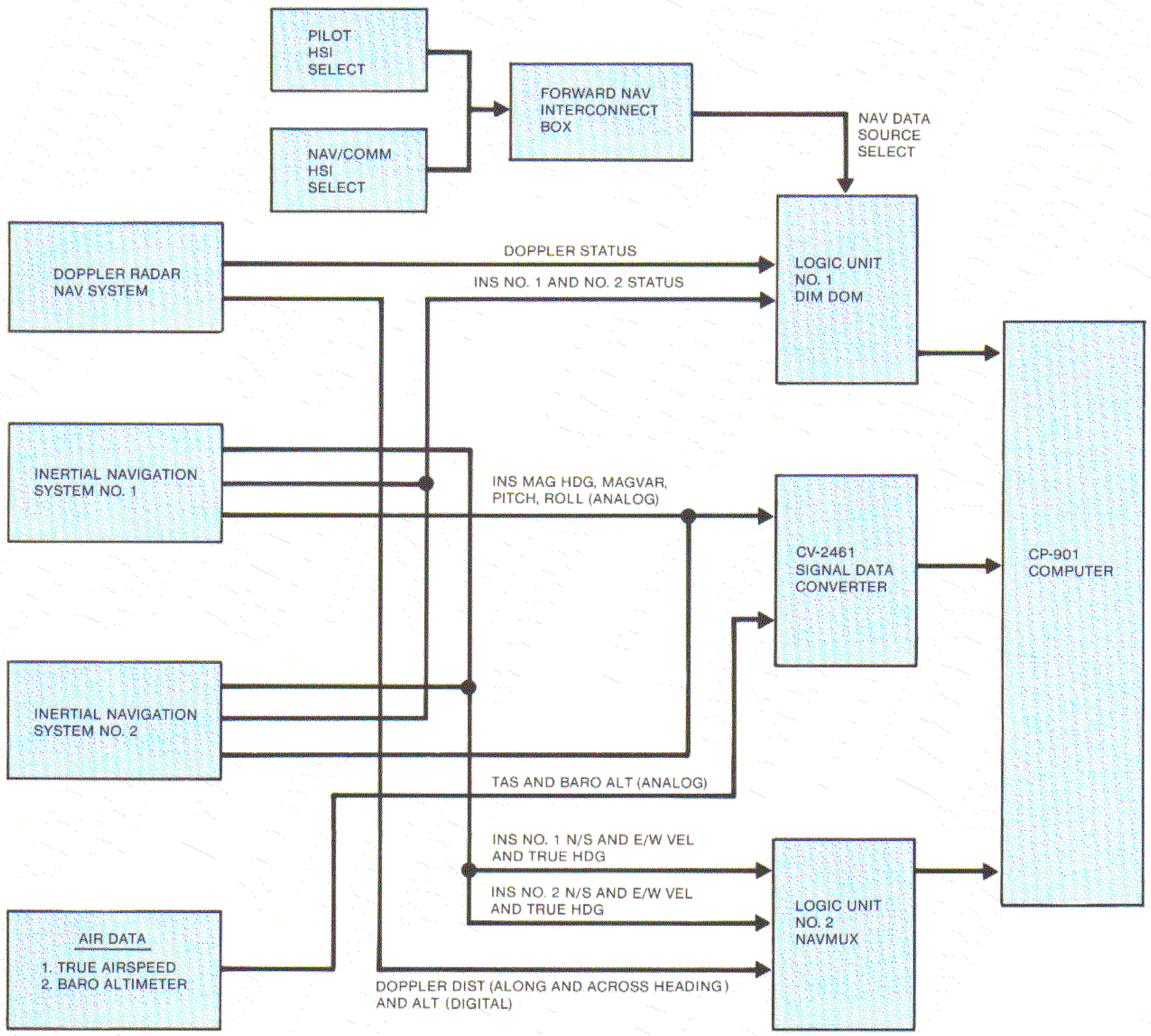


Figure 12. Basic P-3C Navigation System/Tactical Computer Interface

INERTIAL NAVIGATION SYSTEM/CP-901 COMPUTER INTERFACE

The CP-901/ASQ-114 Computer's software automatically selects one of the aircraft's three navigation systems (Doppler, Inertial, or Air Data) to provide the primary and backup dead-reckoning navigation data. The navigation system assignment priorities depend on the navigation mode that is in use. These priorities are:

GEONAV	Primary-Inertial	Provides N/S and E/W velocity, and true heading to the CP-901 computer.
	Backup-Doppler	Provides distance along and across the heading; true heading is supplied by the INS.
TACNAV	Primary-Doppler	Same as above.
	Backup-Inertial	Same as above.

This automatic navigation mode selection can be overridden by the NAV/COMM operator. Note in Figure 12 that *both* inertial systems interface (for data) through the Logic Unit No. 2 Navigation Multiplexer (NAVMUX). The CP-901 Computer uses data from only one inertial navigation system at a time when it performs dead reckoning with the inertial navigation system. The CP-901 Computer software selects which INS will be used to provide data, based on: (1) the condition of the INS status discrete signal inputs to Logic Unit No. 1 DIMDOM,[8] and (2) assessment of the validity of the ground speed and true heading data from the INS.

The inertial navigation system selected by the NAV/COMM operator as the data source for his Horizontal Situation Indicator (HSI) is used initially by the CP-901 Computer. The computer's software periodically checks (via Logic Unit No. 1) the NAV/COMM HSI source-select status, the INS DIGITAL VALID discrete signal, and the INS NAV MODE discrete signal (see Figure 13). If the selected inertial navigation system's operational status becomes INVALID (either DIGI-

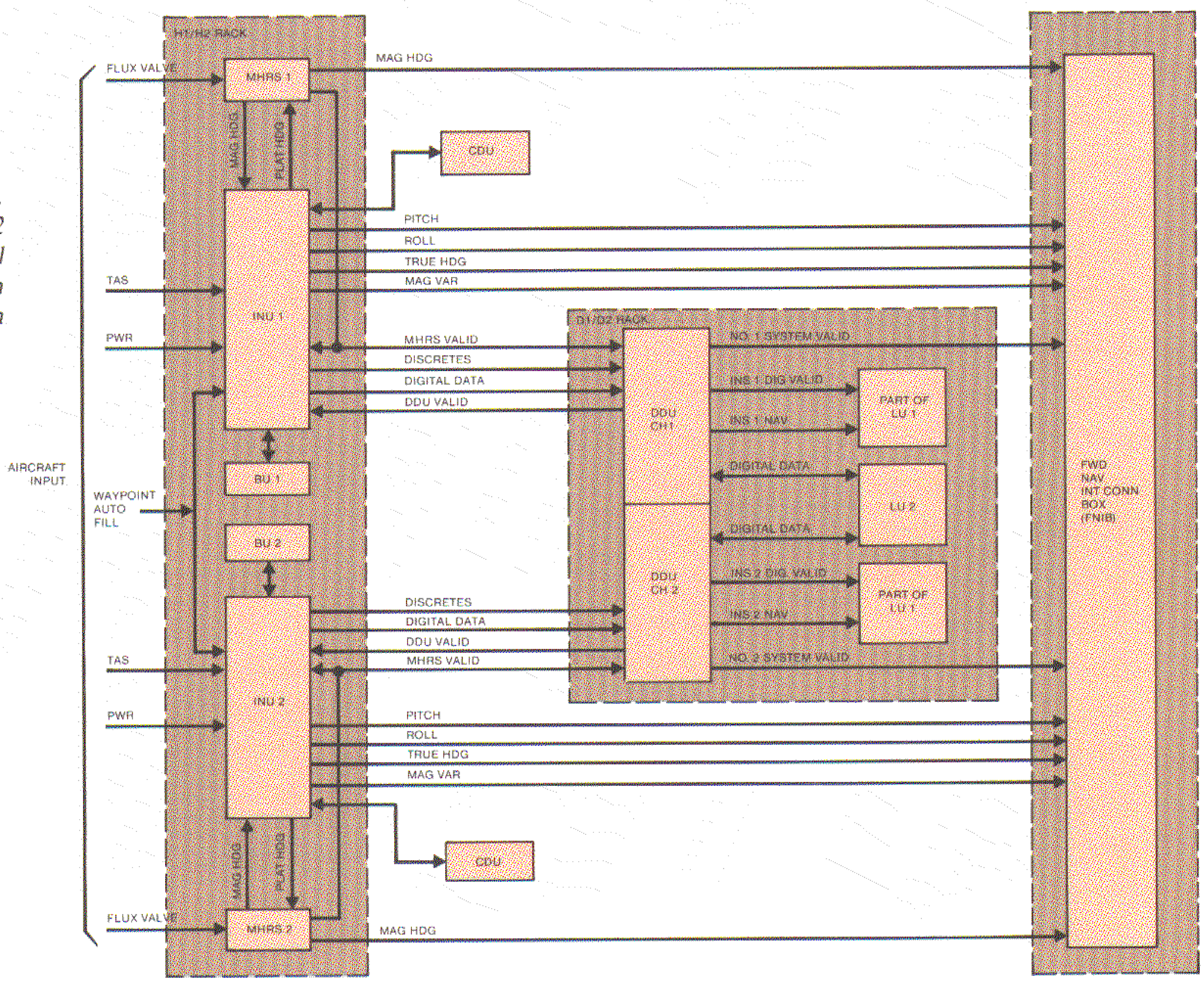


Figure 13. P-3C LTN-72 Inertial Navigation System Block Diagram

TABLE III. LTN-72 Inertial Navigation System

	WEIGHT	INSTALLED LOCATION
LITTON AERO PRODUCTS		
INERTIAL NAVIGATION UNIT (INU)	56.4 LBS	H RACK
CONTROL DISPLAY UNIT (CDU)	4.8 LBS	NAV/COMM STATION
MODE SELECTOR UNIT (MSU)	0.7 LB	FLIGHT STATION
DIGITAL DATA UNIT (DDU)	8.25 LBS	D1 RACK
SPERRY FLIGHT SYSTEMS		
MAGNETIC HEADING REFERENCE SYSTEM		
COMPASS COUPLER	7.2 LBS	H RACK
COMPASS CONTROLLER	0.8 LB	FLIGHT STATION
MARATHON		
INS BATTERY UNIT	17.0 LBS	H RACK
LOCKHEED		
NAVIGATION POWER ALARM CONTROL (A509 NAV PAC)	2.1 LBS	H RACK

TAL or NAV MODE), the de-selected INS is automatically used as the source for dead-reckoning data (assuming that *its* INS status discrete signals are VALID) until the status of the selected INS resets, and a cue is provided to the NAV/COMM that the selected INS is unreliable. If both system status discrete signals are INVALID, the computer's software selects the backup navigation data source (Doppler or Air Data).

The CP-901 Computer also performs a ground speed and true heading test that checks to determine if the velocity and heading data received from the inertial navigation system are reasonable. The NAV/COMM operator will receive a cue if either the ground speed or true heading data from the selected INS are determined to be questionable, and at his option he may select the alternate INS at the station HSI control panel. In addition, if the ground speed test fails, the Computer reverts to the de-selected INS for that calculation iteration, and then switches back to the selected system to run another test.

As shown in Figure 12, additional analog attitude and heading information is supplied to the CP-901 Computer via the CV-2461 Signal Data Converter. This information is provided to support other tactical functions that the CP-901 Computer performs, and it includes pitch and roll angles, magnetic heading, and magnetic variation — all supplied as synchro analog signals. The CV-2461 Signal Data Converter converts the analog input angle to an ANEW digital word format for the computer.

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THE LTN-72 INERTIAL NAVIGATION SYSTEM

GENERAL The P-3C Inertial Navigation System consists of a fully-integrated, dual LTN-72 system installation. The two LTN-72 systems operate independently to provide dual sources of: (1) dead-reckoned position; (2) pitch angle, roll angle, and true heading; (3) magnetic heading and magnetic variation; and, (4) North/South and East/West velocity. The LTN-72 systems have provisions to operate for 15 to 30 minutes from power supplied by an INS-dedicated battery pack. Figure 13 is a block diagram that depicts the dual LTN-72 system installation interface in the P-3C aircraft.

The LTN-72 Inertial Navigation System was introduced in P-3C production at aircraft BUNO 161005 and up.[9] All prior P-3C aircraft have been converted from the ASN-84 INS to the LTN-72 INS by retrofit. The LTN-72 Inertial Navigation System hardware for the P-3C aircraft installation is listed in Table III.

[8]Digital Input Multiplexer/Digital Output Multiplexer.

[9]The first production installation of the LTN-72 INS in P-3C aircraft was in aircraft BUNO 161001.

POWER DISTRIBUTION The electrical power block diagram for a fully operational LTN-72 Inertial Navigation System installation in the P-3C aircraft as shown in Figure 14. During system warm-up, the Inertial Navigation Unit requires 115 VAC RUN and HEATER power to enable the system to progress through the alignment sequence. INS No. 1 is powered by the Monitorable Essential AC Bus, and INS No. 2 is powered by the Main AC Bus A. Each INU also requires that a minimum INS Battery voltage of +17.5 VDC be available before alignment can begin. If the battery voltage is less than this value or the circuit breaker on the battery is open, the system cannot be turned on. The battery is normally maintained at +26.5 VDC during system operation by a two-stage charger in the INU.

If the system is left to operate on battery power, there will be an automatic, orderly shutdown when the battery decays to +19.2 VDC. A fully discharged battery (to the +17.5 VDC cutoff) will require approximately 20 minutes of system ON operation for it to regain full charge.

The Inertial Navigation Unit receives 26 VAC, 400 Hz synchro excitation power from the Forward Navigation Interconnection Box (FNIB) and from the CPK-28/A24G-9 True Air Speed Computer. Circuit breakers (one per inertial system) on the FNIB provide 26 VAC Phase A power from an internal supply that generates synchro excitation for navigation system equipment. Most of the aircraft navigation system is referenced to the Phase A power from this supply. The

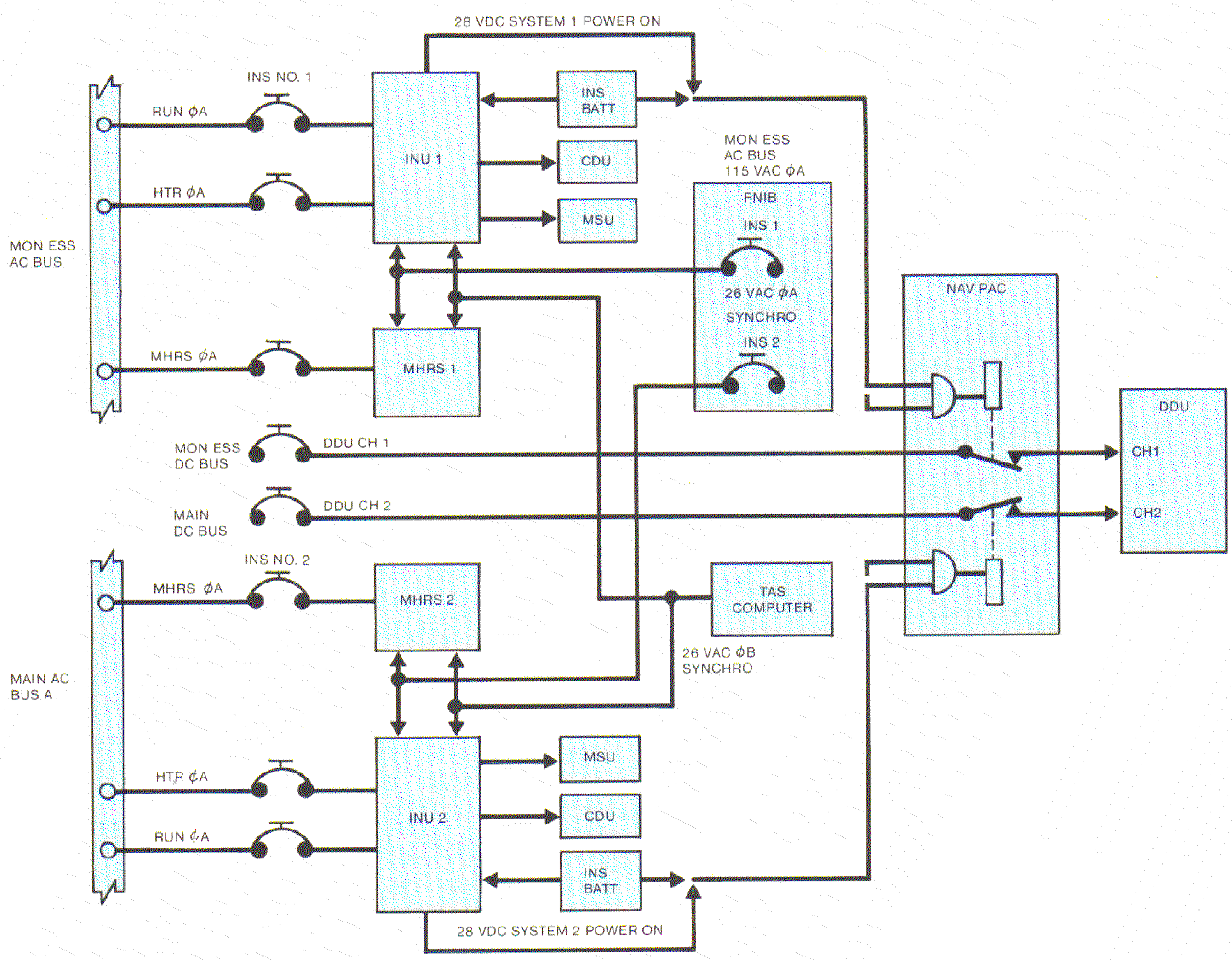


Figure 14. P-3C/LTN-72 Inertial Navigation System Electrical Power Distribution

FNIB is powered by the Monitorable Essential AC Bus. Loss of power from the MON ESS Bus will result in the loss of all attitude and heading output signals from *both* inertial systems. These signals will be lost, even though Inertial Navigation Unit No. 1 will continue to operate on INS No. 1 battery power and Inertial Navigation Unit No. 2 will continue to operate on power from Main AC Bus A.[10]

The 26 VAC Phase B synchro excitation signal is supplied to the Inertial Navigation System from the True Air Speed computer. This signal provides the INS with the reference that it requires to calculate wind speed and direction, and magnetic variation. The dependence of INS magnetic variation calculations on TAS power will be discussed in the Magnetic Heading Reference System section of this article. TAS power is controlled by a switch at the NAV/COMM station.

INERTIAL NAVIGATION UNIT The P-3C aircraft's LTN-72 Inertial Navigation Units are located in Electronic Racks H1 and H2. The INUs contain the system inertial components and navigation computer. Figure 15 shows the LTN-72 INU and

[10]If the attitude and heading signals from both LTN-72 Inertial Navigation Systems are lost, the pilot and copilot will still have use of the Flight Director attitude display by selecting the standby Gryo as the signal source.

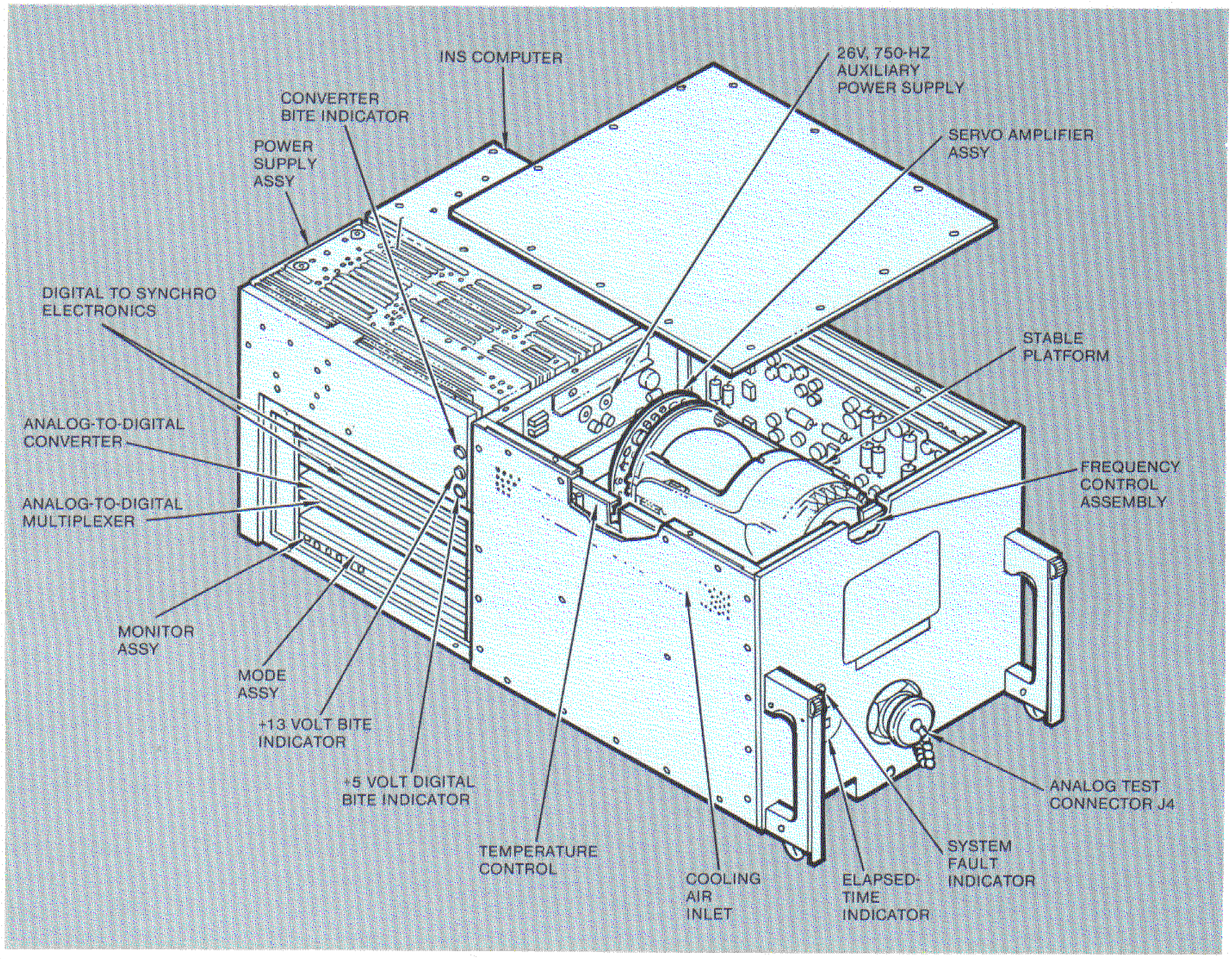


Figure 15. LTN-72 Inertial Navigation Unit

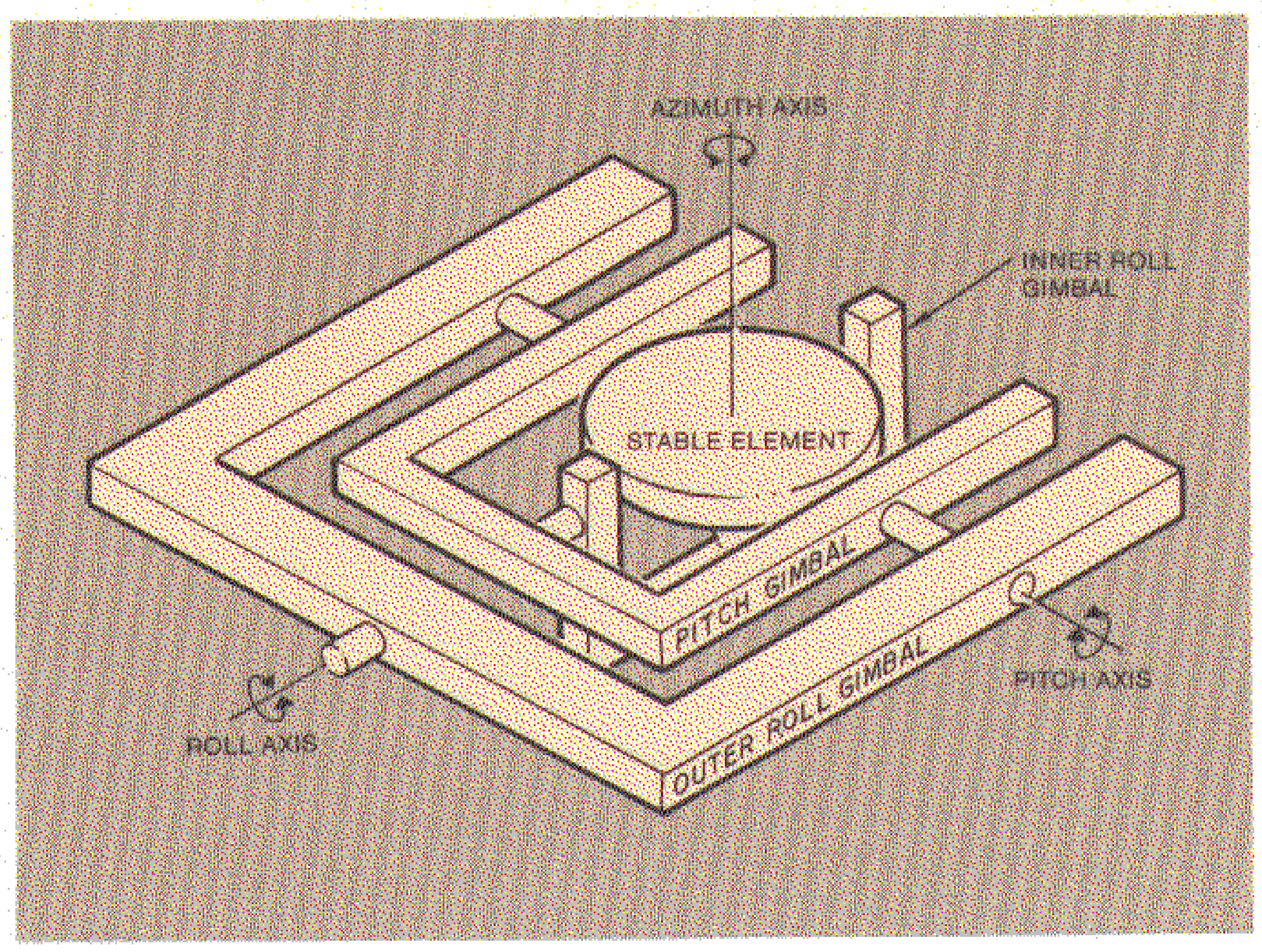


Figure 16. LTN-72 Inertial Navigation System Cantilevered Four-Gimbal Platform

identifies some of its internal components. The unit's gyroscopes and accelerometers are contained in an assembly called the "stable element." The platform consists of the stable element, mounted in a cantilevered, four-gimbal configuration (see Figure 16). The four-gimbal arrangement accommodates pitch, inner-roll, outer-roll, and azimuth. The redundant inner-roll gimbal provides the stable platform with the freedom of movement that is required to prevent the platform from tumbling when the aircraft attitude is at very high pitch angles.

The platform assembly is mounted in the forward portion of the Inertial Navigation Unit case, along with the electronic modules that operate the gimbal torquers, condition the accelerometer outputs, and provide temperature control. The rear portion of the INU contains the C-4000 general purpose computer, including the computer memory, power supply, and input/output (I/O) and additional support electronics (attitude and heading interface and the system condition monitors).

INU Interface The Inertial Navigation Unit's interface with other aircraft systems and equipment is shown in Figure 13. Aircraft attitude (pitch and roll) and heading are detected by synchro transmitters mounted in the INU gimbals (see Figure 8), and switched to the aircraft systems listed in Table IV. Digital data required for the P-3C CP-901/ASQ-114 Tactical Computer are transmitted via the LTN-72 Digital Data Unit (DDU).

Attitude and Heading Interface The aircraft pitch and roll attitude signals are generated in the inertial navigation unit by the synchro transmitters, one synchro on each gimbal. These analog electrical signals are amplified by the INU attitude repeater and split into five individual analog outputs. The P-3 installation requires only two of the available analog attitude outputs. These are called the Nos. 1 and 2 Pitch and Roll Attitude signals.

The No. 1 Pitch and Roll signals are used (when selected by the operator) for aircraft attitude by the Pilot's (INS No. 1) and the Copilot's (INS No. 2) AJN-15 Flight Director Indicators. The No. 1 Pitch and Roll Attitude outputs are dedicated to

TABLE IV. P-3C Aircraft/LTN-72 Inertial Navigation System Attitude and Heading Interface

PITCH AND ROLL NO. 1	PILOT AND COPILOT FLIGHT DIRECTORS AJN-15 FLIGHT DIRECTOR STEERING COMPUTER (FDSC)
PITCH AND ROLL NO. 2	ASW-31 AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS) APS-115 RADAR ANTENNA POSITION PROGRAMMERS (APP) DOPPLER RADAR NAVIGATION SYSTEM AWG-19 HARPOON AIRCRAFT COMMAND LAUNCH CONTROL SYSTEM (HACLS) CV-2461 SIGNAL DATA CONVERTER (SDC)
TRUE HEADING	PILOT, COPILOT, AND NAVIGATOR HORIZONTAL SITUATION INDICATOR (HSI) APS-115 RADAR APP AWG-19 HACLS LTN-211 OMEGA
MAGNETIC HEADING	PILOT, COPILOT, AND NAVIGATOR HSI VOR NAVIGATION SYSTEM TACAN NAVIGATION SYSTEM CV-2461 SIGNAL DATA CONVERTER LTN-211 OMEGA
MAGNETIC VARIATION	CV-2461 SIGNAL DATA CONVERTER

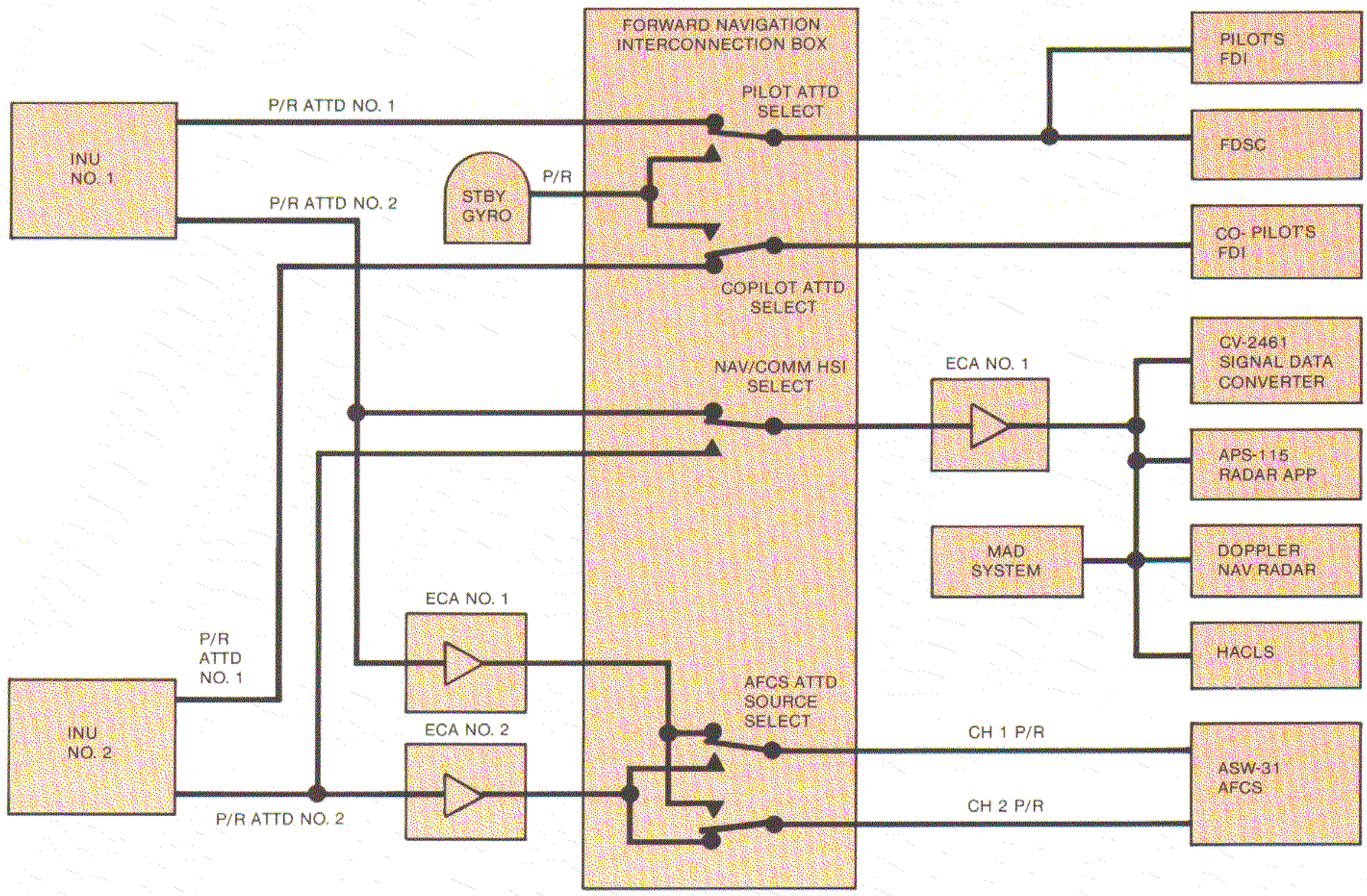


Figure 17. P-3C Aircraft/LTN-72 Inertial Navigation System Attitude Interface

the Flight Director Indicators in order to isolate the attitude signals from the affects of failures in peripheral equipment. The No. 2 Pitch and Roll analog signals supply aircraft attitude (as selected by the pilot or NAV/COMM) to the Automatic Flight Control System, Search Radar, Doppler Radar, MAD System, CP-901/ASQ-114 Computer (via the CV-2461 Signal Data Converter), and the Harpoon Aircraft Command and Launch System (HACLS). As shown in Figure 17, attitude signals provided to these systems are interfaced through the AM-4923/A Electronic Control Amplifier (ECA).

These operational amplifiers buffer the Pitch and Roll Attitude No. 2 analog output signals from the affects of the load that is imposed by the avionics that are in use. The two ECA units have twelve synchro repeater (amplifier) channels per box, and are located in Electronics Rack B1/B2.

In our discussion of inertial navigation fundamentals, we described the wander-azimuth operation of the LTN-72 Inertial Navigation Unit. Recall that for a wander-azimuth system, the *platform azimuth synchro transmitter* outputs an angle that is arbitrary but defined, and is stabilized gyroscopically. As such, we refer to this angle between the stable platform reference and the aircraft navigation centerline as the *platform heading* (see Figure 11).

The actual true heading of the aircraft is calculated by establishing the alpha (wander) angle during Inertial Navigation System alignment, and then summing it to the platform heading. Remember that the alpha angle changes constantly, and it is updated by rotating the computer axes as the aircraft changes latitude and longitude during flight. The calculated true heading is transmitted in a digital format to the CP-901 Computer via the Digital Data Unit and the Navigation Multiplexer in Logic Unit No. 2. The Inertial Navigation Unit also converts true heading from a digital representation to an analog signal and sends it to avionics systems and displays.

A diagram of the LTN-72 Inertial Navigation System analog heading interface is shown in Figure 18. The mechanical platform heading signal provides the INU computer with the number-of-degrees of heading change when the aircraft turns. The platform heading signal is also supplied as an input to the Magnetic Heading Reference System (MHRS) to provide that system with a stabilized heading change reference. The function of the MHRS will be discussed in detail later in this article.

The analog heading and attitude output signals are monitored continuously by the INU's general purpose C-4000 computer during inertial system operation. The computer's system condition monitors will detect failures of the attitude or heading synchros and their associated power amplifiers, or loss of 26 VAC synchro excitation. The LTN-72 Inertial Navigation Unit outputs the following logic discrete signals to annunciate failures to the aircraft systems:

- INS Primary Attitude Valid
- INS Auxiliary Attitude Valid
- INS Platform Heading Valid
- INS True Heading Valid

These discrete signals are 28 VDC when the system operating condition is valid, and they shift to a 0 VDC level when the system operating condition is invalid. They are combined in the Digital Data Unit (DDU) to generate the *Inertial Navigation System Valid* discrete signal. The INS Valid signal controls the GYRO flags on the Flight Director Indicators and provides system status to other avionics that use the INS synchro outputs (see Figure 19).

Digital Interface The LTN-72 Inertial Navigation System digital data output provides the CP-901 Computer with inertially-generated north/south velocity, east/west velocity, and true heading sig-

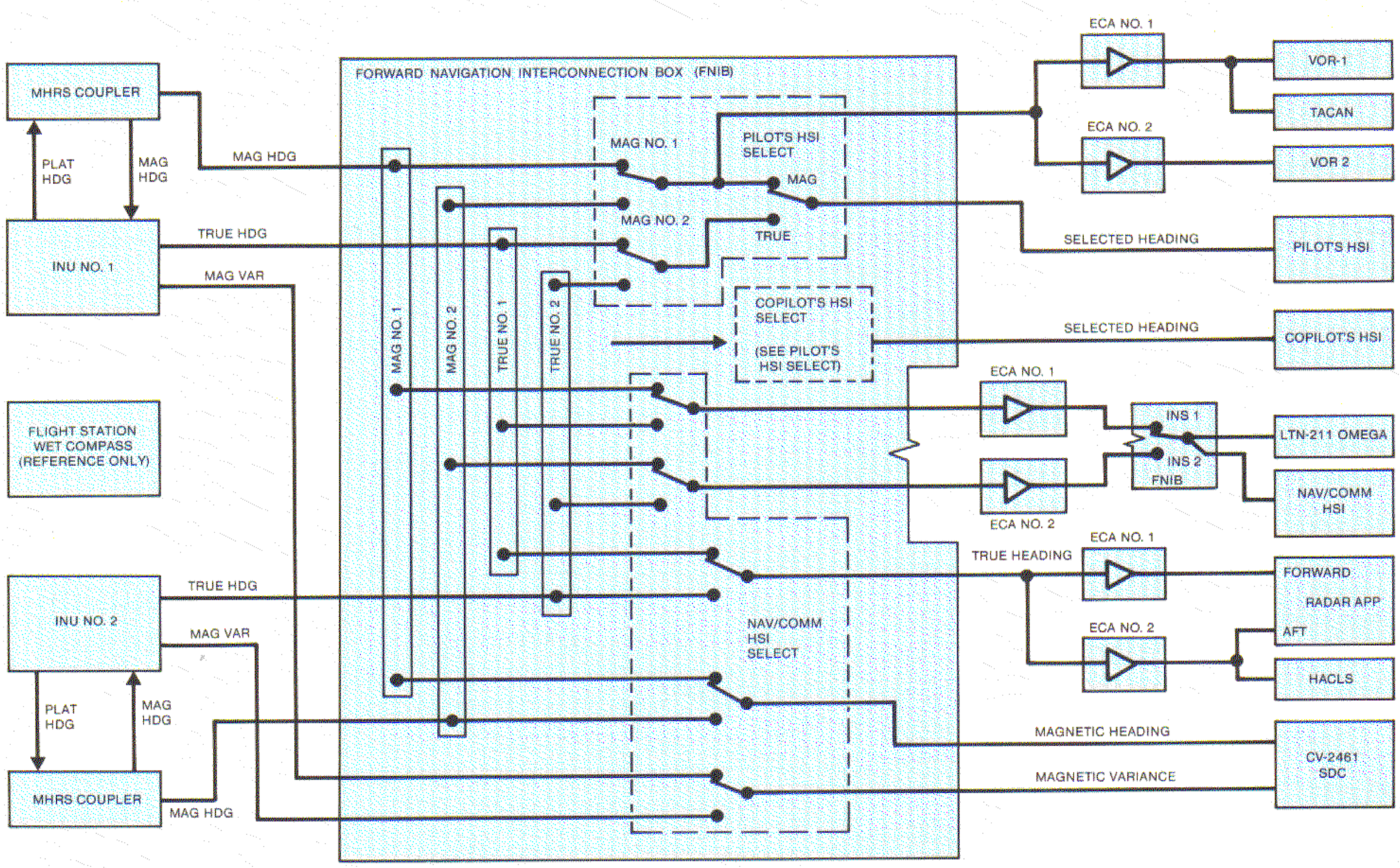


Figure 18. P-3C Navigation System Analog Heading Interface

nals. The data format available from the LTN-72 INU is specified by ARINC Characteristic 561, which facilitates use of this system in commercial aircraft. For the LTN-72 Inertial Navigation System to be compatible with the P-3C Digital Computer System, the inertial system's 32-bit ARINC data format must be altered to the aircraft system's 22-bit format. The Digital Data Unit receives the digital words transmitted by the Inertial Navigation Unit, and provides the three navigation parameters in the P-3C format to the Navigation Multiplexer in Logic Unit No. 2. In addition, the DDU monitors the INU Digital Valid discrete signal and combines it with the DDU BITE status signal to generate a system Digital Valid discrete signal. The operation of the DDU will be discussed further later in this article.

INS Input Interface The attitude, heading, and digital interface for the LTN-72 Inertial Navigation System was designed to be compatible with the existing ASN-84 Inertial Navigation System interface. However, there are two major differences between the LTN-72 and ASN-84 system requirements. The LTN-72 system provides the crew with wind speed-and-direction data that is not available from the ASN-84 system. In order to calculate wind data, the LTN-72 Inertial Navigation Unit is equipped with a true airspeed input. The CPK-28/A24G-9 True Airspeed Computer has a suitable synchro output that supplies the required signal to the INU Air Data input port.

The other significant difference between the LTN-72 and the ASN-84 Inertial Navigation System interface involves the barometric altitude input to the Inertial Navigation Unit that is used to correct arc-length in the dead-reckoning distance-traveled calculations. The ASN-84 system used digital-encoded barometric altitude signals from the copilot's altimeter to make these corrections. The LTN-72 Inertial Navigation Unit also has an input port that is used for this function in commercial applications. However, the requirement for the LTN-72 system to perform magnetic vari-

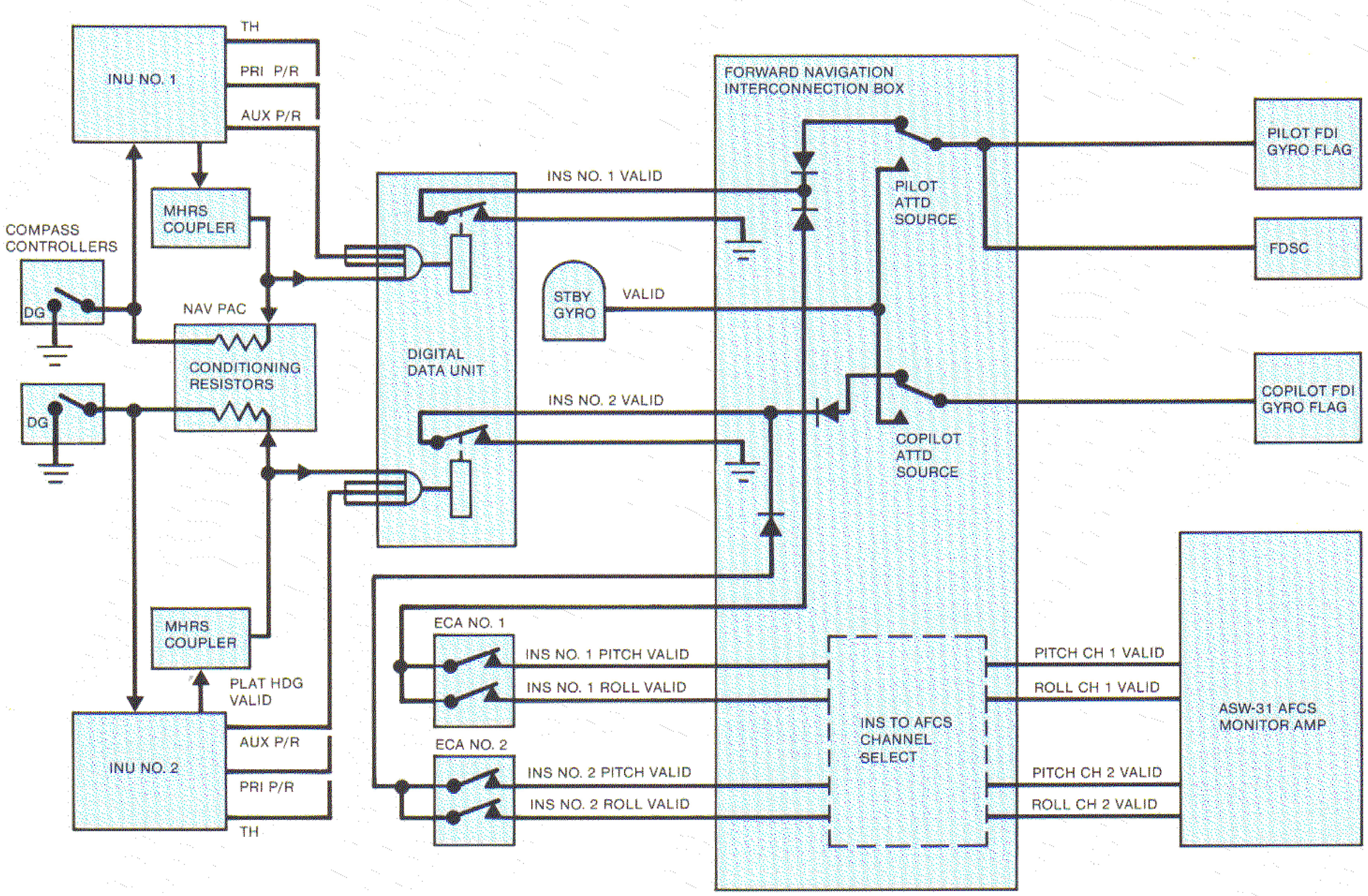


Figure 19. P-3C Aircraft/LTN-72 Inertial Navigation System Analog Validity Interface

ation calculations for the P-3C mission avionics made it necessary to use this port for a magnetic heading input. The arc-length correction term for the LTN-72 system has been incorporated in its software as a fixed 15,000 ft-altitude. The copilot's digital-encoded barometric altimeter signal is not used in the P-3C aircraft.

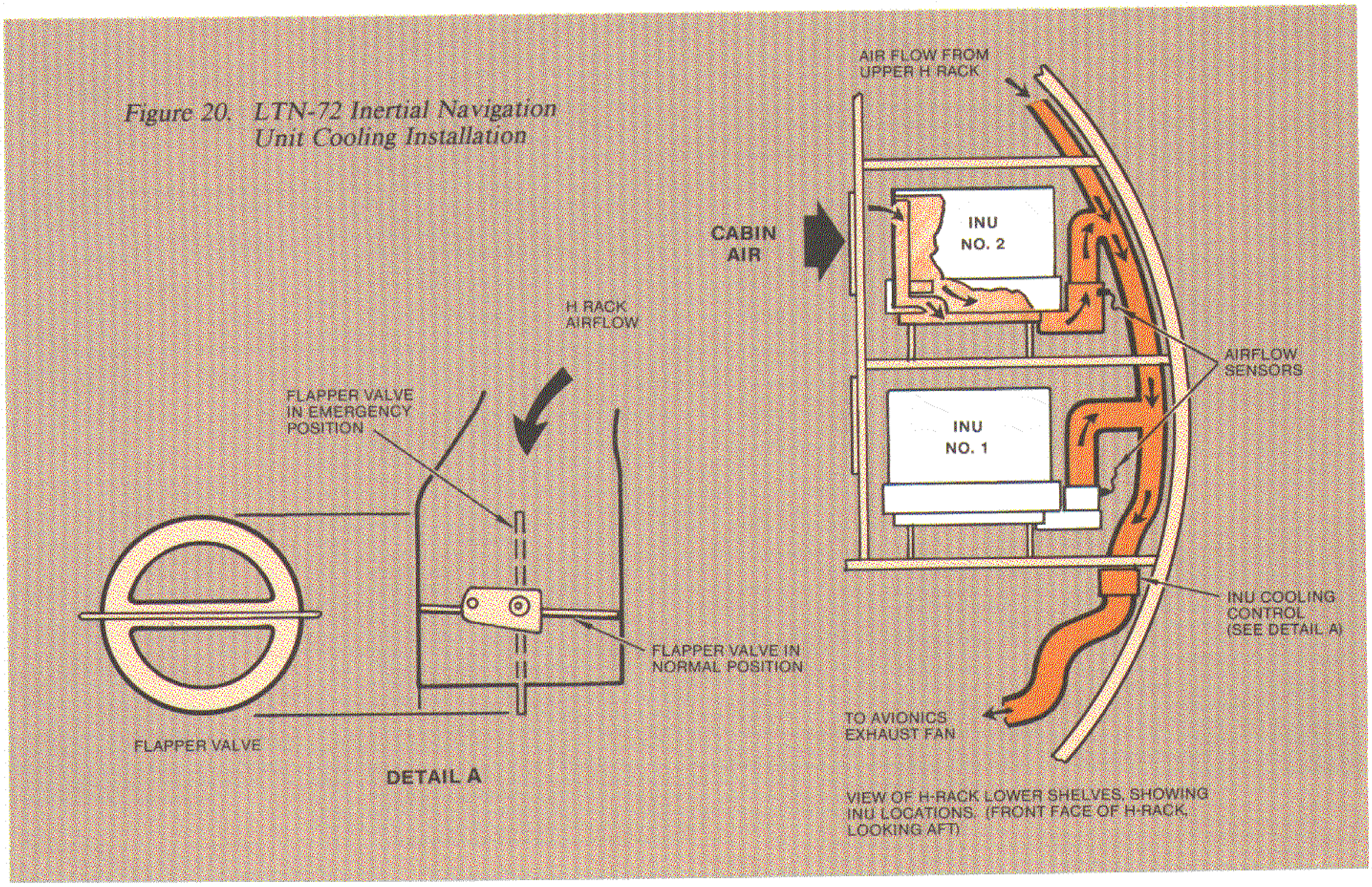
Temperature Control The Inertial Navigation Unit requires a controlled environment for its stable element to ensure that the platform's sensitive instruments (accelerometers and gyroscopes) operate accurately. During alignment, the first step is to heat the stable platform, gyroscopes, and the INU ambient environment to the proper operating temperature. In the following paragraphs we shall discuss the heating cycle and temperature control. We will discuss the details of the alignment sequence separately, later in this article.

The Inertial Navigation Unit's stable platform and compartment have three types of heaters. First, the *platform* is heated to 63 degrees C (145 degrees F) by heating elements that are mounted to the azimuth gimbal structure. Second, the *platform compartment* is heated by applying 115 VAC 400-Hz power to heating elements that warm the ambient air to 47 degrees C (117 degrees F). And third, the *two gyroscope assemblies* are maintained at an operating temperature of 76 degrees C (170 degrees F) by internal heating elements.

The INU temperature control system maintains the four main INU components (two gyros, the stable element, and the platform compartment) at the correct operating temperature by sensing the ambient temperature and automatically cycling the power to the heater elements.

The temperature in the Inertial Navigation Unit platform compartment is controlled automatically. This is achieved by balancing the heat supplied by the compartment's electric heaters with the cooling effect produced by the continuous circulation of ambient air in the platform compartment. The cooling air is circulated by two internal blowers, which move the air in the compartment past heat exchangers that are mounted on the sides of

Figure 20. LTN-72 Inertial Navigation Unit Cooling Installation



the Inertial Navigation Unit (see Figure 15). The heat exchangers are cooled by airflow from the aircraft cabin, created by operation of the Cabin Exhaust Fan. This configuration of air circulation reduces the chance of airborne contaminants collecting in the platform compartment. Figure 20 shows how the cooling airflow circulates in Electronics Rack H1/H2. This figure also shows that the cooling air plenums on the bottom of the INU mounting trays are plumbed directly into the rack exhaust duct.

Originally, the Inertial Navigation Unit installation called for additional fans to exhaust the INU cooling air into Electronics Rack H1/H2. This air would then have been removed from the rack by exhaust tuskers plumbed to the Cabin Exhaust Fan. When the design was reviewed, it became evident that the Cabin Exhaust Fan would remove all of the heat from the rack, making the INU fans redundant. Therefore, the INU fans were deleted, and the plenums were plumbed direct to the rack exhaust. This resulted in a cooler environment for the H1/H2 rack, and avoided the reliability problems normally associated with rack exhaust fans. However, this configuration did generate the need to monitor the airflow through the H1/H2 rack because of dependency on the Cabin Exhaust Fan to create air flow. The airflow monitoring function of the Inertial Navigation System will be discussed later in the section of this article that deals with the Navigation Power Alarm Control (NAV PAC).

An additional feature of the INU cooling air installation is the INU Emergency Airflow Control. This control is operated manually by moving a lever mounted on the front face of Electronics Rack H1. As shown in Figure 20, when the lever is in the NORMAL position, a ported plate (flapper valve) is positioned in the "H" rack exhaust duct to throttle the airflow. This exhaust duct is sized to provide adequate airflow to cool the INU if cabin exhaust fan operation degrades or fails. When the cabin exhaust fan is operating normally, the size of the "H" rack exhaust duct will allow airflow through the rack that exceeds INU cooling requirements. Although this would have no adverse effect on equipment in the "H" rack, it would make less cooling air available to other avionics racks in the aircraft and expose the equipment installed in these racks to higher ambient temperatures.

The ports in the Emergency Airflow Control Valve (flapper valve) throttle the exhaust airflow from the "H" rack down to the actual exhaust airflow called for in the equipment specifications. If a failure occurs that degrades the cooling for the Inertial Navigation Unit, the cooling airflow through the "H" rack can be improved by moving the Emergency Airflow Control lever to the EMERGENCY position. This repositions the valve flapper to remove the airflow restriction ports from the exhaust airflow, thus ensuring that the maximum amount of cooling air available will circulate through the "H" rack. Under these circumstances, the additional cooling airflow reduces the possibility that the INUs will be subjected to excessive temperature. This, in turn, reduces the potential that either inertial system will be forced to shutdown automatically because of overtemperature operation.

Temperature Monitors The Inertial Navigation Unit power supply will be shutdown automatically if the stable element temperature exceeds 88 degrees C (190 degrees F), if the platform compartment ambient temperature exceeds 71 degrees C (160 degrees F), or if the power supply temperature exceeds 84 degrees C (183 degrees F). If this occurs on the ground, INS operation can be restored by turning the system's Mode Selector Unit (MSU) control switch OFF, and then positioning the switch to ALIGN to recycle the system's alignment sequence.

If the temperature in the Inertial Navigation Unit's digital electronics assembly exceeds 79 degrees C (174 degrees F), the INU will shut down its digital power supply and the system will revert to operating in the Attitude Reference (ATT REF) mode for the remainder of the flight. During operation in the ATT REF mode, the Inertial Navigation System provides *only* pitch, roll, and platform heading information. The navigation capability of the INU is disabled for the remainder of the flight.

CONTROL DISPLAY UNIT The LTN-72 Inertial Navigation System Control Display Unit (CDU) displays navigation data, permits the NAV/COMM operator to enter the present position and waypoint coordinates, and provides control of track leg functions. In the P-3C aircraft, two CDUs are installed at the NAV/COMM station, one for

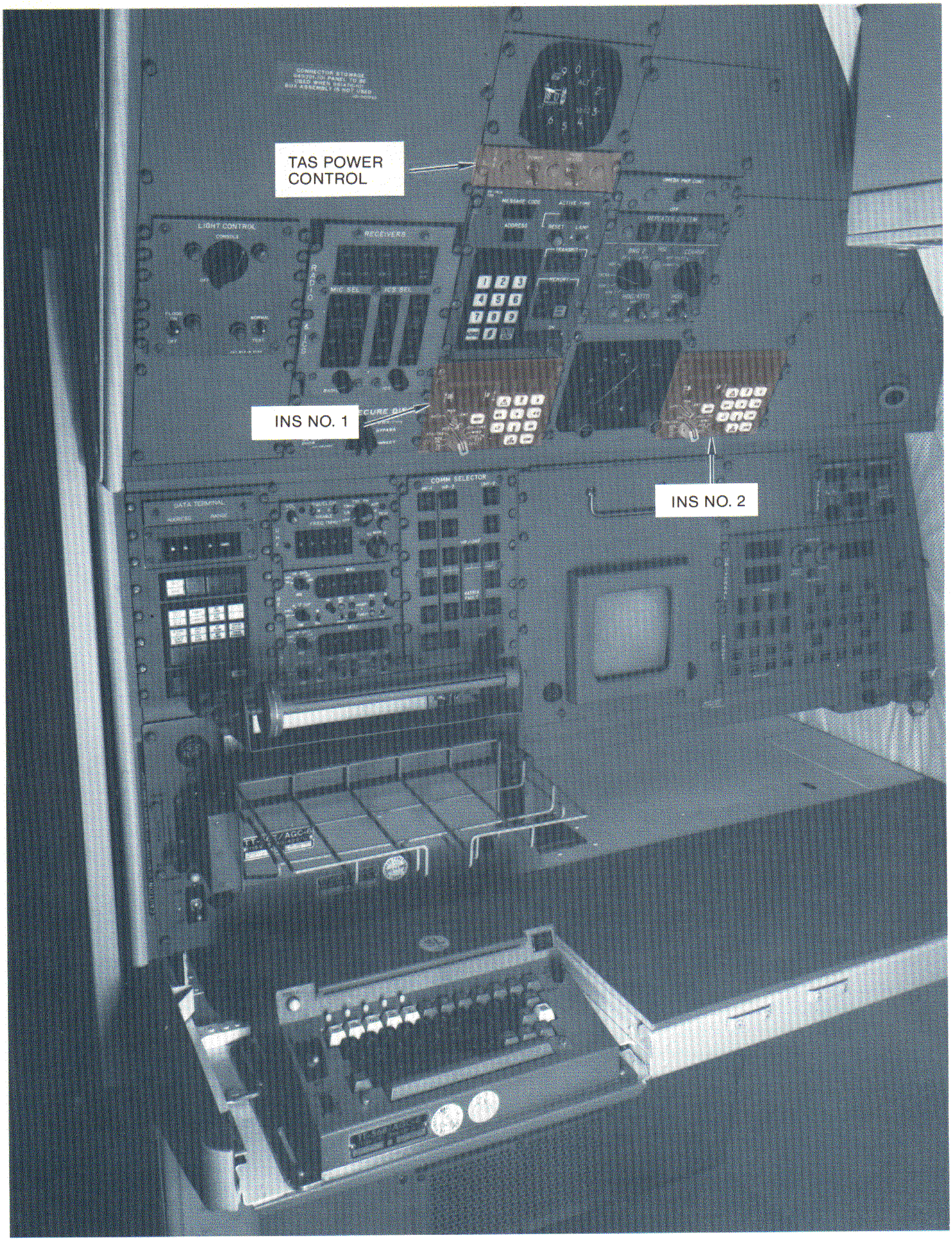


Figure 21. P-3C NAV/COMM Station, Showing LTN-72 Control Display Units

each of the two Inertial Navigation Systems (see Figure 21).

The Control Display Unit keyboard, shown in Figure 22, is used to enter the aircraft position coordinates into the Inertial Navigation Unit's computer. System alignment initialization, waypoint entry, and in-flight position updates are all performed with the CDU keyboard. During the position entry procedure, the operator checks the entered data by observing the left and right numerical displays on the CDU. Position latitude and longitude are keyed-in and displayed with resolution to a tenth of an arc-minute.[11]

The LTN-72 Inertial Navigation System makes extensive en route navigation information available to the flight crew. For example, the INS can store up to nine waypoints, monitor the aircraft position on the present track, and predict navigation data for subsequent track legs along the flight plan.

The Control Display Unit Selector Switch gives the crew access to select and display the following navigation data.[12]

- XTK/TKE (Cross-Track Distance/Track Angle Error) — Displays on the left display the cross-track distance (L or R) that the aircraft is displaced from the desired track, and displays on the right display the track-angle error (L or R) that the aircraft is displaced from the desired track.

- DIS/TIME (Distance/Time) — Displays the distance-to-go (on the left display) to the currently-selected waypoint, and the time-to-go (on the right display) to the waypoint.

- DSR TK STS (Desired Track Status) — Displays the desired track angle (on the left display) between selected waypoints. Displays status during the alignment sequence on the right display.

- HDG DA (Heading-Drift Angle) — Displays the aircraft heading on the left display, and the aircraft drift angle (L or R) on the right display.

[11] One arc-minute of latitude is equivalent to one nautical mile, and one arc-minute of longitude is equal to one nautical mile times the cosine of the latitude.

[12] Distances and speeds are in nautical miles and knots, respectively.

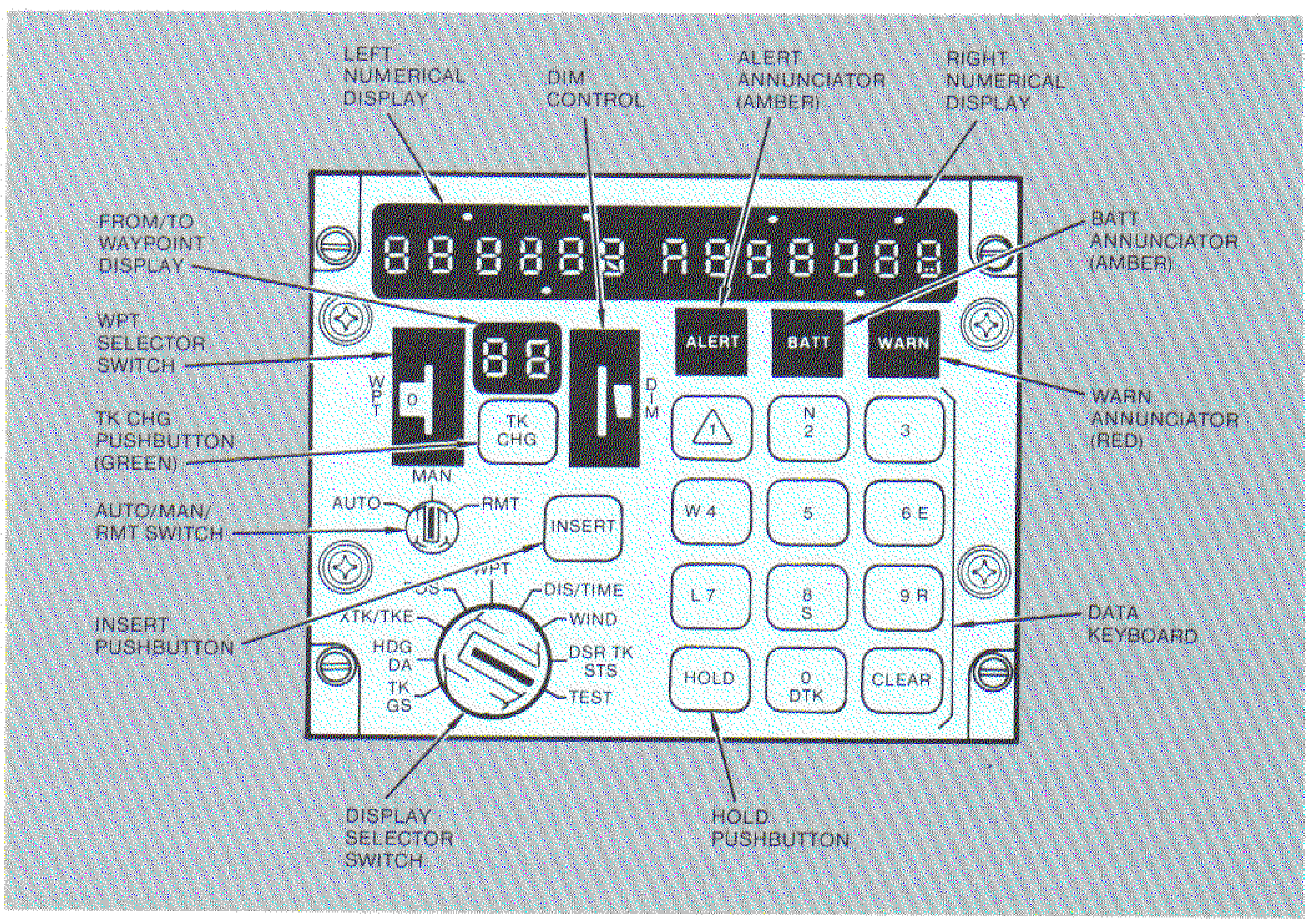


Figure 22. LTN-72 Inertial Navigation System Control Display Unit Panel

- POS (Present Position) — Displays the latitude of the aircraft on the left display, and its longitude on the right display.

- TK GS (Track-Ground Speed) — Displays the aircraft track angle (with respect to true north) on the left display, and the ground speed on the right display.

- WPT (Waypoint) — Displays the latitude (on the left display) and longitude (on the right display) of up to nine stored waypoints, corresponding to the waypoint digit on the WPT selector.

- WIND — Displays the wind direction (on the left display), and the wind speed (on the right display). The LTN-72 system software has a function that allows the operator to set a threshold true airspeed, below which no wind information will be calculated or displayed. This function is called Manual Wind Blanking, and its use is detailed in the P-3 NATOPS Flight Manual.

The Control Display Unit Selector Switch also has a TEST position, which enables the NAV/COMM operator or maintenance technician to test both the INU digital system and the CDU displays and lamps.

The Control Display Unit's AUTO/MAN/RMT selector switch establishes the operational mode of the Inertial Navigation System's en route function:

- AUTO (Automatic) — The track leg navigation calculations will automatically shift to the next sequential track leg as a waypoint is passed. The track leg FROM/TO display will be updated automatically. The CDU ALERT light will illuminate two minutes before the aircraft arrives at the track-change waypoint and goes out when the Inertial Navigation System track change is made. (The ALERT light will not illuminate if the aircraft ground speed is less than 125 knots.)

- MAN (Manual) — Track-leg calculations will change when the TK CHG (Track Change) push button is pressed. The CDU ALERT light will illuminate two minutes before the aircraft reaches the track change waypoint, as in the AUTO mode. The light flashes 30 seconds before the waypoint, and it will continue to flash if the waypoint is passed without the operator making a track change.

- RMT (Remote) — This function allows the operator to assess the remaining legs of the flight plan without disturbing the present en route navigation calculations. This selection is also used for the semiautomatic autofill function, in which data entry to one LTN-72 Inertial Navigation System is transferred automatically to the other system. In the P-3C aircraft, this function can also be used to display calculated magnetic variation. (Position the CDU Selector Switch to DSR TK/STS.)

In addition to the amber track-change ALERT light, the Control Display Unit has an amber BATT annunciator and a red WARN annunciator. The BATT light illuminates when the LTN-72 Inertial Navigation System is operating on its battery pack. The WARN light is illuminated to annunciate that the INS is malfunctioning.

Lockheed ORION Service Digest

MODE SELECTOR UNIT The Mode Selector Unit (MSU) provides operator control of power initialization and of the operating mode of the Inertial Navigation System. Each INS is equipped with an MSU installed in the flight station. The INS No. 1 Mode Selector Unit is installed in the pilot's side console, and the INS No. 2 Mode Selector Unit is mounted in the copilot's side console.

The Mode Selector Unit switch is armed with 28 VDC power generated by the Inertial Navigation Unit power supply. It is armed whenever *both* of the following two conditions are met: (1) the Iner-

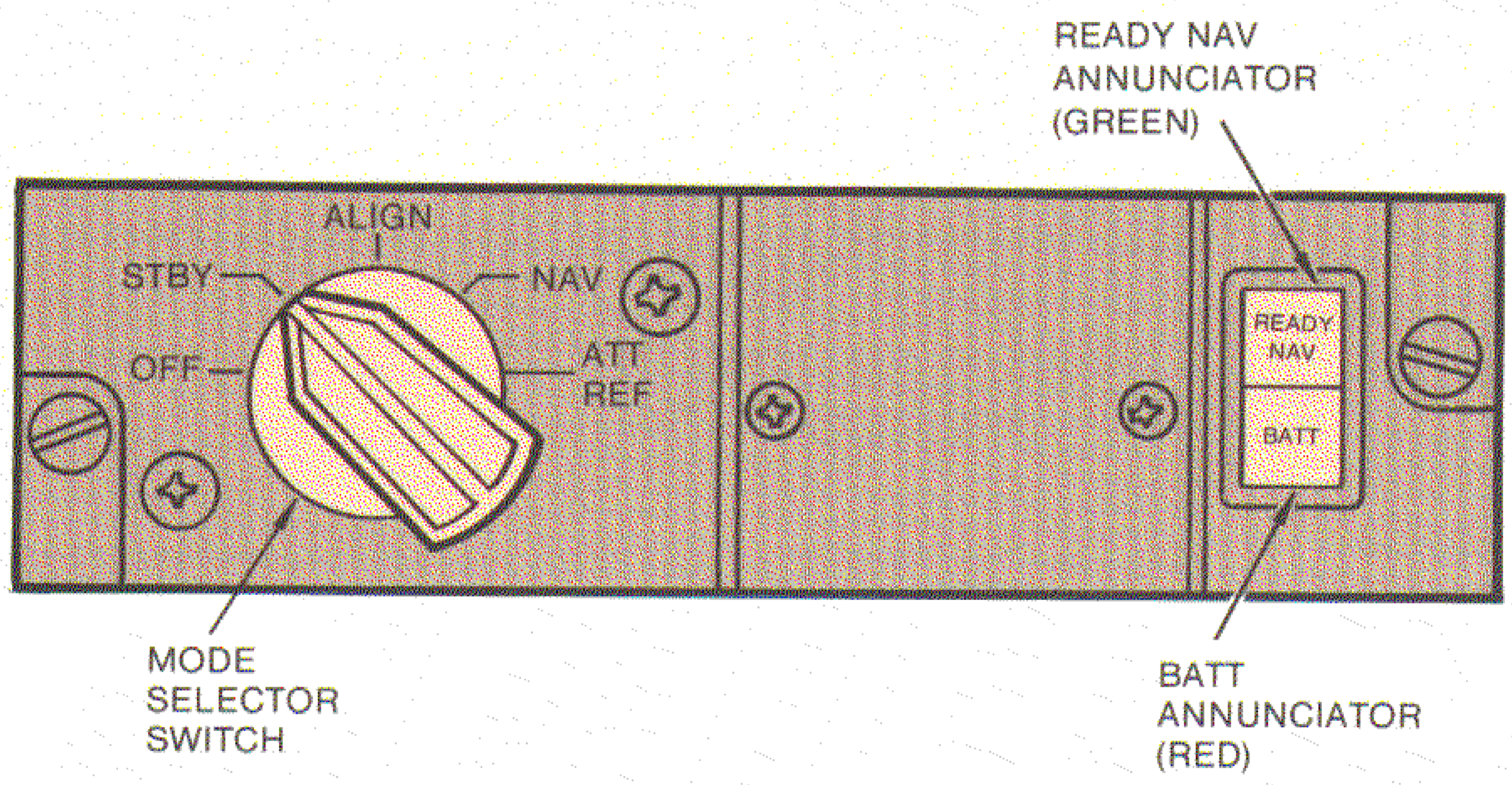


Figure 23.
Inertial Navigation
System Mode
Selector Unit Panel

tial Navigation System back-up battery is installed, and charged to at least 17.5 VDC; and, (2) the power supply is energized by 115 VAC 400-Hz aircraft power. The Mode Selector Unit is shown in Figure 23.

The Mode Selector Unit is also equipped with two annunciator lights. The green RDY NAV annunciator will illuminate when ALIGN is selected *and* when the system alignment sequence is complete. This indicates that the Inertial Navigation System is ready to operate in the NAV mode. The red BATT annunciator will illuminate when the INS is operating on back-up battery power, *and* the back-up battery voltage has been depleted below 17.5 VDC (less than the minimum voltage required to operate the system). The affected inertial navigation system will shut down automatically if this occurs.

DIGITAL DATA UNIT One of the basic design goals of the P-3C/LTN-72 Inertial Navigation System installation was to integrate it with the existing aircraft navigation system with a minimum of modification. The main problem was to interface the LTN-72 system's commercial ARINC 561 digital data format[13] with the aircraft's existing input Navigation Multiplexer (NAVMUX) in Logic Unit No. 2. The Inertial Navigation Unit's true heading and velocity outputs (which are used by the CP-901/ASQ-114 Tactical Computer to perform dead-reckoning calculations) had to be converted from a 32-bit serial word (label, data and sign) to an equivalent 22-bit serial word for input to the Logic Unit's NAVMUX. The Digital Data Unit (DDU) was developed to provide this conversion. Figure 24 shows the DDU installed in Electronics Rack D1.

[13]The format and characteristics of the digital data from the INU are described by ARINC Report 419-3 (Group C). The LTN-72 Inertial Navigation Unit provides both binary and binary-coded decimal output ports. The P-3C's Digital Data Unit receives data in the binary format.

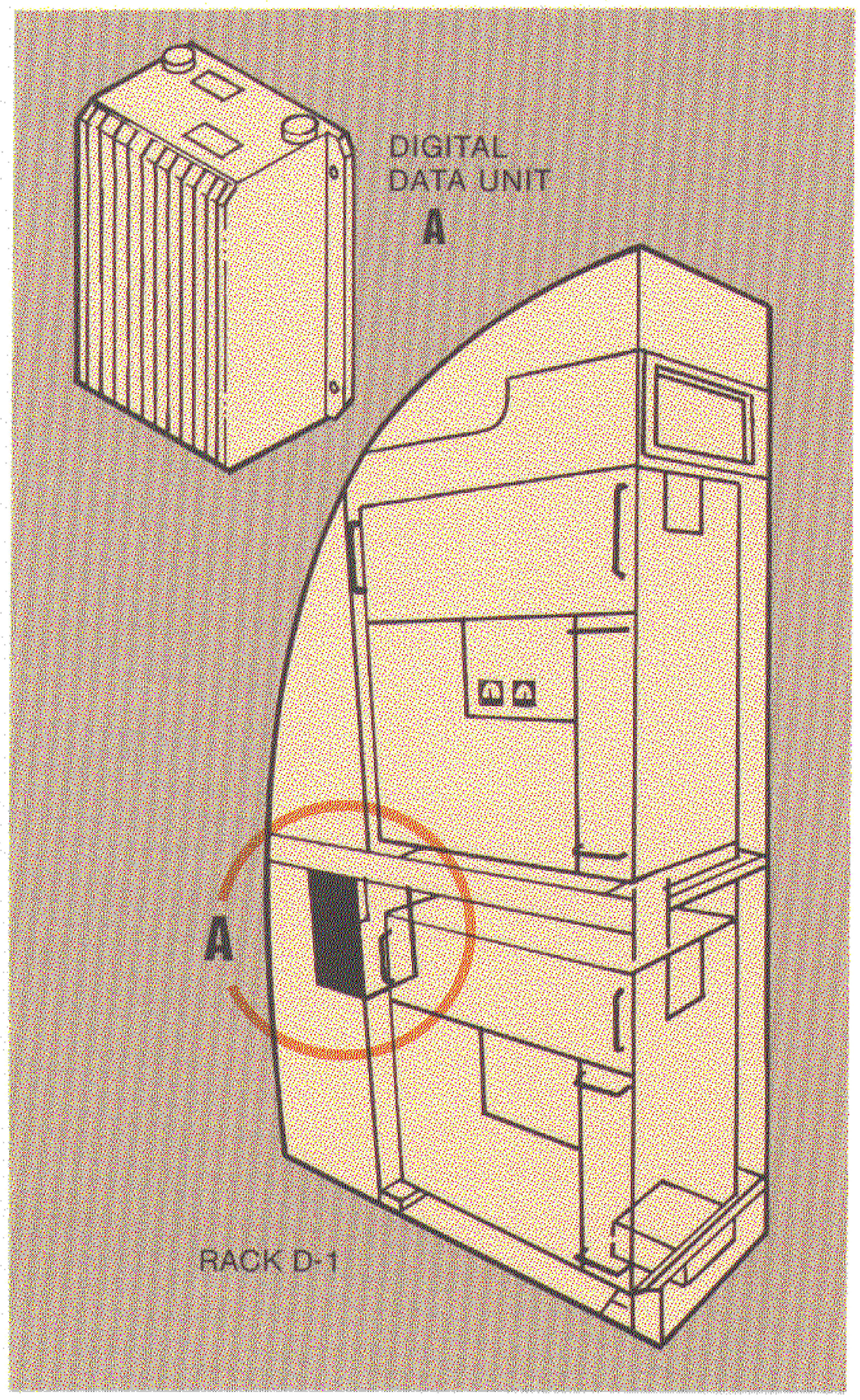


Figure 24. LTN-72 Inertial Navigation System Digital Data Unit Installation

The Digital Data Unit provides four basic functions for each of the two LTN-72 Inertial Navigation System installations:

- The digital data interface between the Inertial Navigation Unit and the Logic Unit No. 2 Navigation Multiplexer.
- Provides the LTN-72 Inertial Navigation System's operational mode status to the CP-901/ASQ-114 Tactical Computer via Logic Unit No. 1.
- Combines the Inertial Navigation System validity logic to generate a single SYSTEM VALID signal for the avionics that use INS analog (attitude and heading) outputs.
- Combines the Inertial Navigation Unit's Digital VALID discrete signal with the Digital Data Unit's VALID signal to generate a discrete signal for SYSTEM DIGITAL VALID.

Figure 25 is a functional block diagram of the Digital Data Unit interface for INS/Channel No.1.

The Digital Data Unit is a two-channel device. Channel No. 1 provides the full-time interface for INS No. 1, and Channel No. 2 provides the interface for INS No. 2. The two data channels operate independently, and do not transfer data from one channel to the other. The DDU design requirements and functions are:

- The two identical data channels are electrically isolated.
- The power source is 28 VDC aircraft power.
- After a power interrupt, the system will resume normal operation with no operator action.
- Both channels of the unit accept three navigation parameters from the Inertial Navigation Unit, and store these data in an ARINC BINARY format.
- Both channels provide, upon demand, the three navigation parameters, and the necessary control to Logic Unit No. 2.

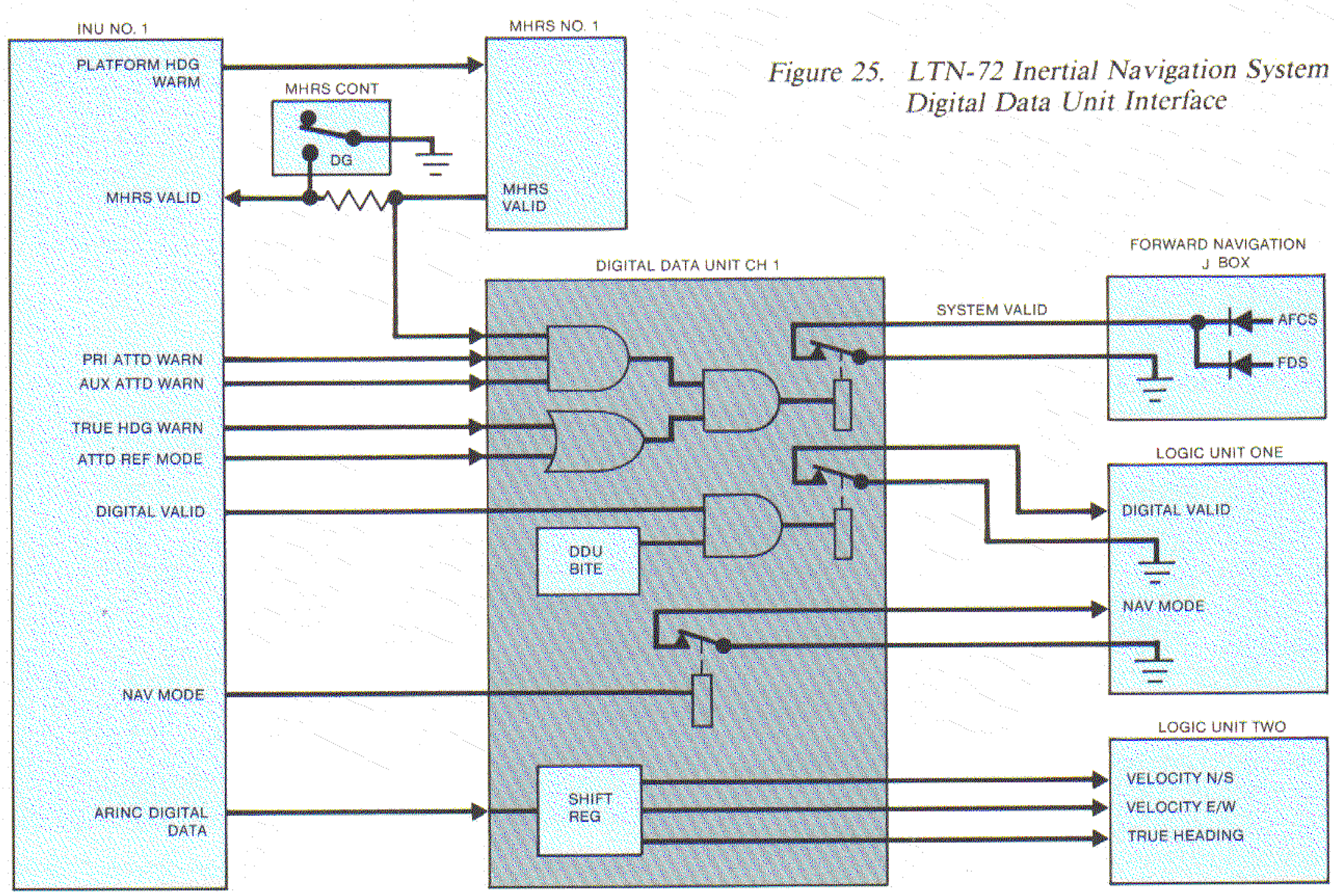


Figure 25. LTN-72 Inertial Navigation System Digital Data Unit Interface

- Each Digital Data Unit channel contains internal Built-in Test (BIT) circuitry that generates a DDU VALID discrete signal. The DDU also has the circuitry that combines this discrete signal with the INU digital valid discrete signal and generates a system digital status signal for the P-3C CP-901/ASQ-114 Tactical Computer via Logic Unit No. 1.

- Each Digital Data Unit channel combines the Inertial Navigation Unit attitude and heading validity discrete signals to create a SYSTEM VALID signal for the Forward Navigation Interconnect Box.

- The Digital Data Unit's input and data transfer functions are mutually independent. Data transfer from the Inertial Navigation Unit to the DDU does not affect (and is not affected by) data transfer between the DDU and Logic Unit No. 1 or Logic Unit No. 2.

The DDU wiring interface for digital data transfer is shown in Figure 26. The ARINC data use +12 volts to represent a logic-1 level, and 0 volt to represent a logic-0 level. The Logic Unit No. 2 input from the DDU requires that a logic-1 level be 0 volt, and a logic-0 level be +4 volts.

MAGNETIC HEADING REFERENCE SYSTEM One of the fundamental functions of the P-3 aircraft navigation system is to determine the aircraft's magnetic and true headings. Long-range geographic navigation (GEONAV) and Tactical Navigation (TACNAV) are referenced to true north. TACAN and VOR "en route" radio navigation aids are referenced to magnetic north. The Inertial Navigation System measures heading relative to the geographic latitude and longitude grid referenced to true north. This satisfies the heading requirements for the tactical navigation mode. The Magnetic Heading Reference System references aircraft heading to magnetic north to satisfy the aircraft's airways navigation requirements.

All P-3 aircraft are equipped with two remote-sensing magnetic heading systems and one standby wet compass. Each remote system uses an ML-1 Remote Compass Transmitter (magnetic flux valve) to detect the earth's magnetic field, and to

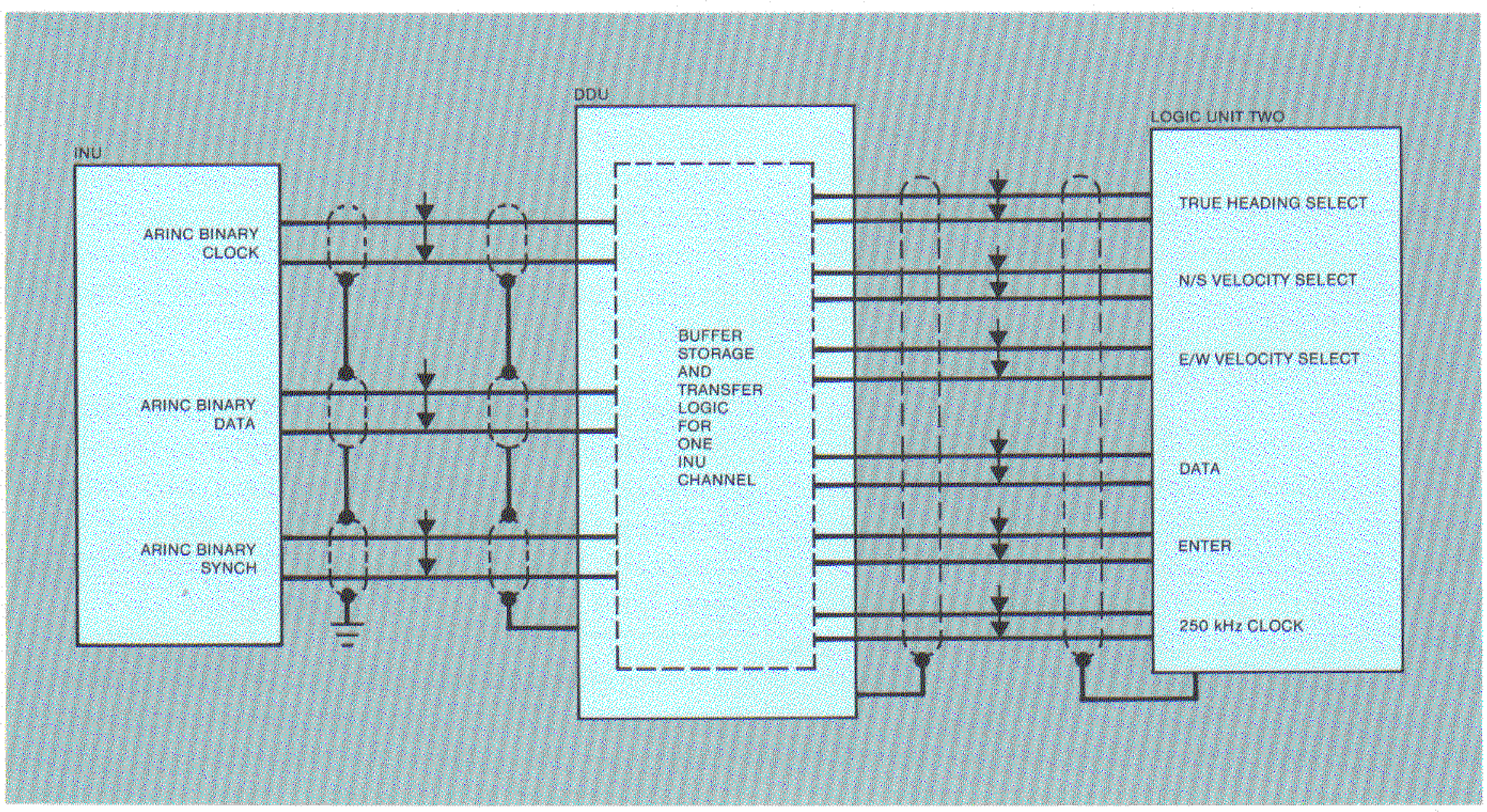


Figure 26. LTN-72 Inertial Navigation System/Logic Unit No. 2 Digital Interface

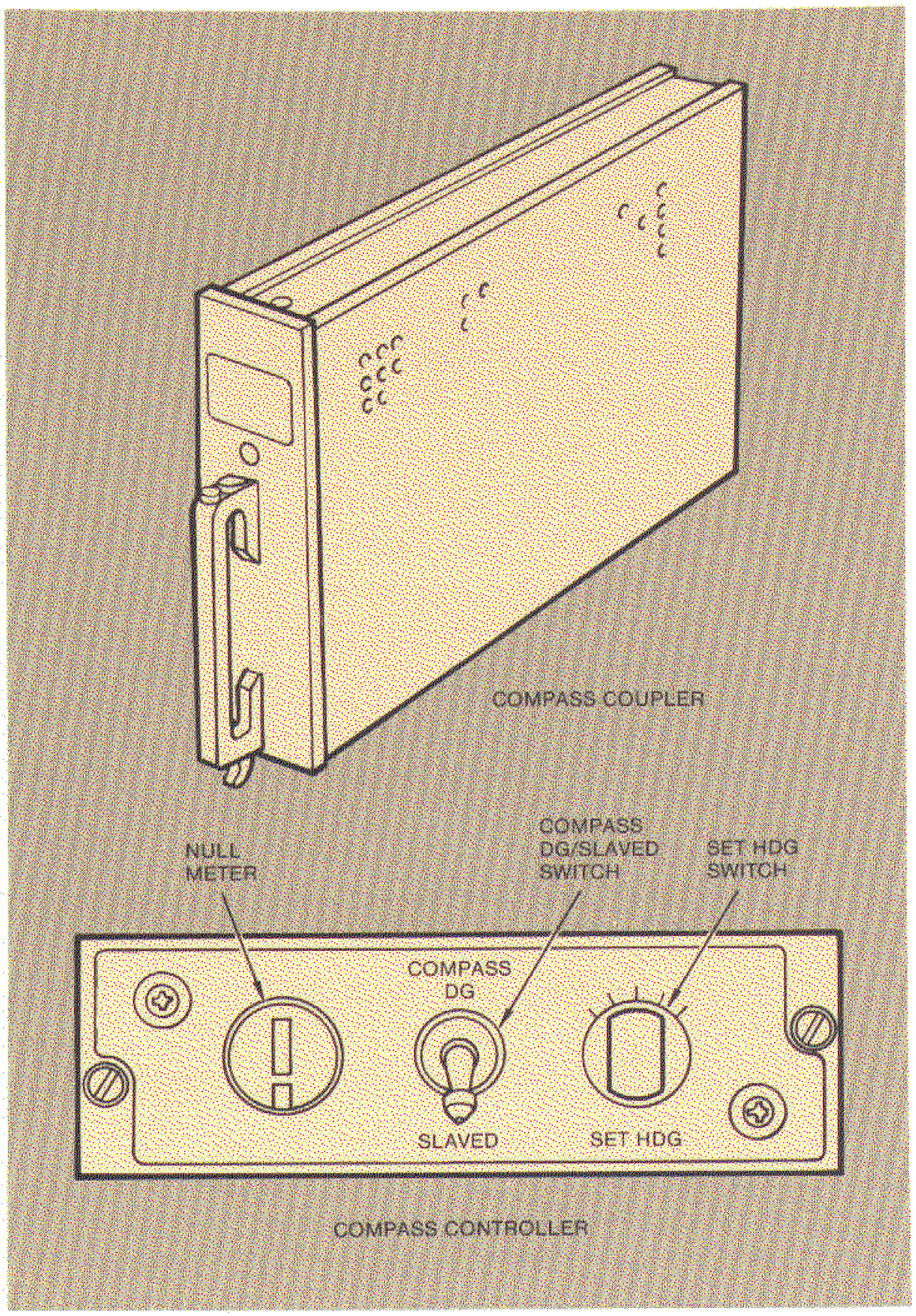


Figure 27. Sperry Magnetic Heading Reference System Components — Compass Coupler and Compass Controller

provide a raw magnetic heading signal to a compass coupler servo. The compass coupler combines the raw magnetic heading signal with the directional stability of an azimuth gyro. The output of the coupler device is an analog signal that is referenced to the navigation system 400-Hz excitation phase.

The ASN-42 Inertial Navigation System and the ASN-50 Attitude Heading Reference System, installed in production P-3A and P-3B model aircraft, were equipped with an integrated coupler mechanism. The two ASN-84 Inertial Navigation Systems, installed in P-3C production aircraft until 1979, also had an internal magnetic heading coupler. (Provision for the magnetic heading signal was an ancillary function of the ASN-84 inertial system.) The LTN-72 is a pure inertial navigation system and, as such, has no provisions for magnetic heading coupling. Consequently, P-3 aircraft fitted with the LTN-72 system were equipped with a Magnetic Heading Reference System (MHRS) to provide the aircraft magnetic heading signals. An off-the-shelf commercial MHRS, manufactured by Sperry Flight Systems, was selected to complement the LTN-72 Inertial Navigation System. Selection of the Sperry MHRS was based on its airline-proven reliability, and its compatibility with the ARINC 561 INS.

The Magnetic Heading Reference System has two components — the Compass Coupler and the Compass Controller (see Figure 27). The P-3C aircraft has two MHRS installations, one interfaced with each LTN-72 Inertial Navigation System. The MHRS Compass Couplers require 115 VAC Phase A power. The MHRS No. 1 is powered from the MONITORABLE ESSENTIAL AC BUS (along with the INS No. 1), and the MHRS No. 2 is powered from MAIN AC BUS A. There is no electrical interconnection between the two systems. The Compass Couplers are mounted in Electronics Rack H2 with the other Inertial Navigation System equipment.

The Magnetic Heading Reference System Compass Controllers are mounted in the flight station side consoles, adjacent to the Inertial Navigation System Mode Selectors. The MHRS operating mode is selected with the Compass Controller SLAVED/DG switch. In the SLAVED mode, the Compass Coupler tracks the ML-1 Flux Valve magnetic field sensor, and positions the heading output transmitters to provide the magnetic heading signal. During SLAVED operation, the Compass Controller NULL METER indicates if the Compass Coupler is synchronized to the magnetic flux valve. During operation in the DG (Directional Gyro) mode, the MHRS provides a heading output signal that is stabilized by the INS azimuth gyro.

Slaved Mode Operation of the Magnetic Heading Reference System Compass Coupler in the Slaved Magnetic Heading mode requires two external heading inputs (see Figure 28) — the flux valve magnetic heading signal and the Inertial Navigation Unit platform heading signal.

The coupler applies 23.5-volt, 400-Hz excitation to the flux valve. The raw magnetic heading signal from the flux valve is an 800-Hz, 3-wire signal that is supplied as a long-term, earth's-field heading reference to the Compass Coupler. This signal is subject to short-term errors that occur during aircraft maneuvering. Short-term errors are produced by the change in orientation (pitch and roll angles) between the flux valve and the earth's magnetic field. These short-term errors are eliminated by stabilizing the operation of the Compass Coupler with the platform heading signal from the Inertial Navigation Unit.

In our earlier discussion of inertial navigation theory, we stated that control of the platform is maintained by the Z-gyro. This produces a directionally-stabilized synchro signal that provides a turn-rate reference for the Compass Coupler. The turn-rate reference signal will dampen Compass Coupler operation, providing a stable Compass Card/HSI slew rate. The function of the Compass Coupler servomechanism is to combine the stabilized platform heading and flux valve inputs to properly position the coupler's magnetic heading output synchros (seven per coupler). Only two of the Compass Coupler's seven available output synchros are used in the P-3C LTN-72 Inertial Navigation System installation.

Figure 18 shows the Magnetic Heading Reference System interface with the aircraft flight instruments and navigation systems. The MHRS transmits a magnetic heading signal to the Forward Navigation Interconnection Box (FNIB) for distribution to the Horizontal Situation Indicators, to the VOR and TACAN Navigation equipment, and to the CP-901 Computer (via the CV-2461 Signal Data Converter). The excitation for this synchro signal is supplied from the FNIB 26 VAC Phase A INS synchro circuit breakers. Selection of MHRS No. 1 or No. 2 as the heading source for the various systems is accomplished in the FNIB. Crew control of FNIB switching is accomplished with the Horizontal Situation Indicator control panels located at the pilot, copilot, and NAV/COMM stations.

An additional Compass Coupler heading synchro output is used to supply magnetic heading to the Inertial Navigation Unit. The INU requires this magnetic heading signal to calculate the local magnetic variation for input to the CP-901 Tactical Computer (via the CV-2461 Signal Data Converter). A design problem surfaced in implementing this requirement that resulted in the magnetic variation calculation being dependent on the presence of 115 VAC power at the True Airspeed Computer. The following brief description of this problem and its solution may be helpful when isolating problems with magnetic variation on LTN-72 INS-equipped P-3 aircraft.

In commercial applications of the ARINC 561-type Inertial Navigation Unit, true airspeed and barometric altitude inputs are supplied to the

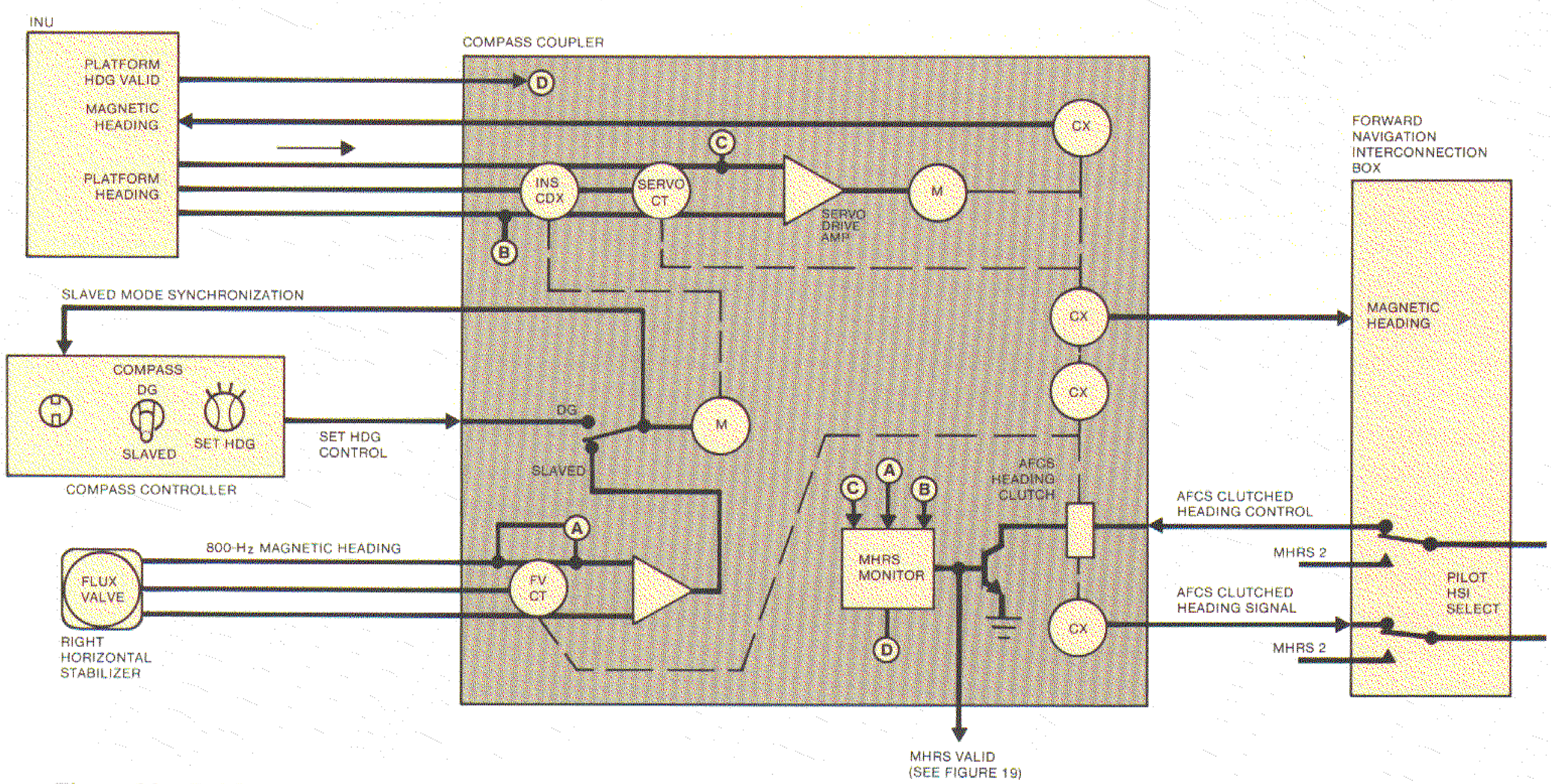


Figure 28. P-3C Magnetic Heading Reference System Functional Diagram

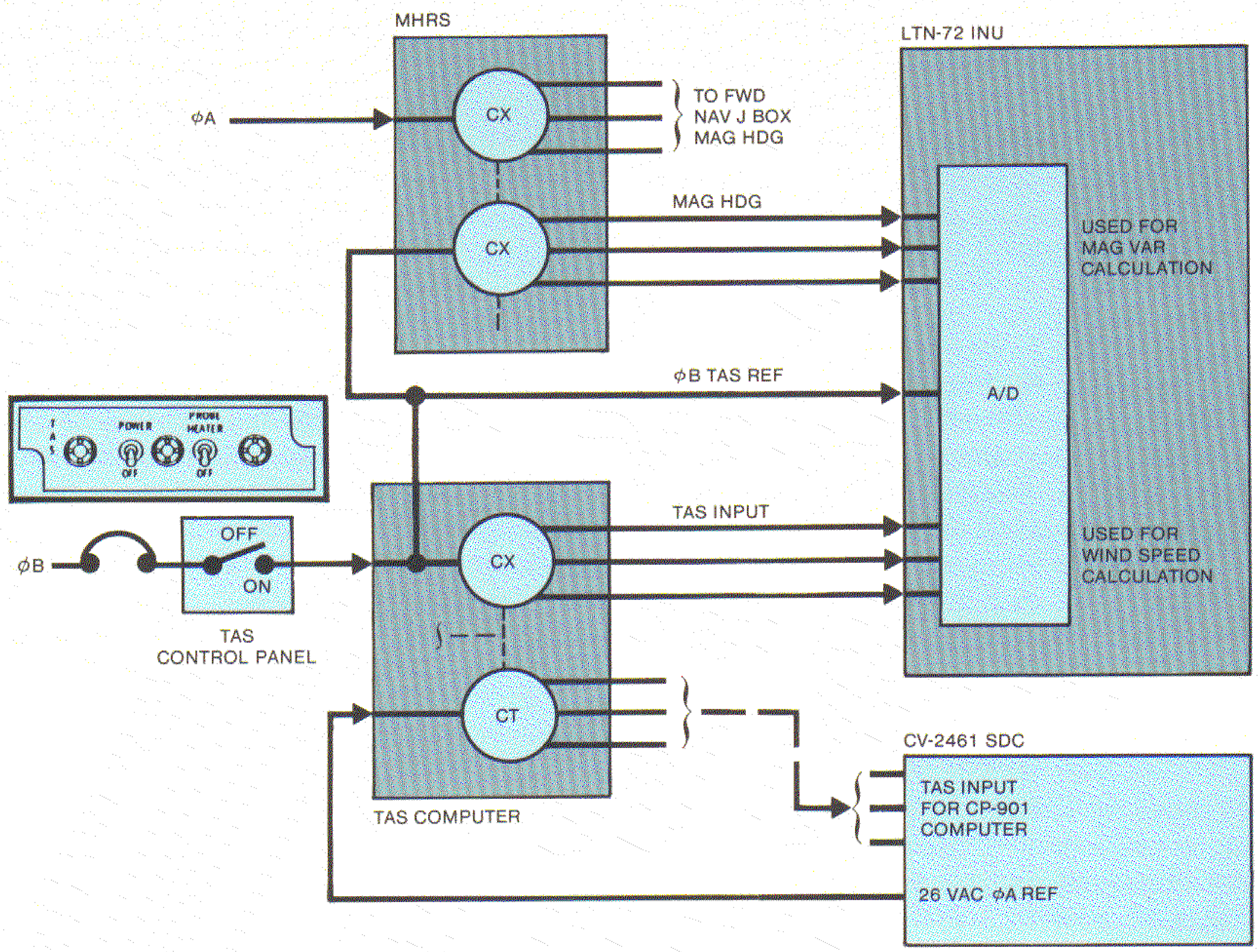


Figure 29. LTN-72 Inertial Navigation System/True Airspeed Computer System Interface

INU to enable it to calculate wind speed/direction and arc-length correction for distance traveled. The P-3C requirement to calculate magnetic variation in the INU required use of one of the Air Data input ports to receive the magnetic heading signal from the Magnetic Heading Reference System. The other INU Air Data input port receives an analog true airspeed signal from the TAS computer. Note that the TAS computer uses 115 VAC Phase B power, rather than the 115 VAC Phase A power used by the rest of the navigation system. The need to provide both magnetic heading and true airspeed signals to the INU, and the design goal to minimize the changes to existing aircraft systems led to supplying a 26 VAC Phase B reference signal (TAS power) to the INU air data excitation input (see Figure 29). As a result, the analog inputs to the INU, and the INU analog input receivers are all referenced to Phase B power. The application of power to this system is controlled by the TAS Power Control at the NAV/COMM station. If this switch is not turned on, the INU/MHRS will appear to function properly but no magnetic variation or wind information will be generated. *There is no cue to alert the operator that there is no TAS power.*

Directional Gyro Mode The Magnetic Heading Reference System Directional Gyro Mode is used for grid navigation techniques that are employed when the aircraft is operated at high latitudes. This function is activated by selecting the Compass Controller switch to DG, and slewing the Compass Coupler servomechanism with the SET HDG knob to a desired position shown on the Horizontal Situation Indicator compass cards.

During subsequent operation, the HSI will display the number of degrees of heading change during a turn. All Compass Coupler slaving to the magnetic flux valves is switched out.

The Magnetic Heading Reference System Compass Coupler also has a clutched synchro transmitter used to provide an autopilot heading hold signal for the ASW-31 or PB-20N Automatic Flight Control System. This clutch is an electromechanical device that couples the synchro rotor to the coupler magnetic heading servo when it is armed by the autopilot, and when the coupler monitor input is valid. The P-3 autopilot systems use only one clutched-heading input at a time, and the source (MHRS No. 1 or No. 2) is switched in the Forward Navigation Interconnection Box as selected by the pilot's Horizontal Situation Indicator control.

When the crew enables the *clutched heading hold* mode for the autopilot, 28 VDC is applied to the selected coupler clutch. If the Compass Coupler is operating properly, an internal ground is applied to the clutch, which engages the rotor shaft to the coupler servo. Any subsequent movement of the coupler servo will produce an output signal, from the synchro, that is proportional to the change in aircraft heading. In this mode, the autopilot will steer the aircraft to null the clutched-heading output. The heading hold output is available in both the SLAVED and DG modes, but the clutch will not engage during the crew-initiated SET HDG command.

MHRS Monitors The Magnetic Heading Reference System Compass Coupler monitor circuit continuously checks the validity of the inputs to the coupler and the null condition of the coupler's servo loop. Table V lists the specific MHRS monitor functions. If a malfunction is detected, the Compass Coupler validity discrete signal is changed from 28 VDC (VALID) to 0 VDC (INVALID). Figure 19 shows the application of the MHRS Valid discrete signal to the Digital Data Unit and the Inertial Navigation Unit. The DDU combines this valid status discrete signal with the INU status signals (Primary and Auxiliary Attitude Warn, Platform Heading Warn and True Heading Warn) to generate the INS System Valid discrete signal.

The MHRS Valid discrete signal is also monitored by the Inertial Navigation Unit. This en-

TABLE V. Magnetic Heading Reference System Monitor Functions

FUNCTION	PARAMETER	CONDITION VALID	CONDITION INVALID	FIG 28 REF
FLUX VALVE MAGNETIC HEADING INPUT	23.5 VAC FLUX VALVE EXCITATION	DETECTS FLOW IN FLUX VALVE EXCITATION WINDING CIRCUIT	LACK OF CURRENT FLOW IN WINDING	A
	COUPLER FLUX VALVE CONTROL TRANSFORMER (B9)	DETECTS CONTINUITY IN THE ROTOR OF THE B9 FVCT	LACK OF CONTINUITY	
PLATFORM DIRECTIONAL GYRO HEADING INPUT	INU PLATFORM HEADING SYNCHRO SIGNAL	SYNCHRO SIGNAL PRESENT AT COUPLER INPUT (INS CDX INPUT B11)	NO SIGNAL PRESENT	B
COUPLER SERVOMECHANISM	SERVO LOOP NULL	DETECTS SIGNAL NULL AT SERVO DRIVE. INDICATES ALL COUPLER SERVOMECHANISM COMPONENTS ARE FUNCTIONING	SIGNAL (AC) PRESENT AT SERVO IMPROPER HEADING SERVO POSITION	C
	SERVO LOOP CONTROL TRANSFORMER	DETECTS CONTINUITY IN ROTOR WINDING	DETECTS OPEN IN ROTOR WINDING	
PLATFORM HDG VALID	DISCRETE SIGNAL FROM INU	28 VDC LEVEL DISCRETE FROM INU INDICATING THAT PLATFORM HEADING OUTPUT IS VALID	0 VDC, INDICATING INVALID INU PLATFORM HEADING	D

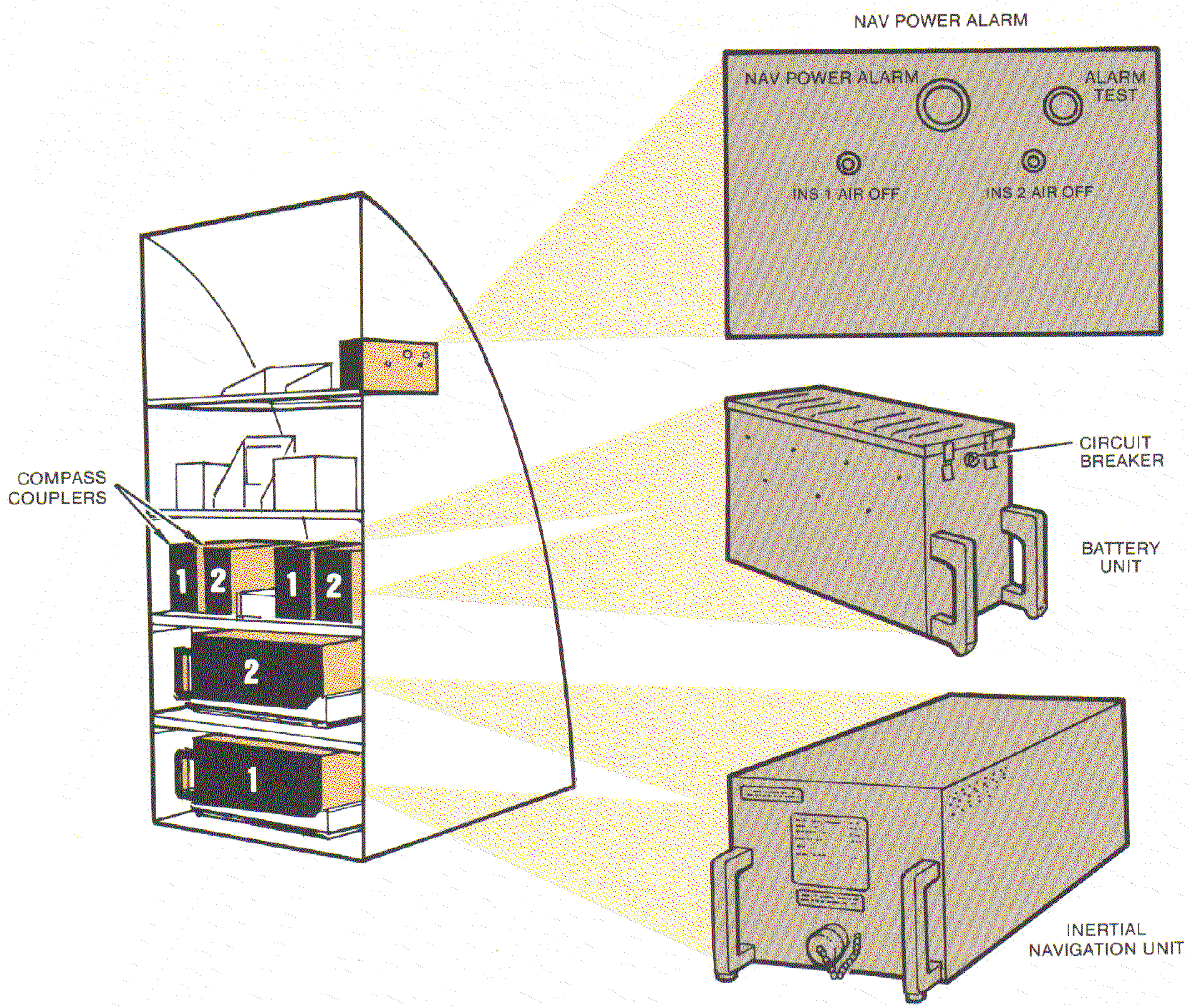


Figure 30. Navigation Power Alarm Control (NAV PAC), Inertial Navigation Unit and INU Battery Installation in Electronics Rack H1/H2

ables the INU to detect a coupler fault and display it to the crew. It also enables the INU to inhibit calculation of magnetic variation if the Compass Coupler magnetic heading signal is not correct, or when the MHRS is operating in the Directional Gyro mode.

The Directional Gyro Mode introduces an anomaly because the *coupler* heading output and status signal to the flight instruments is VALID, but the *magnetic* heading output to the Inertial Navigation Unit is INVALID. Correction of this problem (in the P-3C installation) required the addition of a conditioning resistor that shifts the MHRS VALID status for the INU to INVALID any time that the MHRS is in the DG Mode. Figure 19 shows that the conditioning resistor grounds only the discrete signal to the INU; the VALID discrete signal to the DDU is still at 28 VDC. Also, when the INS is operated in the NAV Mode with the MHRS in the DG Mode, the operator will receive a false MHRS failure warning at the system CDU.

NAVIGATION POWER ALARM CONTROL As the P-3 installation requirements for the ARINC 561-type Inertial Navigation System evolved, it became apparent that several external control and alarm functions were required. In the final design, three functions were defined:

- Sense and annunciate the absence of adequate airflow to cool the two Inertial Navigation Units.

- Control of power application to the Digital Data Unit.

- Condition the MHRS VALID discrete signal to the Inertial Navigation Unit when the MHRS is operating in the Directional Gyro Mode.

The A509 Navigation Power Alarm Control (NAV PAC) was developed to implement these functions.

The requirement to condition the MHRS Valid discrete signal to the Inertial Navigation System when the Magnetic Heading Reference System is operating in the Directional Gyro Mode was discussed in the previous section of this article. The two resistors that perform this function are located in the Navigation Power Alarm Control.

The airflow sensing and warning function is necessary because the Inertial Navigation Units depend upon the cabin exhaust fan to draw air through Electronics Rack H1/2 to cool the INUs. If the cabin exhaust fan fails and the INUs continue to operate, the units may shut down or they may be damaged by overheating. The manual INU EMERGENCY COOLING control can be used to increase the airflow through Electronics Rack H1/2 during flight if sufficient airflow is not present to cool the INUs. However, it is also necessary to alert the crew that this condition exists. The NAV PAC alarm performs this function.

Figure 30 shows the NAV PAC alarm panel on the front face of Electronics Rack H1. The airflow warning consists of two light-emitting diodes (one per INS) and an audio alarm. Figure 31 shows the NAV PAC circuitry. The airflow through the INUs is detected by sensors in the INU plenum exhausts (see Figure 20). They provide a ground to the NAV PAC if either INU experiences *insufficient* cooling airflow for more than 10 seconds. (Note that although these sensors detect airflow, they are not sensitive to the air temperature.) The INS cooling airflow sensing and warning circuit is tested by pressing the PUSH TO TEST switch on the front of the NAV PAC. This grounds both airflow sensor inputs, which illuminates the two light-emitting diodes and sounds the aural beeper. The warning function is powered by the INU 28 VDC power supply and/or the INS battery when the inertial system is turned on.

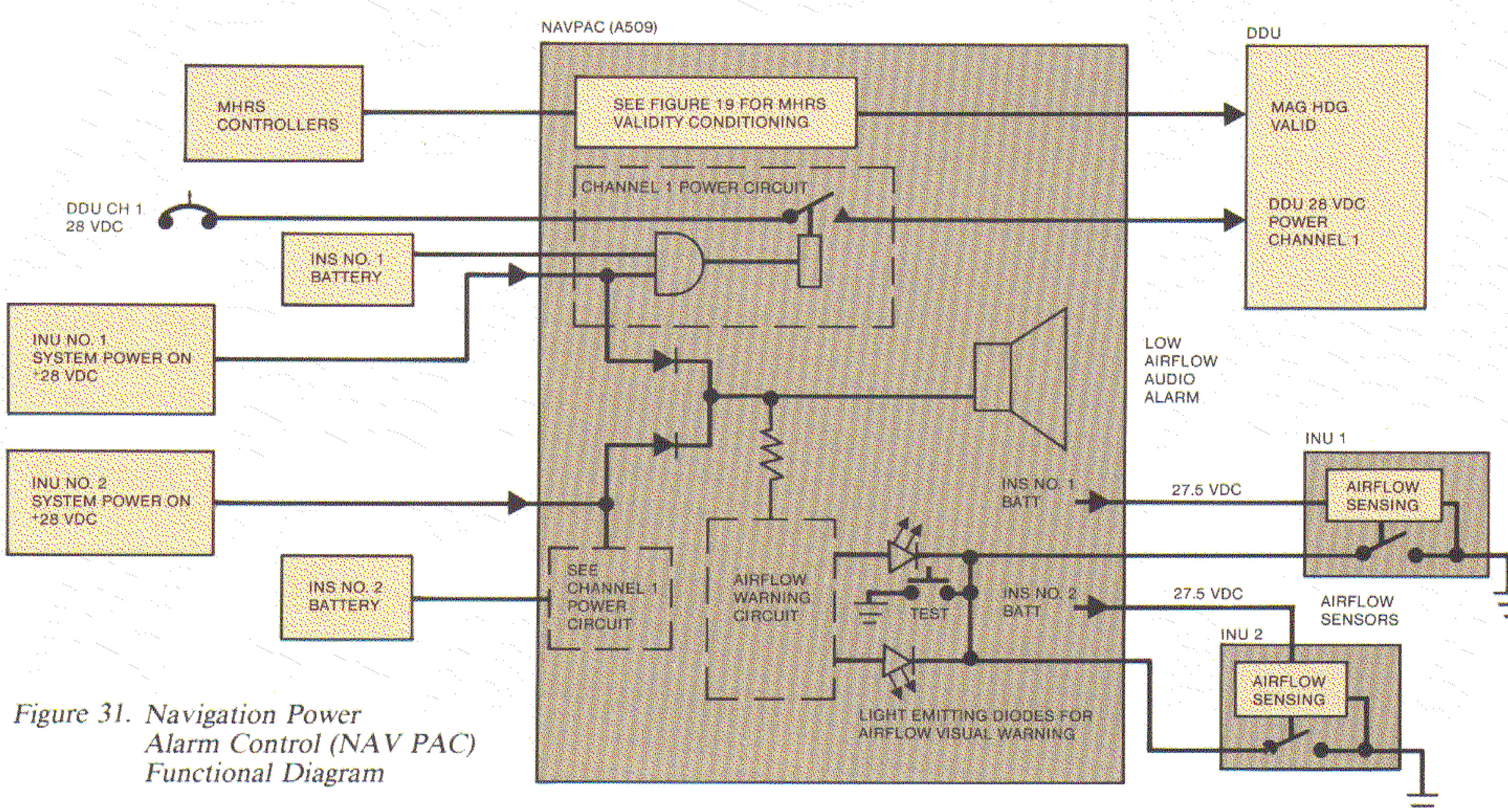


Figure 31. Navigation Power Alarm Control (NAV PAC) Functional Diagram

The Digital Data Unit is a system requirement that is peculiar to the P-3C installation. Each of the two system channels in the DDU operates on 28 VDC, supplied from circuit breakers located on the Aft Electronics Circuit Breaker Panel. These circuit breakers are the DDU CH-1 (MON ESS DC BUS) for INS No. 1, and the DDU CH-2 (MAIN DC BUS) for INS No. 2. When the INS is operating, each of the Inertial Navigation Units generates a 28 VDC System Power-On discrete signal that is applied to DDU power logic in the NAV PAC. When the power-on discrete signal is combined with INS battery voltage, the resulting signal will energize the logic relay that applies power to the DDU from either the DDU CH-1 or DDU CH-2 circuit breaker.

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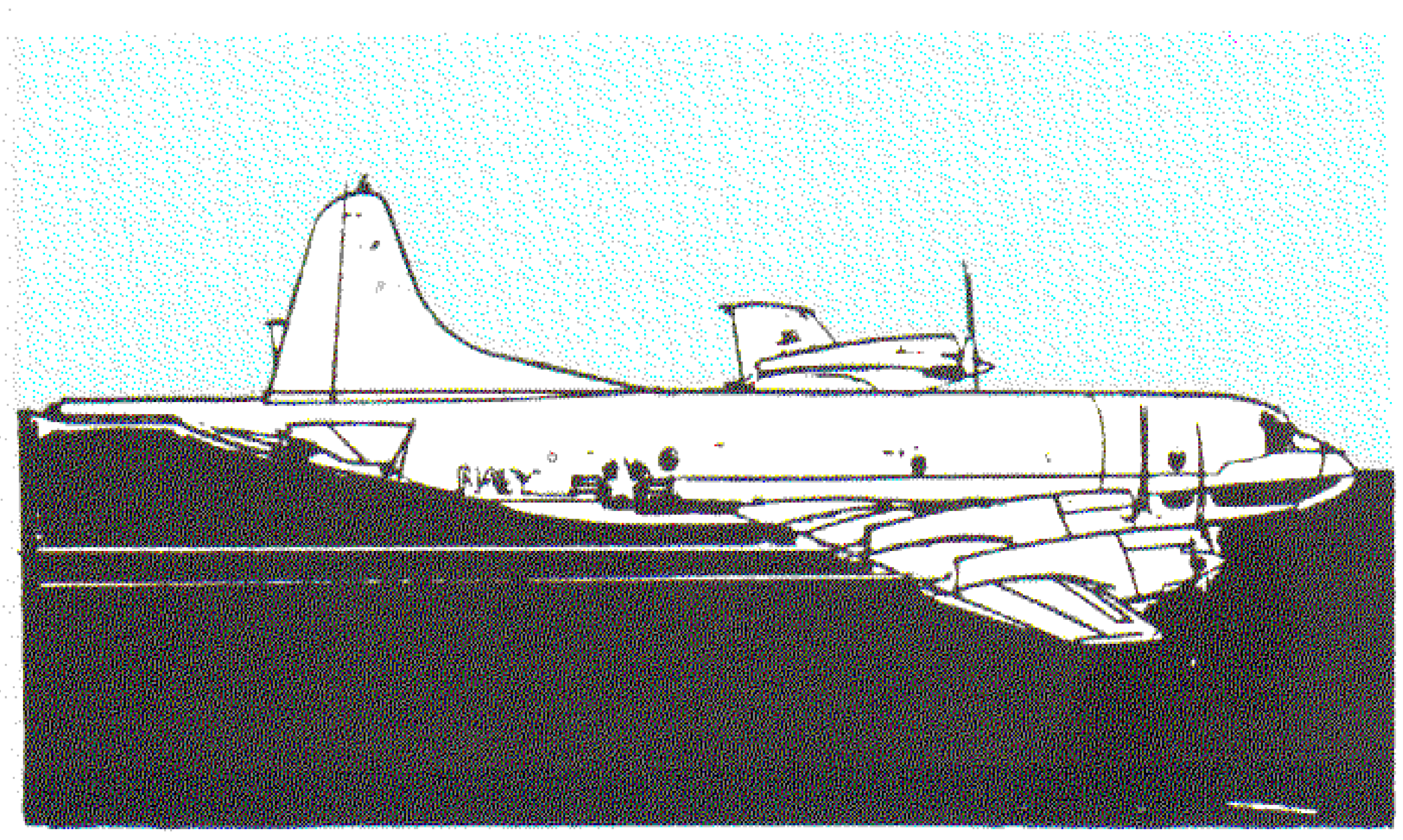
LTN-72 INERTIAL NAVIGATION SYSTEM OPERATION The LTN-72 Inertial Navigation System has three basic operating modes: Pre-Flight Alignment, Navigate, and Attitude Reference. Operation of the LTN-72 system is, to an extent, defined by the ARINC 561 characteristic. The system's operating characteristics were standardized to make the equipment interchangeable with other equipment designed to meet this characteristic, and to make the system operating procedures uniform. This, in turn, simplifies the INS logistics and training requirements. In this section we will discuss *how* the LTN-72 system operates, and describe some of the features and characteristics of each operating mode. We shall also briefly review some fault isolation techniques. Maintenance and operation procedures for the LTN-72 system are presented in the P-3 NATOPS Flight Manual and the P-3 NAV/COMM Crew Station Organizational Maintenance Manual.



Pre-Flight Alignment The accuracy of the dead reckoning navigation that is performed by an inertial navigation system depends upon the quality of the inertial system's pre-flight alignment. During alignment, the system is initialized with the aircraft's current geographic position, and the inertial platform axis and/or computer axis is oriented relative to the earth coordinate system. The LTN-72 system uses a combination of physical platform alignment and computer axis alignment. During alignment, the platform is leveled mechanically, using the gyros and accelerometers to define the level reference.

The LTN-72 system's "wander azimuth" mechanization requires that the inertial platform's heading be established relative to aircraft heading, then the computer axis (or reference frame) must be "rotated" to true north to establish the initial alpha angle. Once initialized with the departure coordinates of the aircraft, the LTN-72 system performs an automatic alignment in approximately 17 minutes. This period includes the time that is required to stabilize the temperature in the Inertial Navigation Unit to operating temperature, and the time required to calculate the gyro bias values. The alignment period may be slightly longer if the system is initialized in a very cold environment.

LTN-72 Inertial Navigation System alignment has three distinct phases: (1) caging the inertial platform, and establishing the proper internal temperature within the Inertial Navigation Unit; (2) leveling the inertial platform; and (3) gyrocompassing. The progression of system alignment through this automatic sequence is tracked by a status number displayed by the Control Display Unit. When alignment is initiated, the system status number 90 is displayed. As alignment proceeds, the system status decrements. When system status 02 is displayed, system alignment is complete. Figure 32 shows the alignment sequence and describes the status number display.

Figure 32. LTN-72 Inertial Navigation System Alignment Sequence

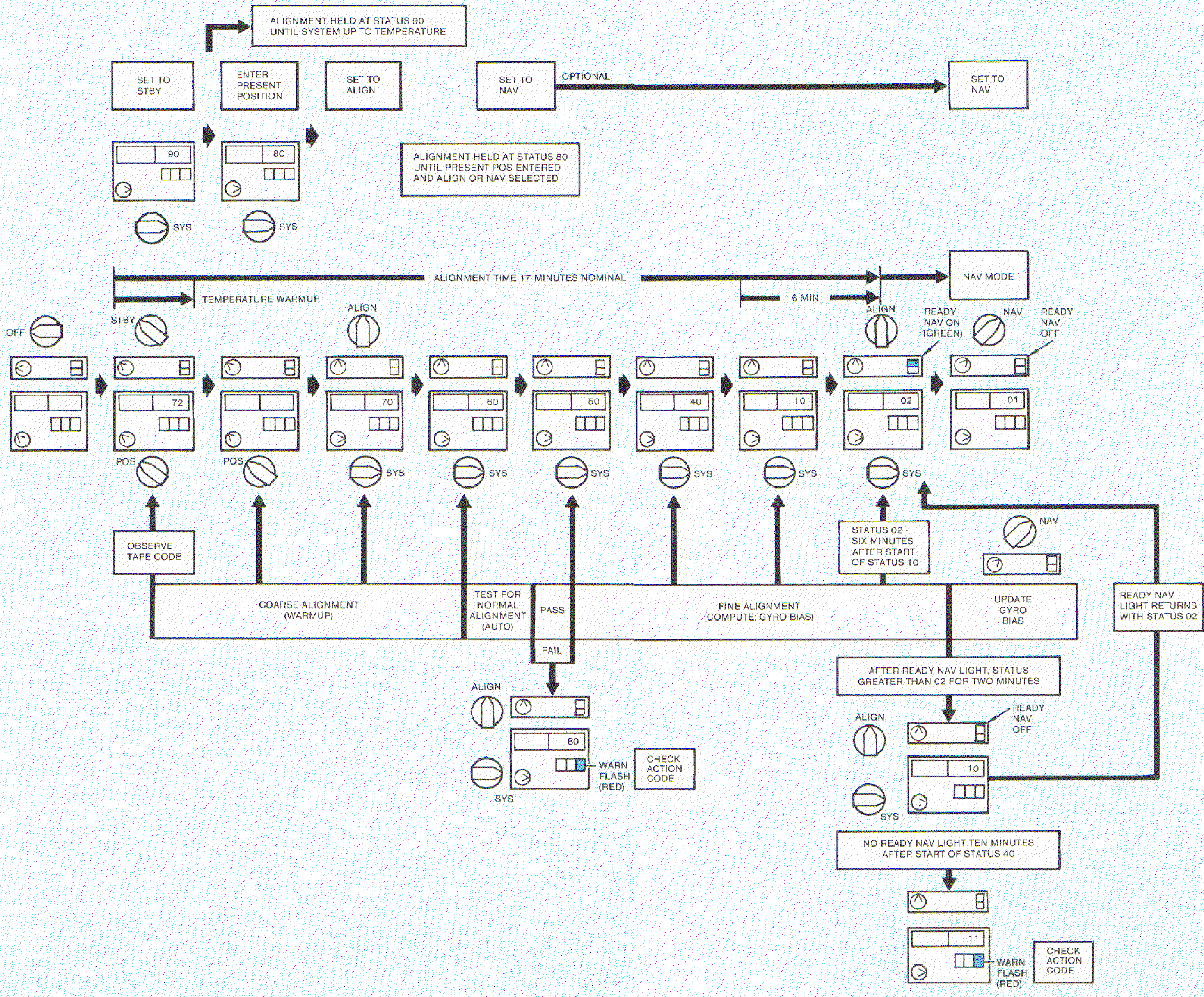


Figure 33.
LTN-72 Inertial Navigation System Alignment Sequence Indications

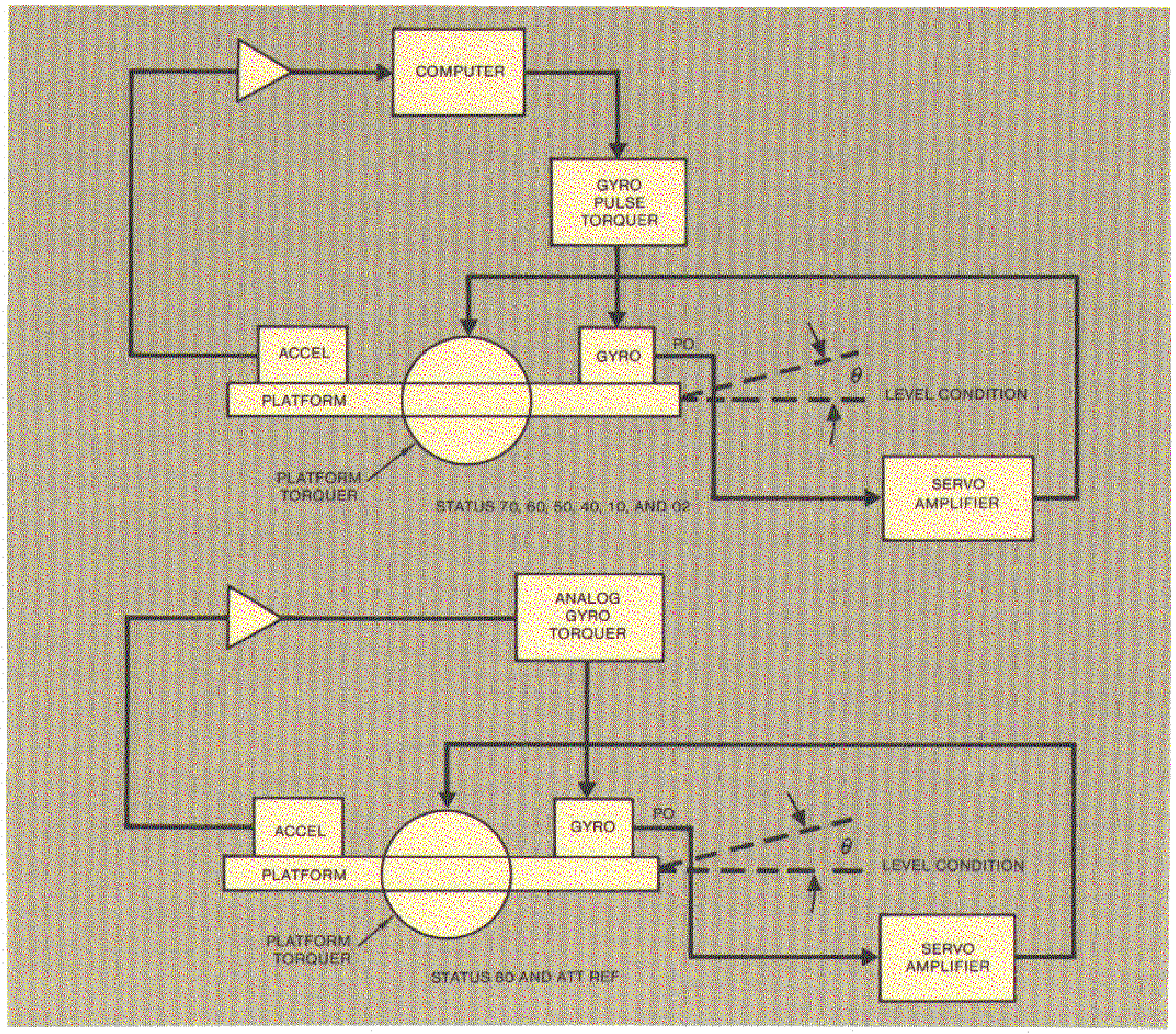


Figure 34. LTN-72 Inertial Platform Alignment Leveling Control

The status number can be observed in the right display of the CDU when the display selector is set to DSR TK/STS.

LTN-72 System alignment is initiated by positioning the Mode Selector Unit switch from OFF to STBY (see Figure 33). This applies power to the platform/stable element heating system and to the computer. During this period (STS 90), the four platform gimbals (inner roll, outer roll, pitch, and azimuth) are caged. Caging is accomplished by driving the roll and azimuth gimbals to null the synchro outputs. The pitch gimbal is driven to null the X-accelerometer output. Caging provides a rough reference during the alignment sequence, from which the platform leveling and gyrocompassing operations can be initiated. Meanwhile, the Inertial Navigation Unit's temperature control system begins to bring the inertial platform up to operating temperature. The status 90 phase of system alignment is complete once the gyroscopes are up to speed. At this point, system alignment status 80 is entered automatically, and the platform begins to define "level." The aircraft's current geographic position should be entered during the system alignment status 90/80 period. System monitors will detect failures and warn the operator if: (1) the INU gyros are not up to operating temperature in six minutes, or (2) the INU gyros are not up-to-speed within one minute after the platform temperature stabilizes.

The Inertial Navigation Unit alignment sequence advances to status 70 when the Mode Selector Unit switch is positioned to ALIGN or NAV (if the present position is entered). From this point, the alignment sequence will require about 12 minutes for completion. The inertial platform is leveled by driving its gimbal pitch and roll torquers (via the gyros) to null the accelerometer outputs (see Figure 34). During this period, the computer calculates the alpha angle (between true north and platform heading) by progressively rotating the computer reference axis.

Initially, the computer applies full earth-rate compensation to the X-level axis gyro. If the aircraft is on a heading of true north, the platform will track the level position as the earth rotates, and the alpha angle will be calculated as zero. If the aircraft's heading is other than true north, the application of full earth-rate compensation to a single level gyro will cause the platform to tilt and cause the accelerometers to sense gravitational acceleration (see Figure 35). The computer then begins to divide the earth-rate compensation between the two level-axis gyros until the accelerometer output is held at null. The ratio between the earth-rate terms, applied to each gyro when the platform is held level, is used to calculate the initial alpha angle.

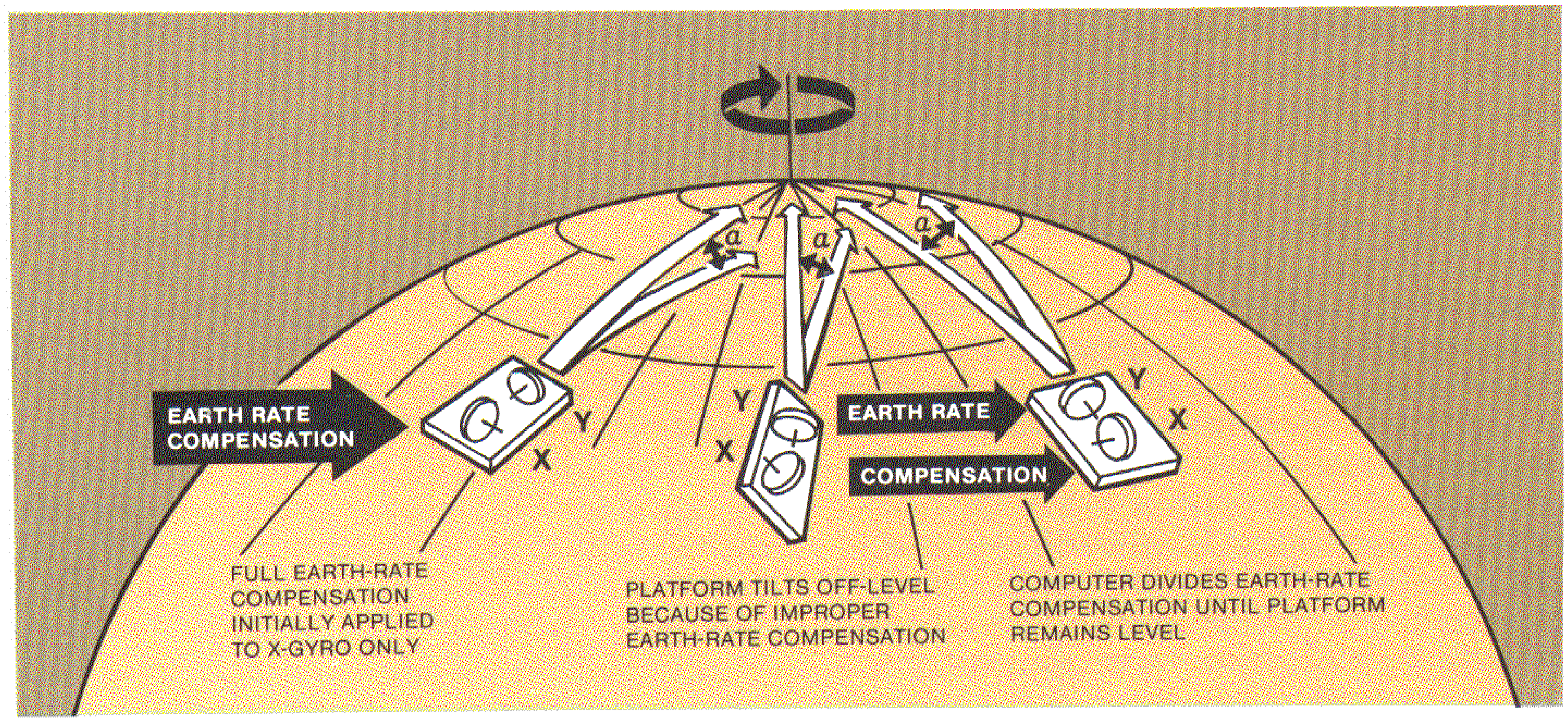
After the alpha angle is established during system alignment status 60, aircraft true heading is calculated. At this point, the system decrements to status 50 and the True Heading Validity discrete signal shifts to VALID. The Digital Data Unit combines this signal with the Primary and Auxiliary Attitude Validity discrete signals and the MHRS Valid discrete signal to generate the INS Valid discrete signal. The INS Valid signal will cause the GYRO flags on the pilot's and copilot's Flight Director Indicators to retract.

The Inertial Navigation System status display continues to decrement during the alignment sequence until status 02 is reached. At this point, INS alignment is complete, the RDY NAV light on the Mode Selector Unit illuminates, and the gyro bias has been calculated. When the operator selects NAV at the Mode Selector Unit, the system status will shift to 01 and the system will begin to operate as an inertial navigator.

Automatic Gyro Biasing All gyros, no matter how carefully they are manufactured, will drift. Causes of drift include mechanical bias, mass imbalance, temperature sensitivity, and magnetic sensitivity. Gyro drift will produce navigation errors when the gyro output drives the platform to an off-level condition, causing the accelerometers to sense gravity. In inertial navigation systems, gyro drift is minimized by applying a correction bias to the gyro to counter the unwanted torque. Biasing techniques have evolved, from using a manually-adjustable potentiometer on the first-generation ASN-42 system to employing fully automatic self-calibration of the three gyro axes in the LTN-72 system.

The LTN-72 system gyro bias calculation is performed automatically during the status 10 alignment period. The *level (X,Y) axes* are tested to measure the actual-north gyro drift. The measured drift is used to determine the required gyro bias correction. The calculated bias is tested to determine if it is a reasonable value. If this value is *less* than 0.112 degree/hour, the computer uses it to update the gyro bias torque.[14] The level-axis gyros are then biased at 100 percent of the calculated bias value for the remainder of the operation cycle. The gyro bias memory is updated by 20 percent of the calculated correction when the system status decrements to 01 (NAV Mode). The stored gyro bias is used during the next alignment cycle to provide a starting point for the gyro self-calibration. This update of the *level axis* memory is called a *Mini Bias.*

Figure 35. Inertial Platform Wander-Azimuth Alignment



The magnitude of apparent gyro drift includes the combined effects of mechanical unbalance and computer-applied earth-rate correction. Errors in the present-position latitude entry during alignment will cause the computer to apply an improper earth-rate correction to the gyro. This causes excessive gyro drift, which the computer will interpret as mechanical drift. As a result, the bias value will not represent the *actual* (mechanical) gyro drift correction that is required. By limiting the update of the gyro bias memory to 20 percent of the calculated bias value, long-term degradation of system accuracy caused by operator error will be reduced.

The accuracy of the level-axis gyro bias (Mini Bias) calculation can be improved by performing the north gyro drift test with the aircraft on different headings. The LTN-72 Inertial Navigation System's computer program contains a feature called Selectable Z-Slew that allows the operator to slew the inertial platform level axis in azimuth (about the Z-axis), so that system alignments can be performed on a heading that is 270 degrees from the aircraft heading.[15] This will enhance the Mini Bias of LTN-72 systems on aircraft that are routinely parked on the same heading.

The Selectable Z-Slew function is selected at the Control Display Unit while the inertial system is in alignment status 80. The operator selects DSR TK/STS on the display selector switch and keys in the word S, L, E, W at the CDU keyboard. After the operator has inserted this command (press INSERT at CDU) and set the Mode Selector Unit to ALIGN, the system alignment status will decrement from 80 to 75 and remain there for 60 minutes as the platform is slewed in azimuth. When the platform is positioned, the system will resume its alignment sequence and decrement from STS 70 to 02. The Mini Bias will be completed, as with a normal alignment, when the MSU is switched to NAV. Improvement of Mini Bias accuracy will be effective if this function is used periodically. However, it will not be effective if it is used more often than once, every other alignment.

The *Z-axis gyro* (azimuth) also has a self-calibration function. The bias value is based on the system error-rate for the last operation period. The Inertial Navigation Unit calculates the error-rate by storing the terminal position from the last flight, and comparing it to the entered position during system alignment. The difference between terminal position and entered position is the accumulated error for the last flight. The accumulated error is divided by the time the system was operating in Navigate during the last flight. This produces an error-rate in nautical miles-per-hour. If the calculated error-rate is equivalent to a drift of less than 0.15 degree/hour, the gyro bias memory is updated by 20 percent of the calculated bias value.

As with level-axis bias corrections, 100 percent of the bias value is applied to the Z-axis gyro for the subsequent operation cycle. If the calculated drift is greater than 0.15 degree/hour, the system rejects the calculated Z-axis gyro bias memory update. The operator will receive an indication at the Control Display Unit during alignment status 02 if the Z-gyro bias update value is excessive, but no operator action is required. This prevents "false" INU removals or re-alignment attempts that occur when the INU has been moved a long distance from the geographic position where it was last shut down. This condition occurs when the INU is installed and aligned after it has been received from a remotely-located repair depot.

Navigate Mode The LTN-72 system operates as a full inertial navigator in the Navigate Mode. The gyroscopes control the inertial platform to keep it level, and the accelerometers sense changes of vehicle acceleration. The computer integrates acceleration over time to calculate velocity, and integrates velocity over time to obtain position. The computer also develops the gyro torque corrections for the inertial platform, and continuously updates the alpha or wander angle. All of the system's navigation functions are accessible at the

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[14] If the calculated bias is excessive (greater than 0.112 degree/hour), the alignment sequence is stopped at STS 10 and the operator is warned by the Control Display Unit that system operation is degraded. The warning consists of a flashing red Master WARN light, and an action code that instructs the operator to recycle the system alignment and re-enter the present position.

[15] Azimuth alignment of the platform normally aligns the X-axis with the aircraft centerline. The computer then calculates the initial alpha angle.

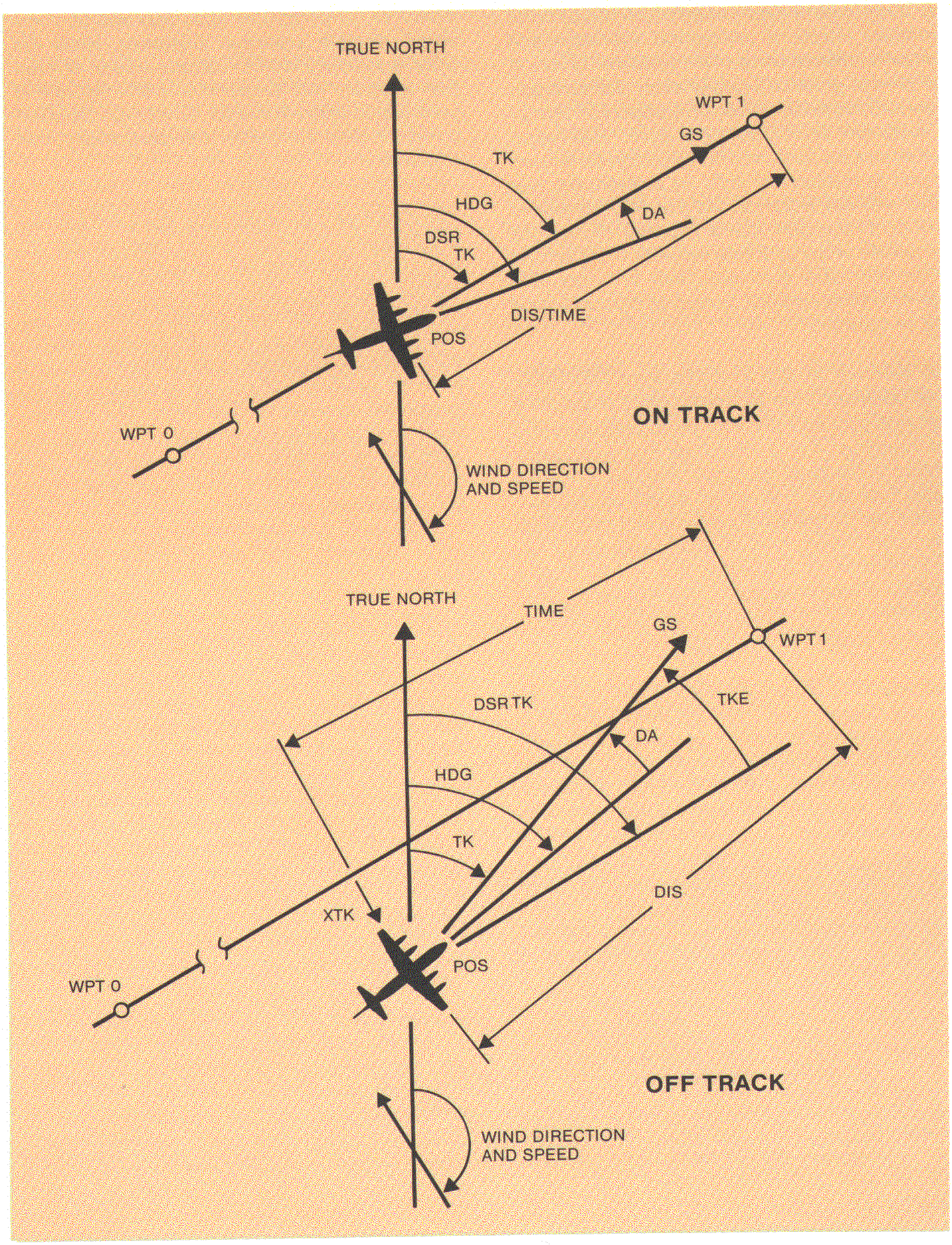


Figure 36. LTN-72 Inertial Navigation System Display Terminology

Control Display Unit. Figure 36 shows how the angles, distances, and other factors are used to track aircraft position.

The LTN-72 Inertial Navigation System will consistently operate with error rates of less than 0.5 nautical mile/hour if care is observed during maintenance and alignment. Although the ARINC Characteristic 561-11 does not specify accuracy requirements for inertial navigation systems, Litton Aero Products, the manufacturer of the LTN-72 system, has specified the basic drift rate for the system as less than 2.0 nautical miles/hour for at least 95 percent of the system's operation time. Litton has also established, and the Navy has approved, guidelines for equipment removal due to terminal position or velocity errors. The recommended limits are:

- For flights of greater than one hour duration, the radial position error in nautical miles (NM) for a single flight exceeds five times the time (T) in the Navigate Mode in hours: (>5T NM).
- For flights of greater than one hour duration, the radial position error in nautical miles (NM) for two consecutive flights exceeds three times the time (T) in the Navigate mode in hours: (>3T NM).
- The ground-speed error in knots for a single flight, or prior to departure, exceeds 15 knots: (>15 knots).
- The ground-speed error in knots for two consecutive flights exceeds 4 plus the time (T) in the Navigate Mode in hours: (> (4 + T) knots).

These recommended limits are less restrictive than the basic drift rate in order to reduce unnecessary Inertial Navigation Unit removals due to errors in system alignment or operation. However, the error criteria established for maintenance or removal may be made more restrictive by operational directives. An error-rate requirement of less than 1.5 nautical miles/hour is common in existing fleet instructions.

Attitude Reference Mode The Inertial Navigation System operates as an attitude/heading reference system in the Attitude Reference Mode. Approximately five minutes are required to align the Inertial Navigation Unit to function in this mode. The inertial platform is caged in all three axes during alignment status 90, and platform analog leveling is performed during alignment status 80. The ATT REF Mode can be entered on the Mode Selector Unit from either the OFF or STBY position. Successful alignment of the INU in the Attitude Reference Mode is indicated by retraction of the GYRO flags in the Flight Director Indicators.

In this mode, the Inertial Navigation Unit's stable element operates as a combined vertical/directional gyro. When ATT REF is selected, the computer is shut down and the platform is maintained level by the gyros. The outputs of the *X- and Y-accelerometers* are connected to apply torque directly to the *Y- and X-gyros,* respectively. When the accelerometers sense an off-level platform condition, they provide transport and earth-rate correction signals to the gyros. The output of the accelerometers is monitored to determine the magnitude of the signal. Large output signals are indicative of aircraft maneuvering, and are not applied to the gyro torquers. This prevents the platform from being driven off-level.

Torque is *not* applied to the Z-gyro in the Attitude Reference mode. However, the Z-gyro does provide a directionally-stable platform heading output during maneuvering by controlling the platform slew in rotation about the azimuth axis. The platform heading synchro output is transmitted to the Magnetic Heading Reference System to stabilize that system's magnetic heading output.

The Inertial Navigation Unit does not calculate true heading (or magnetic variation) when it is operated in Attitude Reference Mode. During INU operation in the Navigate mode, true heading is calculated by the navigation computer and it is provided as a synchro output from a Digital-to-Synchro Converter. In the Attitude Reference Mode, the true heading electrical synchro angle signal is not provided to the aircraft systems.

Off Mode When the system is turned OFF from an operating mode, it will be observed that the system apparently remains ON for approximately two seconds. After switching to OFF, the system enters a gyro de-spin mode to dynamically brake the gyros to a stop prior to complete shutdown.

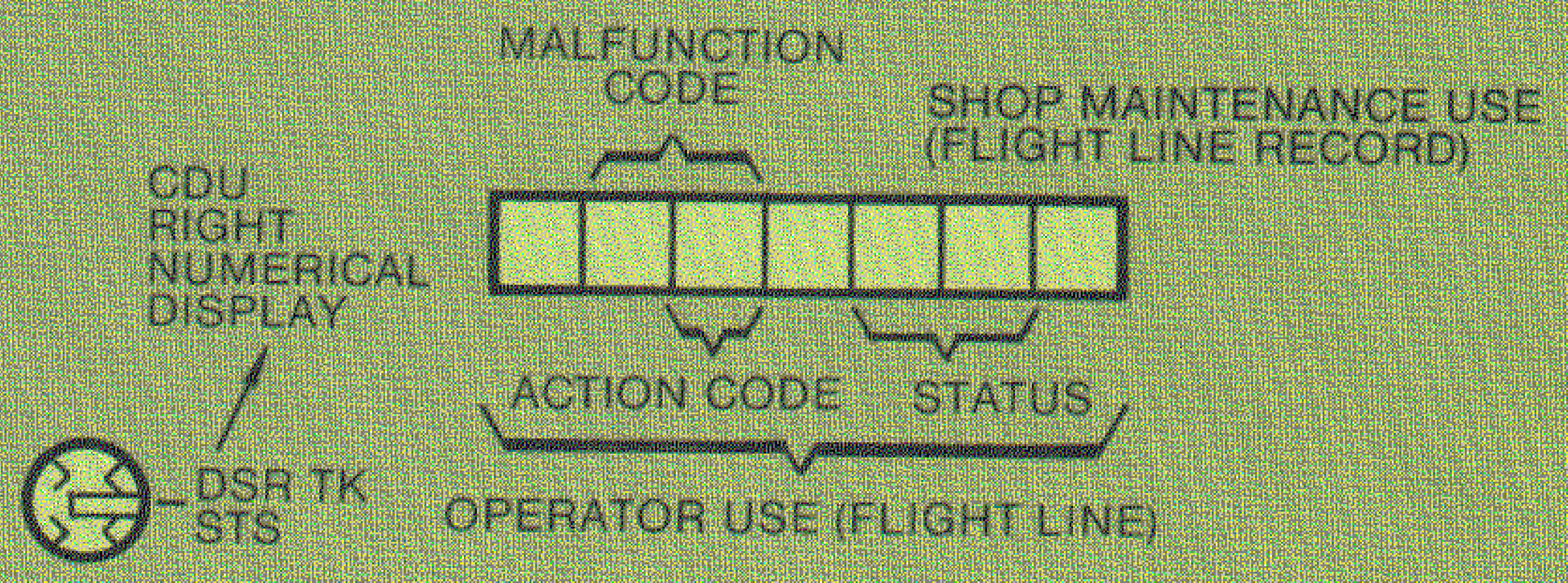
This permits the INU to be handled immediately, with no risk of gyro damage due to tumbling.

Fault Isolation Inertial Navigation System faults can generally be classified as one of the following: (1) Inertial Navigation Unit failure, (2) attitude or heading interface faults, (3) digital interface faults, or (4) Magnetic Heading Reference System faults. The P-3C LTN-72 Inertial Navigation System installation has several features that can aid the technician to diagnose and correct these problems. Procedures for utilizing these features are presented in the P-3 NATOPS Flight Manual, the P-3 NAV/COMM Organizational Maintenance Manual, and the P-3 Maintenance Instruction Manual.

The LTN-72 Inertial Navigation Unit has an extensive Built-In-Test (BIT) and monitor function. Table VI lists the failure modes that are monitored continuously and the warning discrete signals that are generated when a fault occurs. If there is a digital failure, the Inertial Navigation System automatically switches operation to the Attitude Reference Mode. In addition to the warning discrete signals, the system displays coded indications of the nature of the faults and suggested operator actions. The operator can access the Action codes by selecting DSR TK/STS at the Control Display Unit and observing the right display. If a failure has occurred, the appropriate Action code will be displayed. Figure 37 shows the Action codes and the corresponding Malfunc-

TABLE VI. LTN-72 Inertial Navigation System Monitor Functions

MONITOR FUNCTION	INS ANALOG WARN	INS DIGITAL WARN	AUTO SHUTDOWN	AUTO DOWN MODE TO ATT REF	CDU MASTER WARN
PLATFORM OVERTEMP - 88 DEGREES C (190 DEGREES F)	●	●	●		●
POWER SUPPLY OVERTEMP - 84 DEGREES C (183 DEGREES F)	●	●	●		●
PLATFORM COMPARTMENT OVERTEMP - 71 DEGREES C (160 DEGREES F)	●	●	●		●
DIGITAL POWER SUPPLY OVERTEMP - 79 DEGREES C (174 DEGREES F)		●		●	●
GYROSCOPE THERMISTOR OPEN	●	●			●
GYROSCOPE SPIN FAILURE	●	●			●
GYROSCOPE DIGITAL PULSE TORQUER FAILURE		●			●
PLATFORM SERVO FAILURE	●	●	●		●
SYNCHRO EXCITATION (26 VAC) LOSS	●				
ATTITUDE AND HEADING SYNCHRO OUTPUT FAILURE	●				
POWER SUPPLY DC TO DC CONVERSION FAILURE	●	●	●		●
POWER SUPPLY 13 VDC FAILURE	●	●	●		●
POWER SUPPLY 5 VDC DIGITAL FAILURE		●		●	●
INS BATTERY LESS THAN +17.5 VDC	●	●	●		●
COMPUTER CPU FAILURE		●		●	●
PROGRAM CYCLE FAILURE		●		●	●



ACTION CODE	RECOMMENDED ACTION	MALF CODE	SYSTEM MALFUNCTION	MASTER WARN
1(a)(d)	Remove, system maintenance required.	12 13 14 15 16 17 18	$\Sigma \vert \Delta V_Y \vert$ = 0 over 5 minutes $\Sigma \vert \Delta V_Y \vert$ = > 5.75g over 4 seconds $\Sigma \vert \Delta V_X \vert$ = 0 over 5 minutes $\Sigma \vert \Delta V_X \vert$ = > 5.75g over 4 seconds Gyro torquer fail Arithmetic error GS excessive	On steady
2(b)	Cycle system through off-on sequence. Reenter present position.	11 19 20 21 23	Program sum check failure Level velocity excessive Coarse alpha fail Level axis bias update excessive Platform not ready within 1 minute after up to temp	Flash
3(b)	Cycle system back to stby. Reenter present position.		Present position not entered and MSU in ALIGN or NAV	Flash
4(b)	Do not use INS for mag/true heading. Check all associated 26Vac 400 Hz circuit breakers and restart.	04 05 06 07 08 10	Dig sync Ch 4 loop test fail (drift angle) Dig sync Ch 3 loop test fail (TK TKE – DA) Dig sync Ch 2 loop test fail (HDG steering) Dig sync Ch 1 loop test fail (TKE steering) Dig DC loop test fail (X TK) Loss of platform synchro excitation also blanks HDG DA and wind displays	Off
5(c)	No action recommended.	03	Z bias update excessive	Off
6(b)	Turn off, check INS circuit breakers and restart.	24	Platform not up to temp in 5 minutes from turn on	Flash
7	Check for removal of attitude slew test connector.	02	Attitude slew connector in place	Flash
8(f)	Check DDU circuit breaker.	01	DDU not valid	Flashes until manually cleared (e)
8(d)	Check MHRS circuit breaker.	09	MHRS not valid	Flashes until manually cleared (e)

Note Check that DG/SLAVED switch on compass controller is in SLAVED position. This switch in DG position causes action code 8, malfunction code 09.

(a) Will occur in NAV mode only. (d) Gyro flag on FDI and system invalid to autopilot.
(b) Will occur in align mode only. (e) Complete CDU entry if in progress. Clear flashing warn by pressing clear pushbutton.
(c) Will occur at status 02 only. (f) Power or analog failure causes gyro flag on FDI and system invalid to autopilot.

Figure 37. LTN-72 Inertial Navigation System Action and Malfunction Codes

tion codes. The latter code can be accessed by pressing the CDU HOLD button. The Action code display will be replaced by a Malfunction code. Each time that the HOLD button is subsequently depressed, the system will display a Malfunction code for a different detected fault. When the codes of all of the detected faults have been displayed, the Action code reappears.

The Control Display Unit also has a selectable test function that enables the operator to check the display and indicator lights, and the LTN-72 system's digital interface with the P-3C CP-901 Tactical Computer. Setting the CDU display selector to TEST illuminates all of the unit's display elements, pushbuttons and annunciators. In addition, if the CDU selector is set to TEST during the system alignment sequence, the Inertial Navigation Unit will output a digital data test sequence that includes true heading, north/south velocity, and east/west velocity. These data are transmitted to the Digital Data Unit, and on to the Logic Unit No. 2 Navigation Multiplexer. By using the P-3C System Test Program (STP) Navigation Test (NAV NAV), the operator can display the following test values on the NAV/COMM station Auxiliary Readout (ARO):

VEL N	213.32 Knots
VEL E	426.64 Knots
TRUE HD	240.00 DEG

The STP NAV NAV test is an effective end-to-end test of the inertial system digital interface. This test also provides the operator with a list of the digital status validity discrete signals and the INS mode discrete signal (see NAVAIR 01-75PAC-12-3, Section 2). These data can help the operator or technician isolate problems that concern the transfer of digital data from the Inertial Navigation System to the P-3C Tactical Computer.

Heading errors can present some of the more difficult fault isolation problems. The P-3C has five sources of heading reference (see Figure 18). Each of the two LTN-72 inertial systems produces a magnetic heading signal and a true heading signal. The wet compass (mounted above the windshield) is an independent magnetic heading reference. The only common heading reference is the heading synchro in the A270 Navigation Simulator. The procedure for isolating heading errors should begin with a thorough assessment of all sources and displays of magnetic and true heading. This will enable the technician to determine the extent of the problem. Use of the Navigation Simulator allows the technician to start at the Forward Navigation Interconnection Box and isolate the fault to a display error or a source error.

The pitch and roll attitude interface can be approached with the same method. The Navigation Simulator Pitch and Roll synchros cannot be selected for the ASW-31 Automatic Flight Control System or the AJN-15 Flight Director System. Attitude signal errors to these systems are most effectively isolated by using an oscilloscope to check for the synchro input at the equipment.

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