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P-3 INERTIAL NAVIGATION SYSTEM

editor DON WADSWORTH

assistant editors WAYNE CRADDUCK : EUGENE DRICKEY

art VIRGINIA ROBINS

FRONT AND BACK COVER Patrol Squadron Sixteen (VP-16), one of four based at NAS Jacksonville Florida, carries the title "The Eagles" with befitting pride.

First commissioned in 1946 as a Navy Air Reserve Training Squadron (VP-56), "The Eagles" became VP-741 as a result of a 1949 reorganization, and Patrol Squadron Sixteen in 1953 when they were made a part of the regular Navy.

Although the squadron saw active duty during the Korean "police" action in 1951, their principal theater of operations has been the Atlantic and Mediterranean where they worked with the Sixth Fleet and NATO units from Iceland to Morocco, fulfilling assignments with the Spanish Navy, the British Royal Navy and Air Force, the South African Navy, the Portuguese Air Force, and the French Navy. For a period they were host-instructor to Brazilian air and ground crews.

Progressing from the time honored PBY Catalinas through the land-based P-2 Neptune series and finally in 1964 to the Navy's newest ASW patrol aircraft, the P-3 Orion, VP-16's extra-curricular activities have been as variegated as their regular assignments. They have air-dropped mail to ships and Arctic outposts, sought downed aeronauts and astronauts, and once worked a unique switch on the Christmas story by air-dropping candy and gifts to the Eskimo children of Scoresby Sound — a remote village 200 miles north of the Arctic circle.

Their flight skills lend credence to their name. Even as 2-year old fledglings, the Eagles won the Noel Davis "Outstanding Squadron" trophy in 1948, and by compiling the greatest number of accident-free hours in 1958 they earned the Chief of Naval Operations Safety Award. This record was extended to the remarkable total of 45,000 hours in 1961.

The Cuban crises saw VP-16 on surveillance duty around that troubled spot, a task they fulfilled until early in 1964 when VP-16 personnel started training for transition to the P-3 Orion.

Present activities find "The Eagles" spreading their wings on detached assignments from "Rozie Roads" Puerto Rico, Bermuda, and Argentia Newfoundland.

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P-3 Orion Inertial Navigation System

INTRODUCTION

A QUIET REVOLUTION has taken place during the past few years in the science of navigation.

Since the earliest times man has steered a course by periodically orienting himself to one or more "landmarks"—terrestrial or celestial; natural or artificial—whose position was known. He observed these points of reference directly, or in more recent times by means of antennas, to fix his position. Then through computation he could set a course.

Aircraft navigation inherited this traditional philosophy and all the tenets of "dead reckoning" from maritime navigation. Many implements for aerial dead reckoning have been devised, and many are retained to this day for cross-check purposes. However, most effort has been directed to providing a network of radio landmarks and to perfecting airborne electronic "observers" for radio navigation, all of which greatly reduce the chance of human error by providing directly usable information. The speed with which he can acquire and apply accurate information is vital to the aerial navigator, for the high speed and limited endurance of his craft allows very little tolerance for error and the lag of computation.

In respect to speed and accuracy, radio offers a nearly ideal facility for long-range aerial navigation, and the data from radio navigating equipment of today is highly reliable as well. But such systems are utterly dependent on the continuous success of both transmission and reception. When either is interfered with, the radio gear becomes ballast.

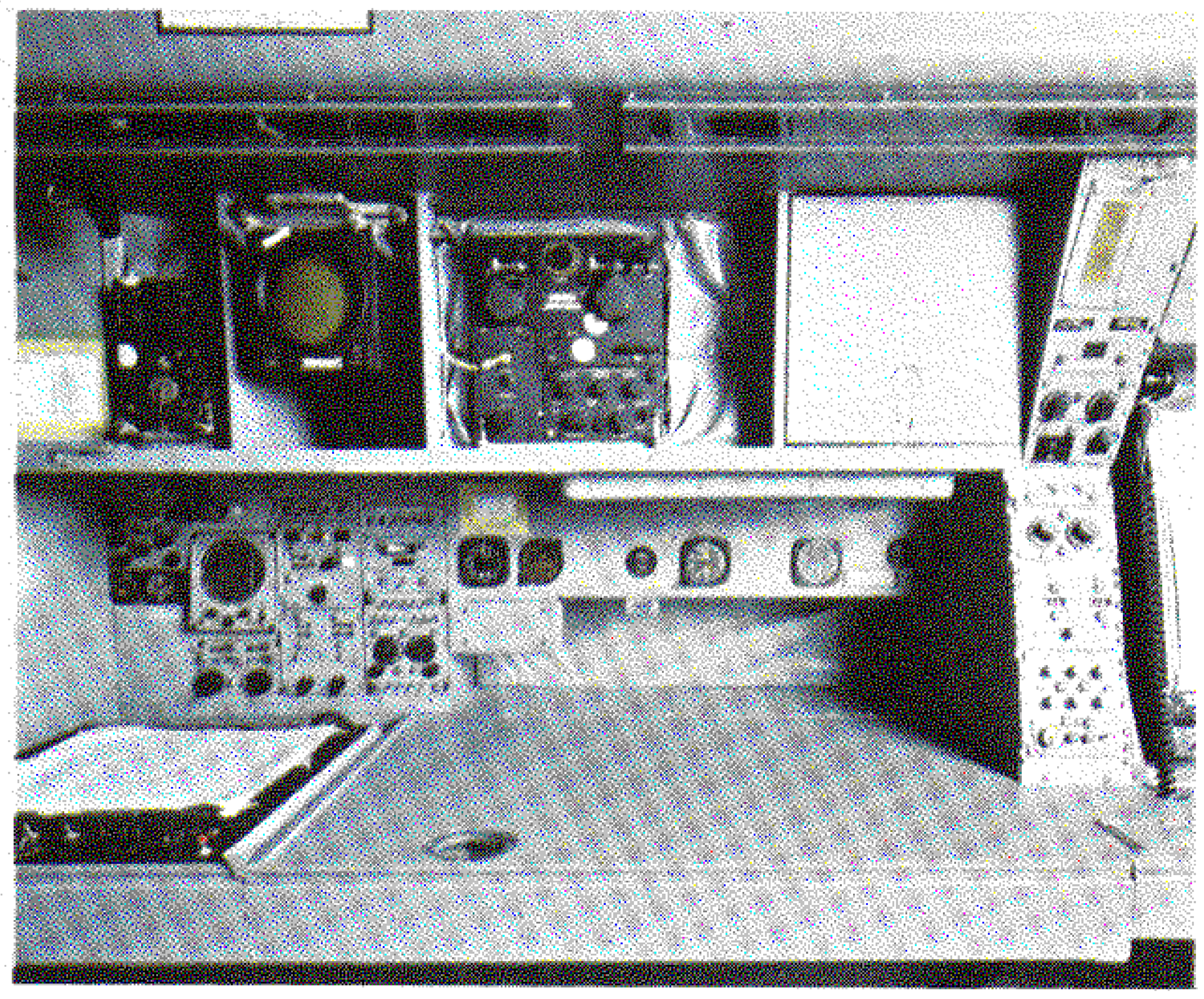
Radio navigation's vulnerablity with respect to reception has been demonstrated by the electronic havoc wrought by "magnetic storms," often lasting for days, that seriously jeopardized long range communication and aerial navigation in general. Of course, magnetic storms are only part of the problems that apply to military aircraft, for they must be prepared for a total radio black-out, and must carry out missions when there are *no* "fix" points to observe, either electronically or visually. Ideally, the military air mission should be guided totally without the prime essential of traditional navigation, i.e., reference to external landmarks.

The only alternative approach is to equip the aircraft to continuously "plot" its own track over the earth's surface from a known location.

This approach was first implemented by Doppler radar, which transmits beams at a depressed angle so that they reflect off the earth's surface. If the beam is transmitted aft, the frequency of the reflected beam "lags" (it is lower than the transmitted frequency). The converse is true in the case of a beam transmitted forward, that is, it is received as an accelerated frequency. The degree of lag or lead is directly proportional to ground velocity. Subsequent electronic computation involving this ground velocity and aircraft heading provides a continuous plot of the aircraft track over the earth's surface.

Doppler radar liberated the aircraft from reliance upon landmarks, providing the self-sufficient navigational system that was needed. Systems of this type were provided for many military aircraft, including the later-model P-2 Neptune, but Doppler navigation too has some inherent limitations. It depends upon terrain irregularities to reflect a usable signal back to the craft, and too-smooth surfaces, especially quiet water, tend to act as a mirror, off which the transmission glances away from the antenna where it cannot be detected. Also, if the reflecting surface has ground speed as in the case of water currents the Doppler return will be in error due to its effect.

The Doppler system proved the advantages of



continuous-plot navigation, but it does not entirely escape the handicaps of its predecessors. It depends upon reception of electronic signals, and for continuous earth-reference it too requires a sort of landmark. A true north reference signal must be provided and, in lieu of a more accurate source, the north reference can only be derived from the traditional, inconstant sources: a compass signal of the earth magnetic field (which in some regions is vague, changeable and subject to disruption) corrected, at intervals for variation. Further, Doppler radiations violate to some extent the principles of transmission security in this day of electronic counter measures.

The continuous-plot facility proved to be especially advantageous to the airborne sub-hunter mission. As usual, the next advance in the art was implemented by the underwater opposition (the *loyal* opposition, happily), and proven spectacularly by the "Nautilus" on a 4-day, 1800-mile cruise, submerged under the Arctic ice-cap—a completely unprecedented feat of navigation.

The heart of the "Nautilus" navigation system was a uniquely stable platform with super-sensitive accelerometers which gauge the changes in inertial force that accompany every change of direction and velocity. This voyage dramatically demonstrated that the technology existed to complete the revolution from systems dependent upon landmarks to a completely self-contained, self-sufficient system.

BuWeps and Lockheed, occupied in formulating the Orion ASW weapons system, decided to provide the new airplane with this latest and most promising facility.

Originally, it had been planned to provide two improved and identical Doppler radar systems for the Orion, each incorporating a low-drift stable gyro and a computer. Now it was decided to retain one Doppler Navigator in order to capitalize on its proven virtues for aerial guidance, and to incorporate the new Inertial Navigator (in such a way that it could be used independently or in conjunction with the Doppler radar) as the "prime" system. There was, at the time, comparatively little experience with airborne inertial systems, but a basically sound design was available at Litton Systems Inc. Although it was foreseen that experience would inevitably dictate modifications to realize the full potential of a new concept on a new aircraft, the designers would have been remiss if they had failed to provide the capability of operating in complete isolation and without the handicap of radio and/or radar emission. Subsequent developments have substantiated this decision.

The inertial system is based upon natural phenomena—as expressed in Newton's laws of motion—which do not vary with local condition, latitude, or aircraft attitude, and system performance will be as precise as the techniques used to fabricate, install, adjust, and operate the equipment. Lockheed's F-104 utilizes a variant application of the Litton Navigation System, and many new and proposed super-sonic aircraft feature inertial systems, for no other type of self-contained navigation system provides precise and continuous position information under such varied conditions.

The inertial navigator is practically essential for extremely high-speed, short-term flight and ideally suited to vehicles utilizing automated flight plans. However, accuracy in the "continuous plot" type of navigation is inversely proportional to elapsed time. Tracking a patrol mission which is slow, tortuous, and of many hours duration is actually a sore test of precision. A guidance error which would have a trivial effect on the short-term flight of a ballistic missile and be of little note to an interceptor may produce a substantially large deviation at the end of a long patrol mission.

Thus, the inertial guidance facility is ideal for the ASW mission, but the ASW mission is not always ideal for the facility; a situation that warrants the utmost care in both maintenance and operation, and is the reason for this Digest issue.

Any article dealing with a comprehensive subject such as the Inertial Navigator must necessarily be somewhat limited in detailed description. Our intent is to present the basic theory of components and the concepts common to all inertial systems, to discuss the P-3 system and its relation to the aircraft, and to offer operating and maintenance suggestions not presented elsewhere.

PART ONE
EARTH REFERENCED AIR NAVIGATION SYSTEMS

The basic purpose of an air navigation system is to determine the geographical position of the aircraft in transit and to provide additional information to guide the aircraft to a predetermined point on the earth. Inertial navigation systems vary in design and complexity to meet specific needs but they all accomplish their purpose by incorporating acceleration sensors (accelerometers), gyroscopes, a computer, and a variety of electro-mechanical devices into their design.

Briefly stated, the inertial system functions in this manner: Each time an aircraft changes velocity or direction of flight, it is said to experience acceleration of some magnitude in some direction. The accelerometers sense and measure these accelerations, and produce an acceleration signal for use by the computer where the various values of accelerations are integrated with time to produce a continually corrected velocity signal. A second computer function processes (integrates) the velocity signal to obtain current earth-reference information — distance traveled and position (latitude-longitude) data. In the following discussion, we have reviewed the basic principles of the inertial system components and explained how their characteristics are combined to meet exacting requirements.

ACCELEROMETERS Accelerometer operation is based on Newton's first law of motion, which states that a mass will remain at rest or in uniform linear motion unless acted upon by a force. This force causes the mass to accelerate (change velocity), according to Newton's second law which states that: Acceleration equals Force divided by Mass. An accelerometer measures the force acting on a known mass, thus effectively measuring the acceleration of that mass and hence of the craft that carries it.

The operating principles of one type of simple accelerometer are depicted in Figures 1a through 1d. In Figure 1a the accelerometer is level—its sensitive axis is at right angles to gravity—and no other accelerating force is present.

In Figure 1b, acceleration has caused the weighted end of the sensitive element to lag the unweighted end, and the indicator reflects a true measure of the accelerating force. However, in order for the accelerometers to supply accurate information with respect to the earth's surface, they must *always* be maintained in a known attitude to that surface even though both the earth and the aircraft are in constant relative motion.

In contrast to Figure 1a and 1b, Figure 1c and 1d depict the accelerometer tilted. Note that in Figure 1c the indicator is reading a gravity force while at rest and in Figure 1d the gravity force and acceleration force counteract and result in a zero indicator reading. From this it becomes apparent that *an accelerometer of this type must be maintained level with respect to gravity if it is to be used to measure movement in a horizontal plane with any degree of accuracy.*

Due to other factors involved, the simple unit just discussed is not suitable for use in an inertial system. A spring's reaction to load will change with age and temperature, and even if this were ignored, the bearing friction disqualifies such a unit from meeting the following stringent requirements for a satisfactory accelerometer:

1. Low threshold sensitivity — the ability to sense an extremely small acceleration from rest.
2. High resolution property—the ability to detect extremely small changes of acceleration.
3. Linearity over a wide range of acceleration —if the output is 1 volt for a one-half g acceleration it must be 4 volts for a two-g acceleration.

Note that even the "true" reading, obtained when the crude accelerometer shown in Figure 1b was perfectly oriented to gravity, was not actually a true reading in respect to the requirement for linearity. If the force being applied in Figure 1b were doubled, it can be seen that the indicator deflection would *not* double because the weighted end has swung away from the true south, and the east-accelerating force is no longer as effective as before.

Figures 1e and 1f reflect the simple accelerometer, modified to more closely meet exacting system requirements, and indicates in simplified form the accelerometers used in the P-3 system. To minimize bearing friction, the sensitive element is floated in fluid, and a pickoff device, a high gain amplifier, and a torquer essentially replace the springs shown in Figure 1a.

It should be noted that the most significant difference between the simple and the more elaborate units shown in Figure 1 lies in the incorporation of a sensitive element that is continuously driven (torqued) by the amplifier towards its zero-signal "at rest" position while an acceleration force is in effect. Operated in this manner, the accelerometer detects only velocity changes along its assigned axis (it ignores acceleration perpendicular to its axis) and does not rotate away to some less-sensitive angular position. Since

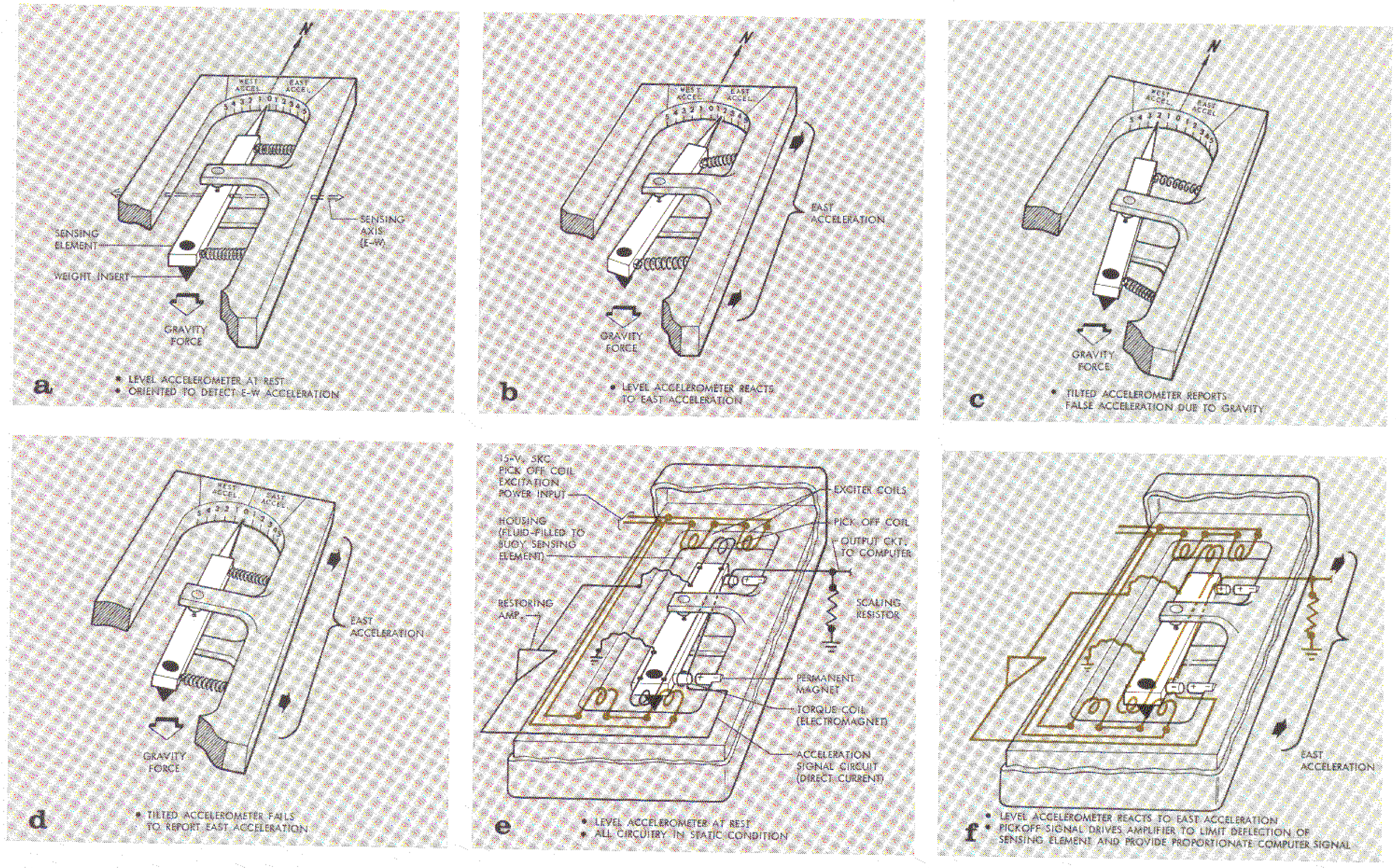


Figure 1 Operating Principles of a Simple Accelerometer

the amplifier output is just sufficient to restrain the sensing element, the amplitude of the output signal is at all times directly proportional to the sensed acceleration and may logically be used by the computer as the electrical equivalent of velocity change.

The accelerometer just discussed has but one sensitive axis. Obviously, two such units at right angles to one another are needed to measure movement of an aircraft over a flat plane. Actually, of course, the earth surface is not a flat plane but has a double curvature and it is not stationary insofar as inertial space is concerned. Since Newton's laws govern linear travel rather than radial travel, it is necessary to compensate the accelerometer signals to meet the needs of a terrestrial navigation facility. The reason for this compensation, and the manner in which corrections are introduced, are explained later in this article.

GYROSCOPES As shown in Figure 1, the accelerometers are totally useless unless they are kept in a known orientation relative to gravity, and to accomplish this while in motion over a rotating sphere (the earth) requires that the accelerometers be mounted on a platform which is stabilized by gyroscopes.

Basically, a gyro is nothing more than a flywheel except that it is used differently. The momentum of a flywheel is used directly to "store" torsional energy, but when it is set in motion, it gains a second property—the unique property of tending to remain stable in space. By mounting a high-speed flywheel in a special pivotal frame (a gimbal mounting) so that it is free-floating in respect to the vehicle, it provides a stable orientation reference for that vehicle while it is in motion. There are many and diverse applications for gyros, and they are not always completely at liberty to move in their mounts. The two P-3 inertial navigator gyros are free-floating, that is, each gyro spin axis is free to pivot a small amount in two directions within its case and is known as a "two-degrees-of-freedom" gyro. As explained in detail later, the platform gimbals extend this unrestrained feature to permit the gyro to remain stable during yaw, pitch, and roll maneuvers of the aircraft.

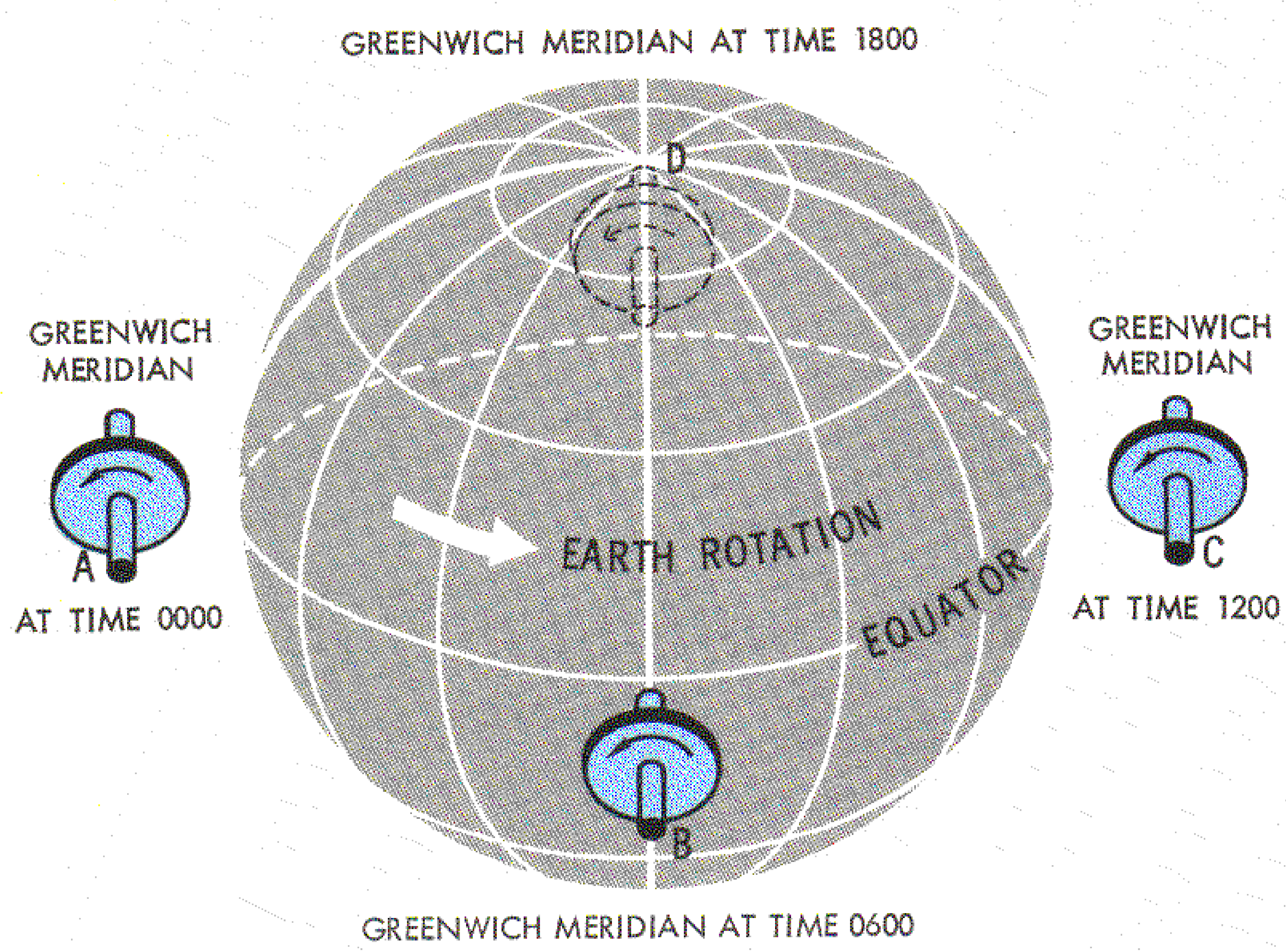


Figure 2a Apparent Gyro Precession due to Earth Rotation

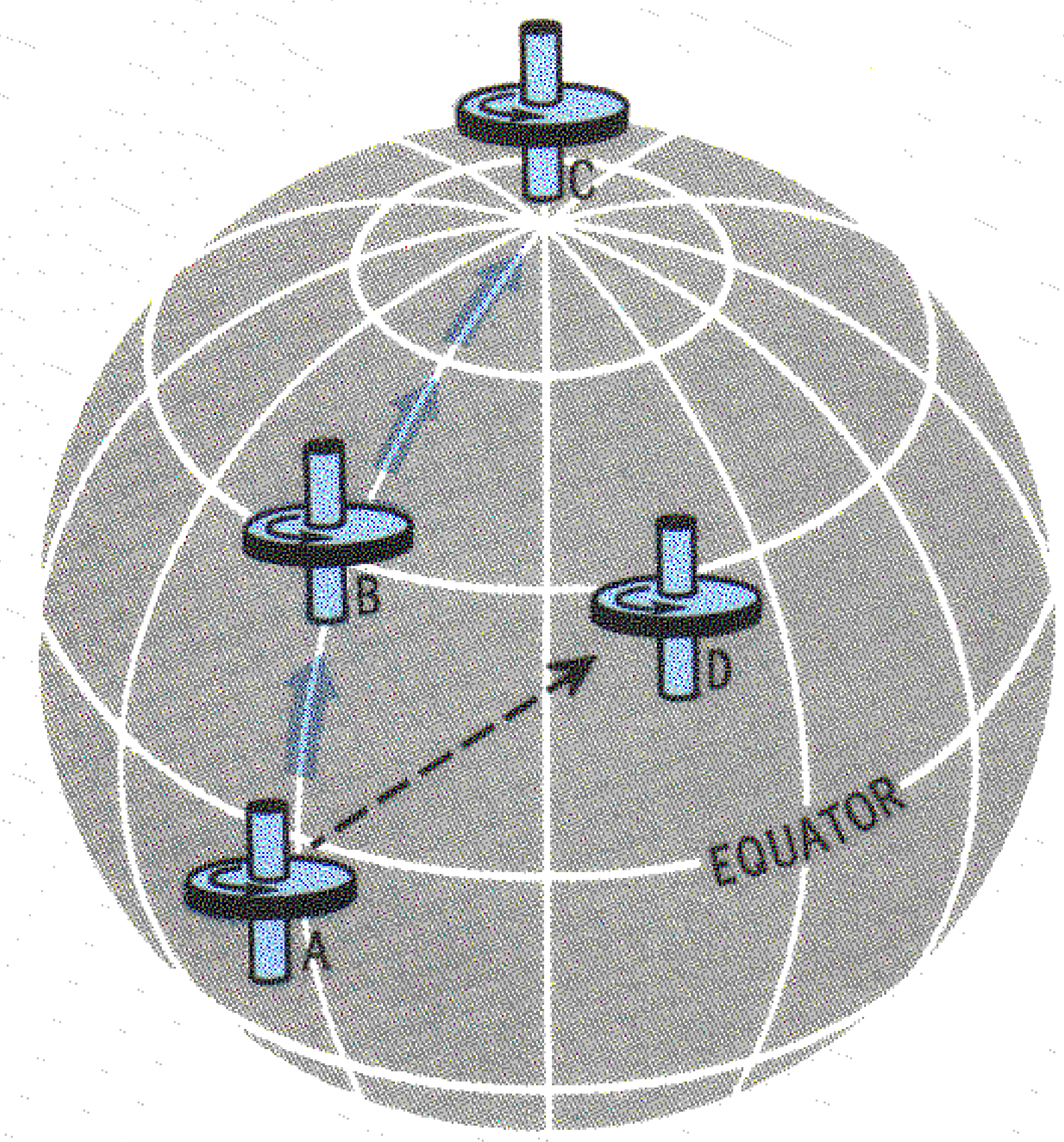


Figure 2b Apparent Gyro Precession When Moved From the Equator to the North Pole

The tendency of the gyro to remain stable in space (gyroscopic inertia) is not an unmixed blessing when it is applied in a global navigation system. Consider a gyro mounted in a frictionless gimbal and oriented in a horizontal plane at the equator with the spin axis pointed E-W as shown in Figure 2a. As indicated, although its spin axis remains fixed in space directly above the Greenwich meridian, to the observer on the ground it will appear to rotate in a vertical plane a full 360° in 24 hours as the earth makes one revolution.

It is also true that a gyro with its spin axis oriented north-south at the equator (see Figure 2b) would *appear* to turn 90° if it were moved from Point A on the equator to Point C at the pole.

When the direction of a gyro's spin-axis is changed, it is said to have precessed. In both of the above instances the spin axis direction in respect to space did *not* change. It appeared to change simply because the observer's orientation with respect to space changed. Such false motion is accurately termed "apparent precession." Obviously, if the gyro shown in Figure 2a were allowed to follow its natural tendencies, it would make a good 24-hr. clock, but as a compass it would be accurate only once a day. However, it can be seen that the rate of its apparent precession is always exactly that of the earth's rotation, and all that is required to keep the axis horizontal is to apply a steady counter-acting force to offset the apparent precession. When the proper force is applied to offset the *apparent* precession of the Figure 2a gyro, the spin-axis will appear to remain perfectly aligned but actually a *true* precession is taking place at the rate of 360° per day.

One of the characteristics of the gyro is its unique reaction to an upsetting force. When sufficient force is applied to overcome the gyro's tendency to remain stable in space (gyroscopic inertia), the resulting precession will be at right-angles to the applied force, and the direction of the precession will be determined by the direction of the gyroscope's spin. Figure 3 depicts a gyro with an applied vertical force (1) and the direction of horizontal movement (Precession 1), as well as a second force (2) applied on a horizontal plane and the vertical precession (Precession 2) that results. If both forces are applied simultaneously, precession about both axes takes place at the same time.

Applying this to the circumstances portrayed in Figure 2a and b, it can be seen that if a true precession is to be induced to offset the gyroscope's apparent vertical precession, *horizontal* rather than vertical forces must be applied to both gyros. In the case of the gyro depicted in Figure 2a (spinning from N to S) force at the E end must be from the N; or force at the W end must be from the S to offset the apparent precession due to earth rotation. To prevent the apparent tipping of the N-S oriented gyro (Figure 2b) as it is carried towards the pole, precessing force on the N axle extension must be from W; force on the S extension must be from E.

Further consideration of Figure 2b, Position D, indicates that the gyro spin axis will *not* continue to point true north-south when it is moved in a north-easterly direction as shown by the dotted arrow. To induce the necessary precession about the *horizontal* plane a *vertical* depressing force must be applied to the north axle extension or a vertical elevating force away from the center of the earth must be applied to the south axle extension to effect a heading correction.

Note that the amount of precessing force on the Figure 2a gyro is constant and predictable *only* if the gyro remains stationary on the earth. If this gyro were in a vehicle traveling east fast enough to circle the globe in one day, the apparent precession rate would double, and twice as much torque would be required to prevent the gyro from tilting. Conversely, circumnavigating the earth west-bound in a day would have the effect of cancelling the apparent precession automatically, and no torqueing force would be needed.

As explained later, in the inertial navigator the torqueing force is provided by means of an electrical signal. The exact amplitude of the signal is a function of earth rotation rate and aircraft velocity, and is supplied automatically to the gyros.

COORDINATE SYSTEM A position or location on the surface of the earth is commonly defined in terms of the latitude-longitude earth coordinate system. In this arrangement, the "parallels" of latitude run east-west and measure north or south position up to 90 degrees from the equator. The lines (meridians) of longitude run true north-south and measure east or west position up to 180° from the Greenwich, England (0°) meridian.

The two axes of the coordinate system correspond to the two horizontal axes of the stable platform. The platform axes are designated as X for the east-west direction and Y for the north-south direction. The third axis is identified as the Z or local vertical axis and its direction is always coincident with gravity. In some systems a third accelerometer is installed to measure vertical acceleration but it is *not* included in the P-3 installation.

Figure 4 depicts a cut-away view of the earth and its geographic coordinates. Note that the angle of both latitude and longitude are measured with reference to angular changes about the earth. The X accelerometer, however, measures linear, rather than radial, east-west acceleration and the Y accelerometer measures north-south linear acceleration. The output of both must be converted to a measure of angular change or motion to be indicative of latitude and longitude changes.

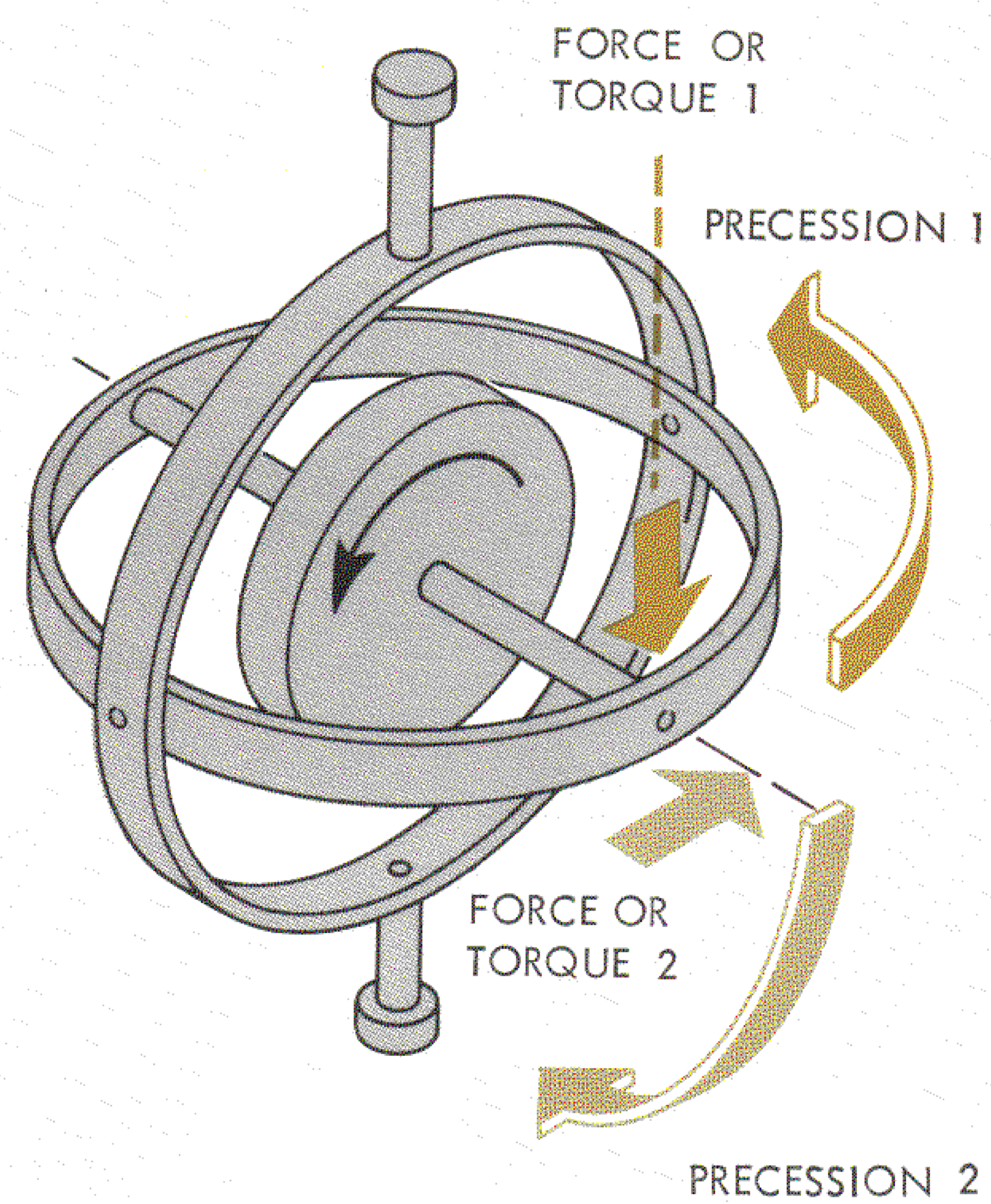


Figure 3 Gyro Precession Occurs in a Plane at Right Angles to the Applied Force

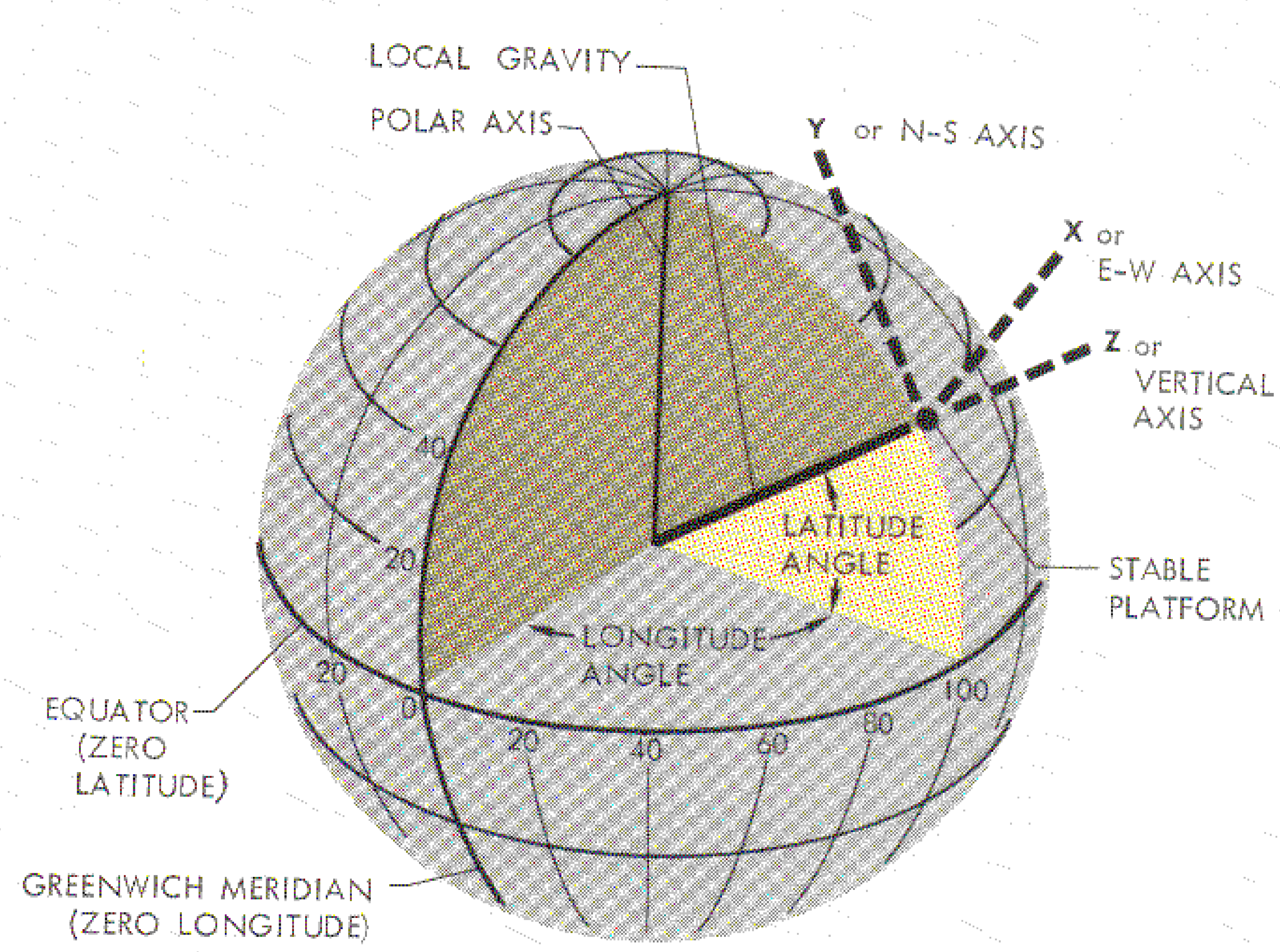


Figure 4 Cut away View of Earth and Its Coordinates Showing Angular Measurement of Latitude — Longitude and the X, Y, and Z Axes of Stable Platform

STABLE PLATFORM As previously stated, the accelerometers cannot sense a true value of acceleration unless they are properly oriented with respect to the surface over which they are to operate. To make this possible, the accelerometers—and the gyros—are arranged on a *stable* platform as shown in simplified form on Figure 5. Note the colored arrows which indicate that the gyros sense motion, and are torqued, at right angles to their spin axis.

The platform support permits it to rotate in any direction, as though it were suspended at a single point. The gyroscope whose spin axis lies athwart the E-W (X) axis senses, and serves to counteract, any tendency of the platform to rotate about the X axis. Accordingly, it is identified as the X gyro and, by the same logic the other gyro is designated as the Y gyro.

The platform is supported by motor-powered gimbals which respond instantly to the pitch, roll, and heading changes sensed by the gyros, and drive in a direction opposite to aircraft motion just sufficient to keep the platform from being disturbed. In effect, the aircraft appears to rotate freely about the stable platform without affecting its orientation in space. Figures 6a, b, c, and d depict gimbal positions for four of the many aircraft attitudes commonly experienced during flight.

If the aircraft were operating over a flat non-rotating plane, the stabilization described above would be

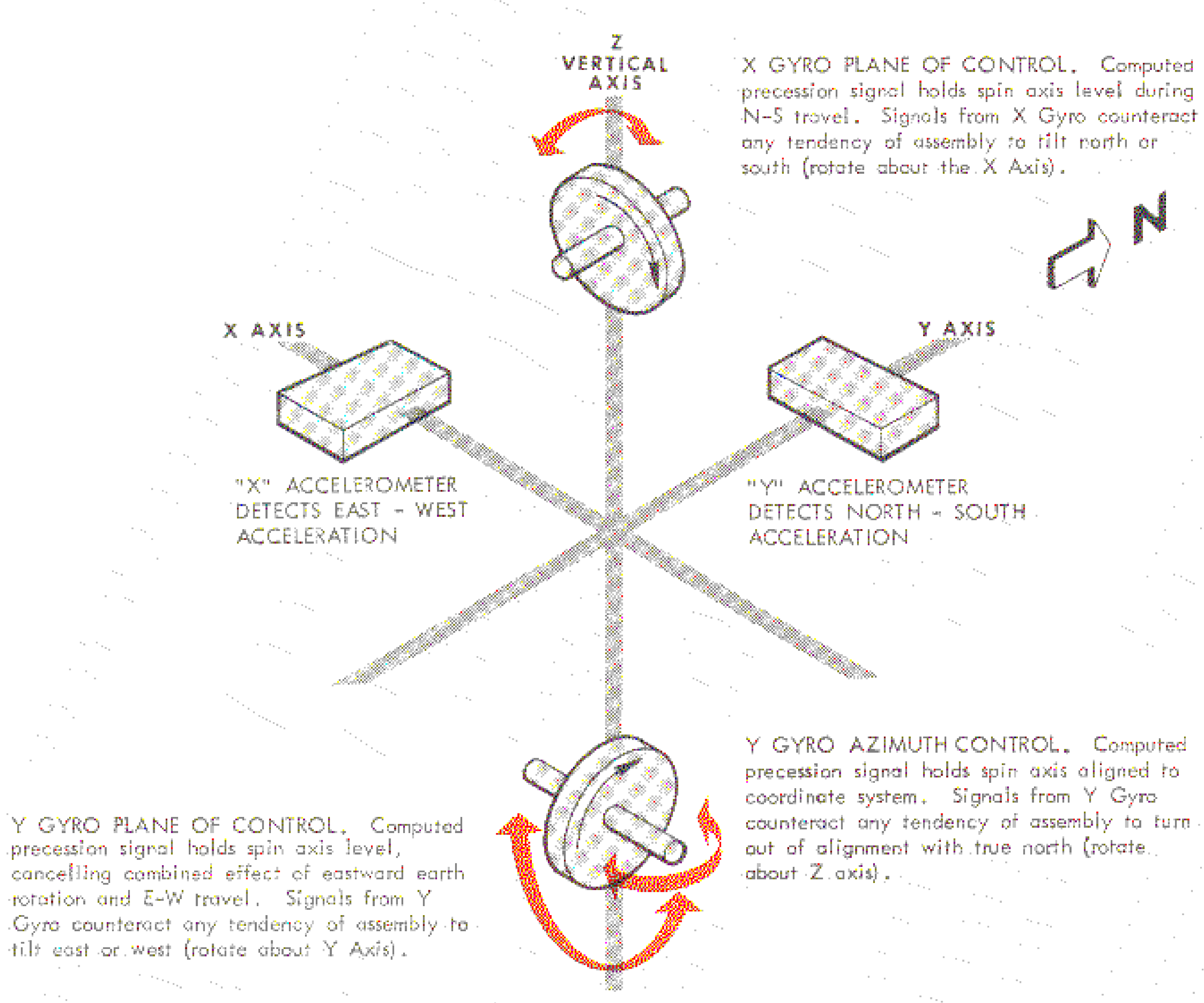


Figure 5 Accelerometer and Gyro Arrangement on Stable Platform. Colored arrows depict motion sensed by gyros and direction in which they are torqued.

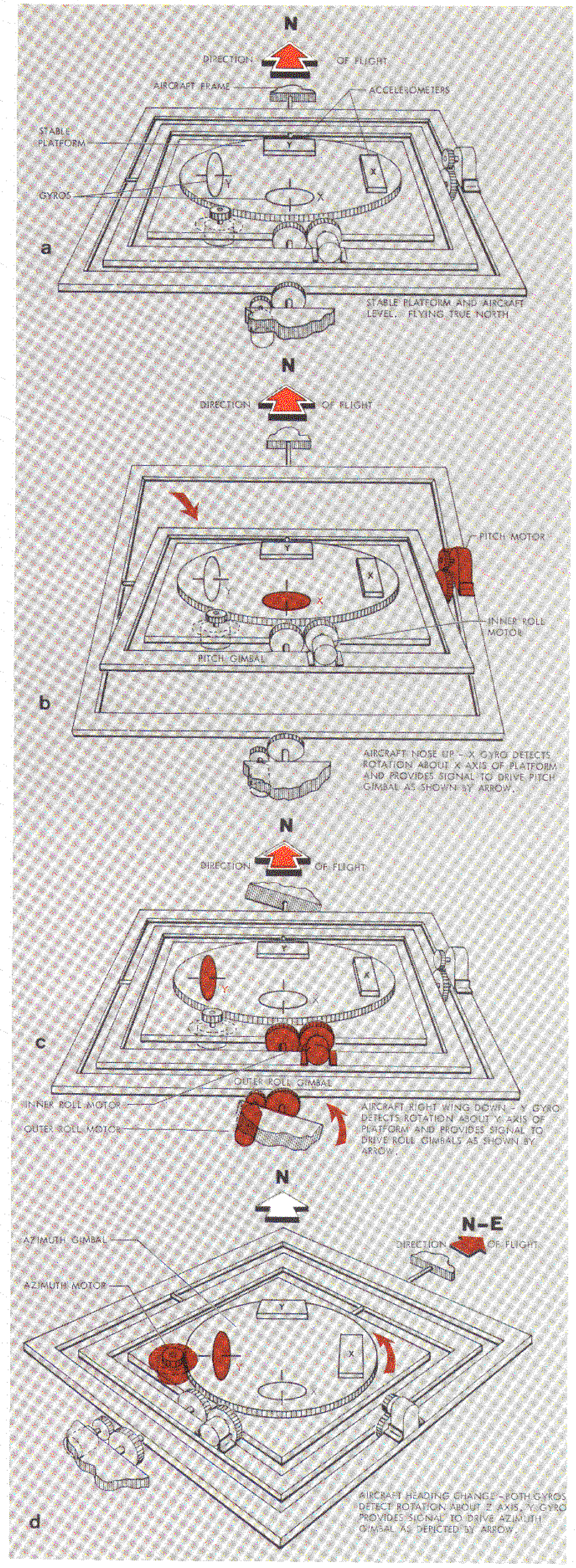
adequate. However, the platform must remain properly leveled and oriented in heading while it is moving over a curved, rotating earth. As noted in the gyroscope discussion, the forces required to level the two gyros are not the same. The spin axis of the X gyro (the upper gyro shown on Figure 5) must be tilted at a rate equivalent to the aircraft angular velocity in a north-south direction, but the Y gyro leveling rate is the *sum* of east-west aircraft velocity and the earth turning rate as well.

Since the gyros are also stable in respect to azimuth, they must be precessed *about* the vertical or Z axis of the platform—to preserve north-south/east-west orientation. When the gyros are precessed in the horizontal plane—or when the aircraft turns—the sensing elements of both gyros detect a slight disturbance at their spin axes, and the signal from either gyro could be used to position the platform. However, only the Y (lower) gyro signal is used in the P-3 inertial system, and its output drives the platform about the vertical axis a precise amount to maintain its heading.

The computer takes the earth rotational rate into account, together with the aircraft rotational rate as it moves over the earth curvature, and supplies a dc signal to the appropriate gyro torquer to force the gyros to precess and the platform to rotate *about* its X, Y, and Z axes. It should be noted that the *aircraft angular rotation* part of the X, Y, and Z torque signals is developed by the computer from the accelerometer-sensed inputs. This arrangement involves a complete loop in which the output of each accelerometer is modified by the computer and returned by way of the gyro torquers and platform gimbals to control the ultimate accelerometer position. This rather extensive circular path is known as the "vehicle torqueing loop." As explained later, the components of this loop produce an oscillatory type of control for the platform known as "Schuler tuning," which is indispensable to the performance of the inertial navigator.

Exact initial alignment of the platform is essential if mission starting errors are to be avoided, that is, the stable platform must be precisely leveled and oriented in heading prior to moving the aircraft. Alignment consists of leveling the platform and accurately "pointing" its axes—the sensitive axes—in the proper directions. Typically, the platform will go through a period of rough alignment with reference to its own mechanical structure and an approxi-

Figure 6 Gimbal Positions During Typical Flight Maneuvers



mate true north position before being accurately fine-aligned to its frame of reference, the global coordinates.

Precise leveling during fine alignment is sensed by the accelerometers. With the aircraft at rest, the output from both should be at zero when alignment is complete. However, the platform will not be level when alignment begins, and a component of gravity will be sensed. During the fine align period, the accelerometer outputs are ultilized to force the gyros to precess and, as the gyros move, they signal the gimbal servos to drive until the accelerometers are level.

The final accurate alignment of the platform to its true north-south and east-west axes is known as "gyrocompassing," and is accomplished after leveling is essentially complete, but the leveling process continues during gyrocompassing in this way: as noted previously, the Y gyro is made to precess 360° per day (15° per hour) to offset the apparent precession due to the earth's rotation. This calculated force is also applied during the alignment operation. If the Y gyro spin axis is not aligned perfectly with the equator, the induced precession will *not* be perfectly offset by the earth rotation, and instead of holding the platform level the precessing force will actually tilt the platform causing both accelerometers to again sense an increment of gravity. During gyro-compassing, both accelerometer outputs are utilized to re-erect the platform (as before), and the Y accelerometer signal is also used to torque the gyros and position the platform about the vertical (Z) axis. When the accelerometers cease generating cyclic correction signals, the platform will be perfectly level and the earth rate precession will be exactly counteracted by the induced precession, a state that is only achieved when the Y gyro spin axis is perfectly parallel with the equator and the X gyro spin axis is aligned true N-S.

COMPUTERS Since accelerometer information in its "raw" form is not directly usable, either for navigation display or controlling the gyro-stabilized platform, an analog* computer is utilized to convert this data into the form required for each individual function.

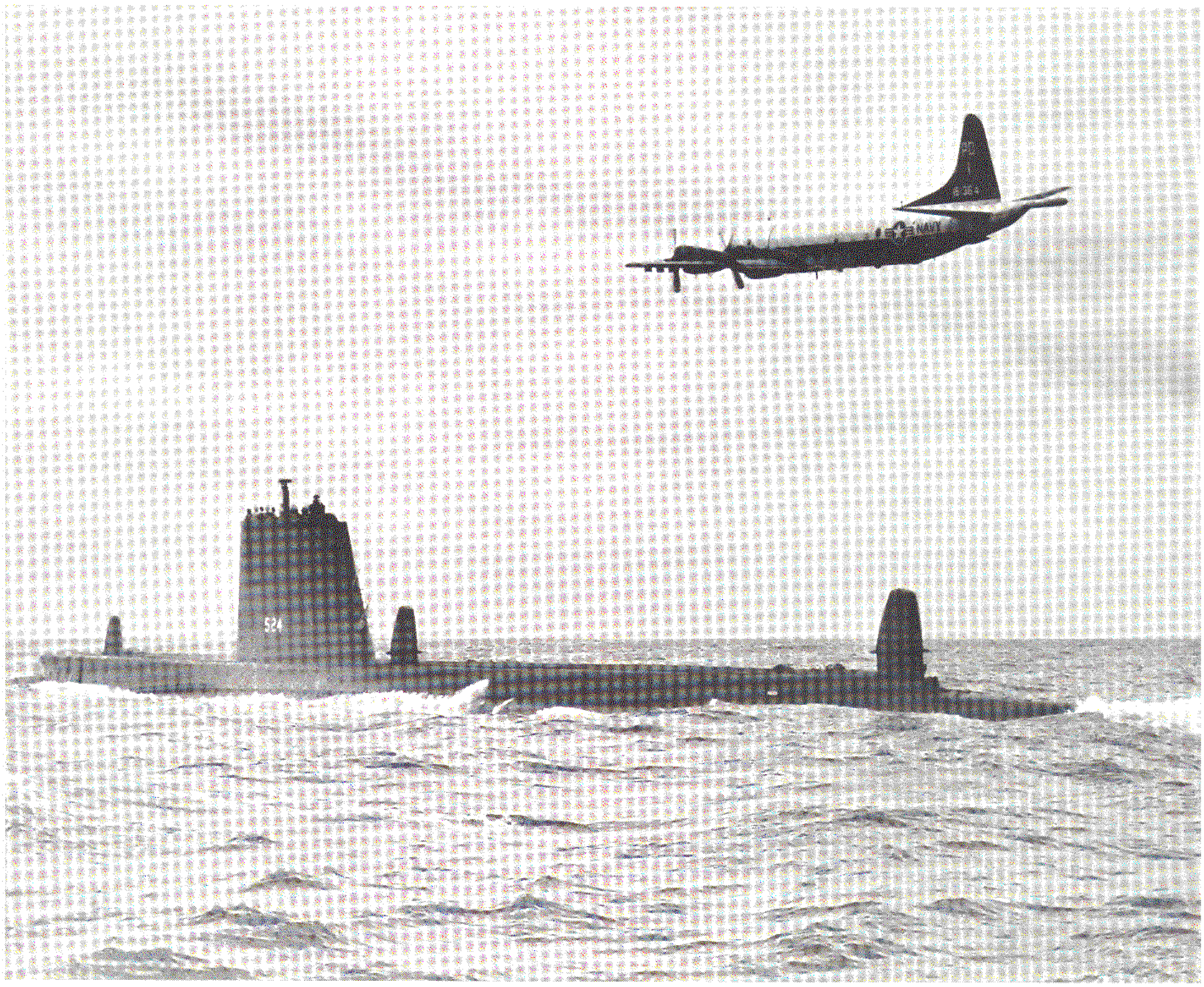
**Since the ASN-42 system was designed, advances in the art of miniaturization have made the digital computer more adaptable to aircraft use. The digital concept offers improved accuracy, and the analog computers discussed here will probably be replaced by digital components if a major improvement modification of the system is implemented in the future.*

An analog computer establishes an electrical quantity to represent a variable physical input, then samples this electrical quantity to determine increments of change in the physical force signal. For example, the analog computer used in the P-3 system utilizes a two-volt signal from the accelerometers that represents one g (32 feet per second/per second) acceleration.

The P-3 analog computer consists of three basic sections: input, computing, and output. The input section is composed primarily of resistors that scale the accelerometer outputs and the correction voltages to the precise values required by the computing section.

The computing section includes both *passive* and *active* elements. The passive elements (resistors and capacitors) serve to scale the input and output values of each computer active element. The active elements (operational amplifiers) perform the multiplication, summation, and integration functions of the computer. An operational amplifier is a very high gain dc amplifier with its gain controlled or regulated by an input feedback network. This type of amplifier will perform a variety of computer functions, for its output is determined by the nature of the feedback network employed.

If a capacitor is substituted for the usual resistive feedback network, the operational amplifier becomes an *integrator.* Stated simply, an integrator repeatedly samples the instantaneous value of its input, multiplies each sample by small increments of time, and then collects all of these products into a signal that accurately reflects the progress of action. If the input to an integrator is *acceleration,* the output will be *velocity.* If the input is *velocity* the output will be either *distance traveled* or *position* information as determined by the type of device to which it is connected.



The computer output section incorporates fixed resistors, variable resistors, transformers, and synchro transmitters. Each is selected to convert the computed data to a voltage or current signal that can be utilized within the system or by associated electronic equipment.

Other portions of the computer generate and scale the special correction voltages such as Coriolis, centripetal, elliptical, and earth rate values—self corrective signals described in subsequent chapters.

SCHULER PENDULUM As stated in the foregoing discussions, the problem of stabilizing the accelerometer platform in respect to vehicular motion (roll, pitch, and yaw) is satisfactorily resolved by the use of gyroscopes. The problem of compensating for the angular velocity of the aircraft around the earth is theoretically resolved by utilizing velocity data to compute the rate and direction of precession; precession which must be induced to overcome the gyroscopes' natural tendency to remain oriented to space rather than to the earth. Note that the induced precession is computed automatically within the system from data obtained from the accelerometer signals, and the accuracy of induced precession is, therefore, predicated on 100% perfection in the accelerometer signals.

Although a very high degree of precision can be attained, absolute perfection is a practical impossibility. Inevitably, in any inertial guidance system, a small error will occur in the accelerometer-generated signal for some reason or for some combination of reasons. As noted in the computer discussion, the acceleration signal is integrated once to produce a velocity signal, and the new signal is then integrated to obtain the distance-traveled and position data. Thus, the original small acceleration error is multiplied by time twice and it grows and accumulates exponentially in proportion to elapsed time.

Generally speaking, any given inertial guidance system installation will exhibit a fairly constant tendency to err, that is, the platform will be inclined to tilt slightly, and consequently each system has an inherent tendency to yield position data that drifts progressively farther from true. The seriousness of the problem, if it is not corrected, is illustrated in the following example:

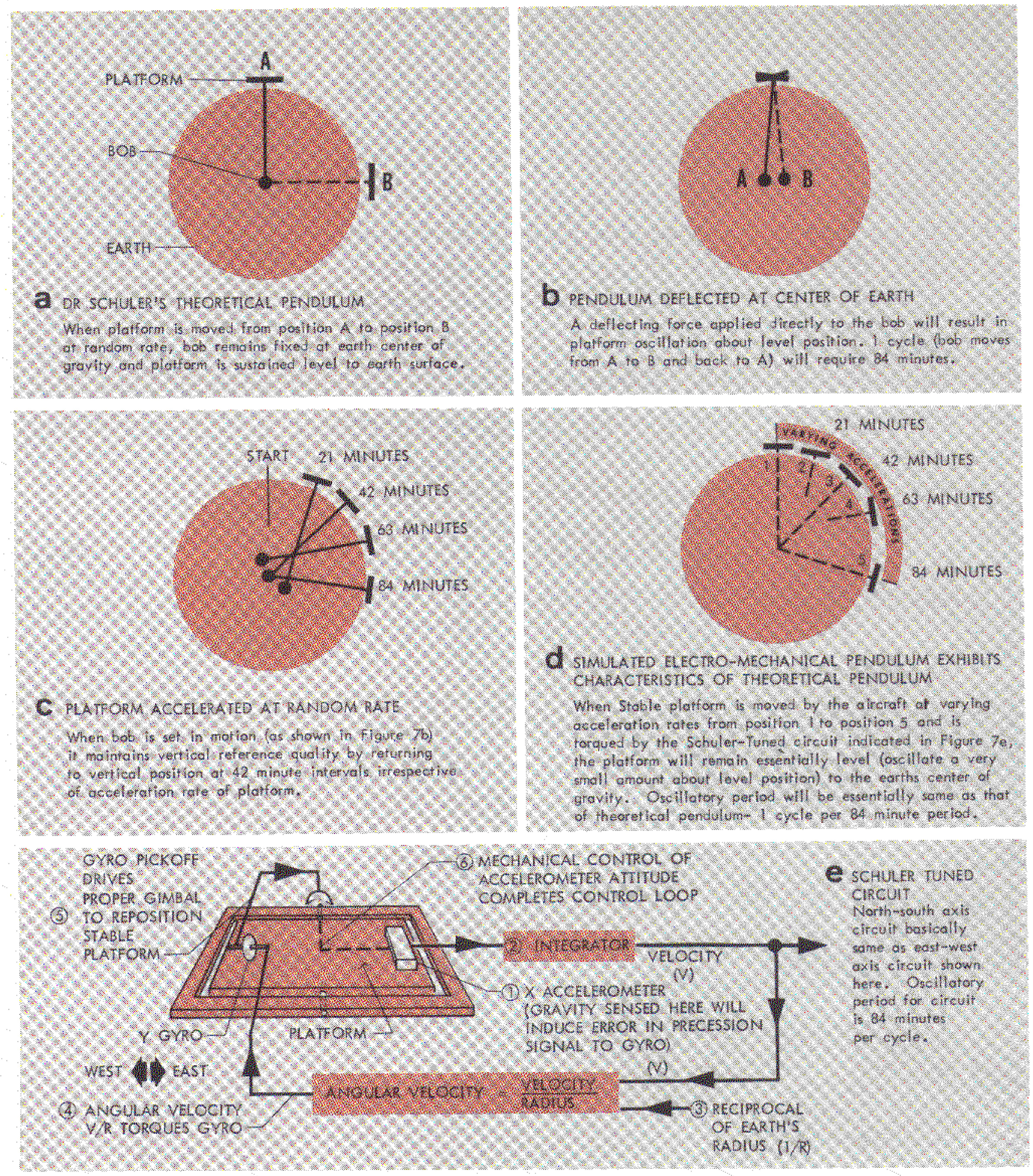
If we assume a 360-knot aircraft is carrying a guidance system with a small natural drift error which leads it to report a position 2 NM off course after 42 minutes of flight (252 NM traveled), it is evident that a system error is in effect equivalent to an accelerometer signal error of .00226 NM per minute, per minute. (An error in reported acceleration of .00226 NM per minute, per minute multiplied by ½ of the elapsed time gives an average velocity error of .04746 NM per minute. After 42 minutes of flight, this average velocity error will cause the system to report a position 2 NM from true.) A 2-NM error is an appreciable error, but not intolerable in view of the fact that the aircraft has traveled 252 miles. However, if the error is allowed to continue to compound at the same rate for an additional 42 minutes (.00226x42x84) the reported position will be 7.97 NM off course, and at the end of a 12-hr. flight the reported position will be 586 NM in error.

Thus it can be seen that if the theories of inertial guidance are to be made practical, it is absolutely essential that the system provide a tolerance for error in the "raw" signals produced by the accelerometers. Although the accelerometer error may originate in other ways, a small, nearly undetectable tilt of the platform due to gyro drift—allowing the accelerometers to sense a small increment of gravity—is generally the root cause of the natural drift exhibited by every installation. A mechanical physicist named Schuler suggested the means by which such a system could be made self correcting, thereby building into the system the required tolerance for error.

Schuler reasoned that the force of gravity is the only vertical reference that can be utilized by a traveling platform, and a plumb-bob is the most logical device to sense gravity force. Unfortunately, a plumb-bob of practical dimensions gives a true indication of gravity only when it is completely stable in space over the earth's center, or in uniform motion about this center. Every acceleration disturbs the bob-weight, that is, it becomes a pendulum. Although a pendulum of known arm has a perfectly predictable period —a set time per cycle—and will return to vertical at a predictable instant after it is set in motion if no further accelerations disturb it, the problem involves a platform with continually varying accelerations in all directions.

Schuler conceived a sort of "ideal" plumb-bob. He reasoned that a plumb-bob whose center of mass coincided with the earth's center of gravity would provide a vertical reference not vulnerable to acceleration of its support. At rest, it would be unaffected by lateral accelerations of its point of suspension. In motion it would retain its usefulness as a vertical reference because its period as a pendulum would be exactly predictable, that is, it would infallibly return to true vertical at predictable intervals regardless of any or all accelerations of its pivot point during the cycle. The period of such a pendulum would be almost exactly 84 minutes, meaning that it would

Figure 7
Principles of the
Schuler Pendulum
and Its Simulation
in the Inertial System



return to true vertical at 42 minute intervals, regardless of where the point of suspension moved during this time. Figure 7a shows the behavior of the Schuler pendulum applied to a vehicle following an arc about the earth center at random velocity, as is the case in an earth-navigational system. As shown, the varying accelerations of its pivot has no affect on the Schuler pendulum's inherent tendency to remain at the earth's center or, if it were set in motion as shown in Figure 7b, to re-align itself to its original "null" point at 42 minute intervals as indicated in Figure 7c.

Of course, it is impossible to construct and utilize a 4000-mile arm pendulum to provide a reference for an earth-navigational system, *but it is possible to duplicate the effect of the pendulum by literally inducing this exact angular oscillation of the platform —a swinging motion of 84 minutes-per-cycle.* Any system, so controlled is said to be "Schuler-tuned."

The way in which this is implemented on the P-3 navigational system is as follows:

As the platform is aligned, a true vertical is established as indicated by Position 1 in Figure 7d. Thereafter, when the system is placed in operation some slight system discrepancy will invariably introduce an error which tilts the platform. The direction of the tilt always results in a gravity-induced acceleration signal and an attendant velocity change. This change exerts an influence on the precession rate which tends to oppose platform movement and the growth of this induced precessing force is carefully controlled* so it will arrest the tilt 21 minutes after it becomes evident.

**This control feature is the essence of "Schuler tuning." Tuning is a design function obtained by incorporating specific component values into the control loops of both the N-S and E-W axes (block diagram of E-W axis shown in Figure 7e) which will sustain the 84-minute oscillatory period.*

Then since velocity and precession force will continue to change until the tilt is eliminated, the platform is motivated towards level for the ensuing 21-minute period, and swings at a maximum rate back through the level position (depicted at Position 3. Figure 7d) and into the opposite excursion. Thus, once the initial tilt occurs, a continuous cyclic chain of events takes place: the platform tilt induces an acceleration signal which feeds a velocity signal which swings the platform at the exact radial rate that a Schuler Pendulum would produce.

The end result of the Schuler cycling can be illustrated by noting its influence on the previous example in which the uncorrected acceleration error resulted in a position error of 2 NM after 42 minutes and almost 8 NM after 84 minutes of flight. This system exhibited an accelerometer error of .00226-miles per minute, per minute. The cycling also induces acceleration errors. We will assume that the mean effect of the induced cyclic error is half the magnitude of the constant natural error and we will assume, further, the worst possible coincidence in which the first Schuler-induced error after takeoff actually augments the natural error instead of detracting from it.

This will cause the reported position to be half-again as far off-course, 3 NM instead of 2, after 42 minutes of operation. During the second half-cycle, the mean effect of the Schuler-induced error eliminates half of the natural drift error (.00226 divided by 2 equals .00113), and the net error accumulated during this leg of the flight is the equivalent of .00113x21x42, or .988 miles. This, added to the accumulated error in the first leg would place the Schuler-tuned aircraft exactly 3.988 NM off-course after 84 minutes of flight instead of 7.97 NM, and at this point 50% of the drift error inherent to this system has been eliminated. After 168 minutes of flight the error will be 8 NM with Schuler-tuning, 31.9 NM without it—the error has been reduced by 75%. It can be seen that the cycling becomes progressively more beneficial on subsequent cycles, and in fact Schuler-tuning on long flights represents the difference between practical accuracy and intolerable errors in the application of inertial guidance theory.

The Schuler oscillations do entail an undesirable feature in that they induce small oscillating acceleration errors in both the N-S and the E-W axes which are superimposed on the drift errors. This leads the system to report a position slightly in error (latitude and longitude) first in one direction, then in the other. Although the average of the reported positions is nearly true, the aircraft tends to weave in shallow excursions about the desired course, and some lost motion is involved. In the P-3 installation, the Doppler radar velocity signal, which is not subject to periodic excursions, is utilized to "damp" the platform oscillations, holding the amplitude of the swing to a minimum value without greatly disturbing its frequency.

ACCELEROMETER OUTPUT CORRECTIONS The arrangement of the accelerometers with their sensitive axes horizontal and perpendicular to one another is perfectly suited to navigating over a stationary flat plane or over flat terrain moving at uniform speed in a straight line. The earth, however, is a rotating sphere and insofar as the platform is concerned only points along the equator can be considered to possess uniform linear motion. Here, and only here, the accelerometer signals can be translated directly into position information. Since very few flights are carried out exactly along the equator, it is necessary to provide an automatic device which will alter the accelerometer signals in order for the system to report information that is meaningful to the map coordinate system. The corrective device is purely electrical. All or part of the circuitry is active whenever velocity signal voltage is present anywhere north or south of the equator. These circuits utilize the velocity signals, modified according to the latitude of the aircraft position, to "graft" artificial acceleration signals in the accelerometer output circuits.

There is a division in logic between the circuits—part being devoted to centripetal affect, part to Coriolis affect—but it should be noted that the corrections are complementary and the division of function is vague.

Although it is convenient to assign a separate purpose to the circuits as we have in the following discussions, the functions overlap and it is not entirely accurate to consider them as separate entities.

Centripetal Correction This function is distinct from that of the Coriolis correction in that it has no relationship to earth dynamics. If the earth were stationary, it would nonetheless be necessary to graft centripetal correction voltages on the accelerometer signals to obtain an accurate plot of any course that does not exactly coincide with one of the earth coordinate great circle routes, i.e., an exact polar orbit or an exact equatorial orbit. Every linear acceleration initiates motion on a great circle route, a route that ultimately circles the earth center, but unless the route is due north-south or directly along the equator the system is incapable of plotting it accurately from the "raw" accelerometer signals.

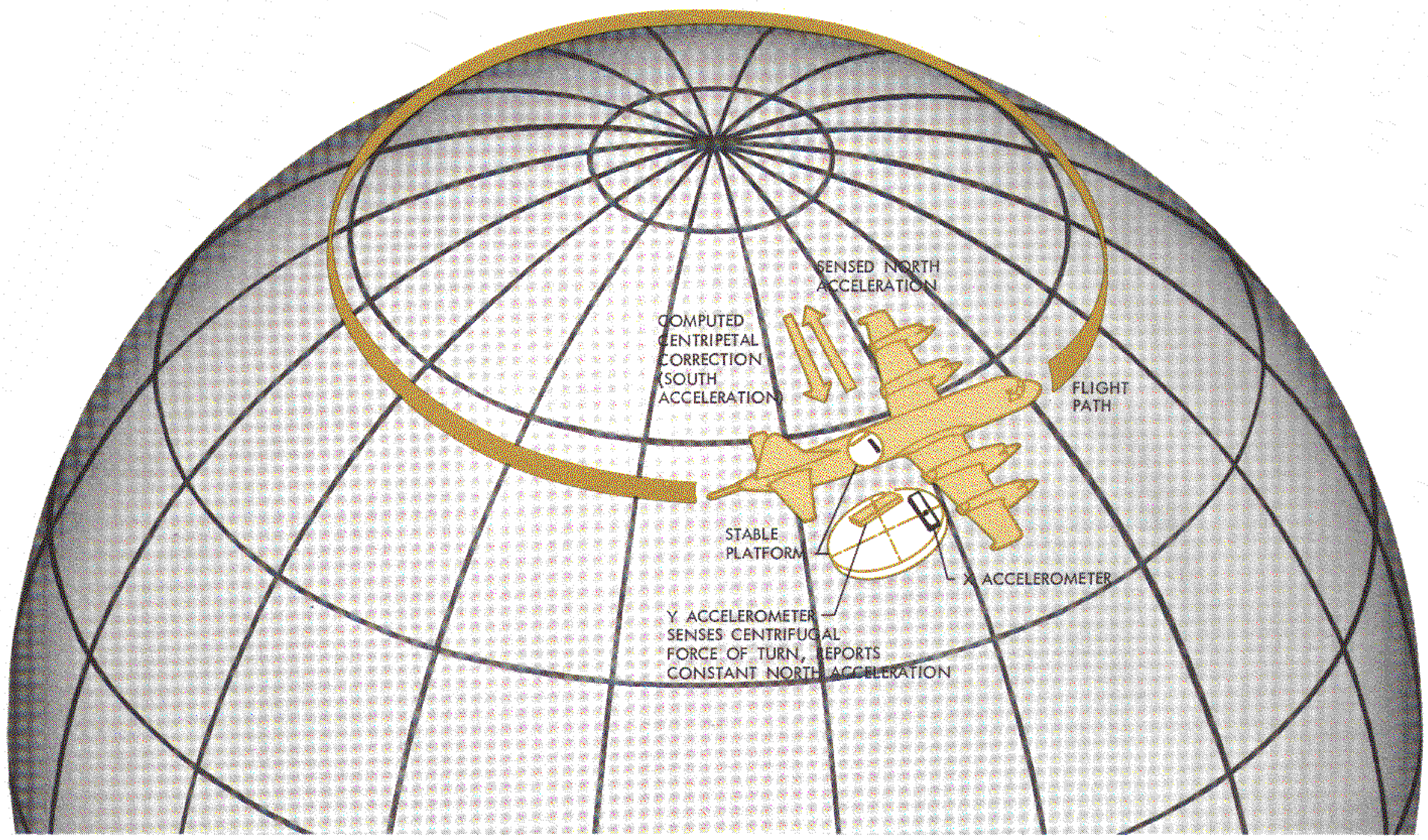


Figure 8a Centripetal Correction During Latitudinal Flight. Using the known factors (aircraft velocity east and latitude) computer supplies artificial south acceleration signal exactly cancelling sensed north acceleration signal, no N-S velocity is reported.

The orbit resulting from linear acceleration, and the logic of applying centripetal corrections to obtain an accurate plot of track can be visualized by imagining any simple circumstance involving a single acceleration and the inertial reaction. For example, if a perfect bowling lane were built completely around the earth at the equator, a bowler could theoretically stand behind the pins, roll the ball in the opposite direction, and (after a few years' time) get a perfect strike on the head pin. But if the lane were built on Latitude 10 N, the ball would invariably veer south, and after a few thousand yards it would fall into the right-hand gutter if it were rolled east, into the left-hand gutter if thrown west—the south gutter in both cases. The reason for this phenomena is revealed by considering such a lane built on a latitude in the Arctic only a few yards from the pole. Here the curve of the lane is obvious, and it is readily apparent that if the ball is accelerated due east it will in fact follow a straight course in inertial space that intersects the outside gutter. If the ball is to roll at uniform speed and remain on the alley, a uniform north-accelerating force must be exerted on the ball in transit. Note that, in this case the north acceleration is a corrective force and does not produce north velocity with respect to the alley.

As depicted in Figure 8a, the N-S accelerometer aboard an aircraft circling the pole finds the same phenomenon. In a steep-banked turn, the centrifugal force tends to deflect the N-S accelerometer, it blindly reports a steady north acceleration and a growing north velocity will be recorded when in fact only east velocity exists.

The fault does not lie in the accelerometers but in the non-linear pattern of the accepted geographic coordinate system to which the platform is slaved. The coincidence of earth axis and pole seems to implicate earth dynamics in the phenomenon, but actually this is not a factor. If the coordinate system were shifted to place the pole at New York and New York was used as the focal point of one accelerometer axis of an inertial navigator circling the city, the accelerometer sensitive to this axis would report acceleration towards it.

With a spherical coordinate system, any linear vehicle acceleration that initially affects both accelerometers will *never* result in the vehicle reaching the pole. The vehicle will follow a great circle track that first approaches one of the poles, flies due E or W for a brief period and then departs the pole as shown in Figure 8b.

The velocity resulting from any given acceleration is initially computed correctly with respect to space, but the direction of the speed is in need of constant alteration if the navigational system is to accurately report a great circle course that crosses both latitude and longitude. On such a course, the aircraft does not turn as it does following a line of latitude, and consequently does not generate uniformly false north acceleration signals. The centripetal correction circuit does, however, continue to "plant" signals of acceleration towards the equator. This has the affect of altering the reported speed (the resultant of velocity along both axes), and since no actual acceleration has taken place to cause a speed change, it is apparent that the centripetal correction circuit must simultaneously plant a positive acceleration in one axis if it plants a negative acceleration in the other. Therefore when a portion of the north or south velocity is canceled, the system must necessarily add a sufficient increment of E or W acceleration to allow the reported total speed to be unchanged while the reported direction of flight is "bent" towards the equator. As shown vectorially in Figure 8b, after a single N-E acceleration has occurred the north vector grows progressively shorter. Note that the resultant speed vector is always of the same magnitude.

In summation, it can be said that an east or west acceleration in either hemisphere contains a "hidden" element of acceleration towards the equator, and the centripetal correction simply acts to reveal this element. If the aircraft velocity is any but due north or south, in the northern hemisphere the centripetal correction "manufactures" a south component of velocity. In the southern hemisphere it manufactures a north velocity.

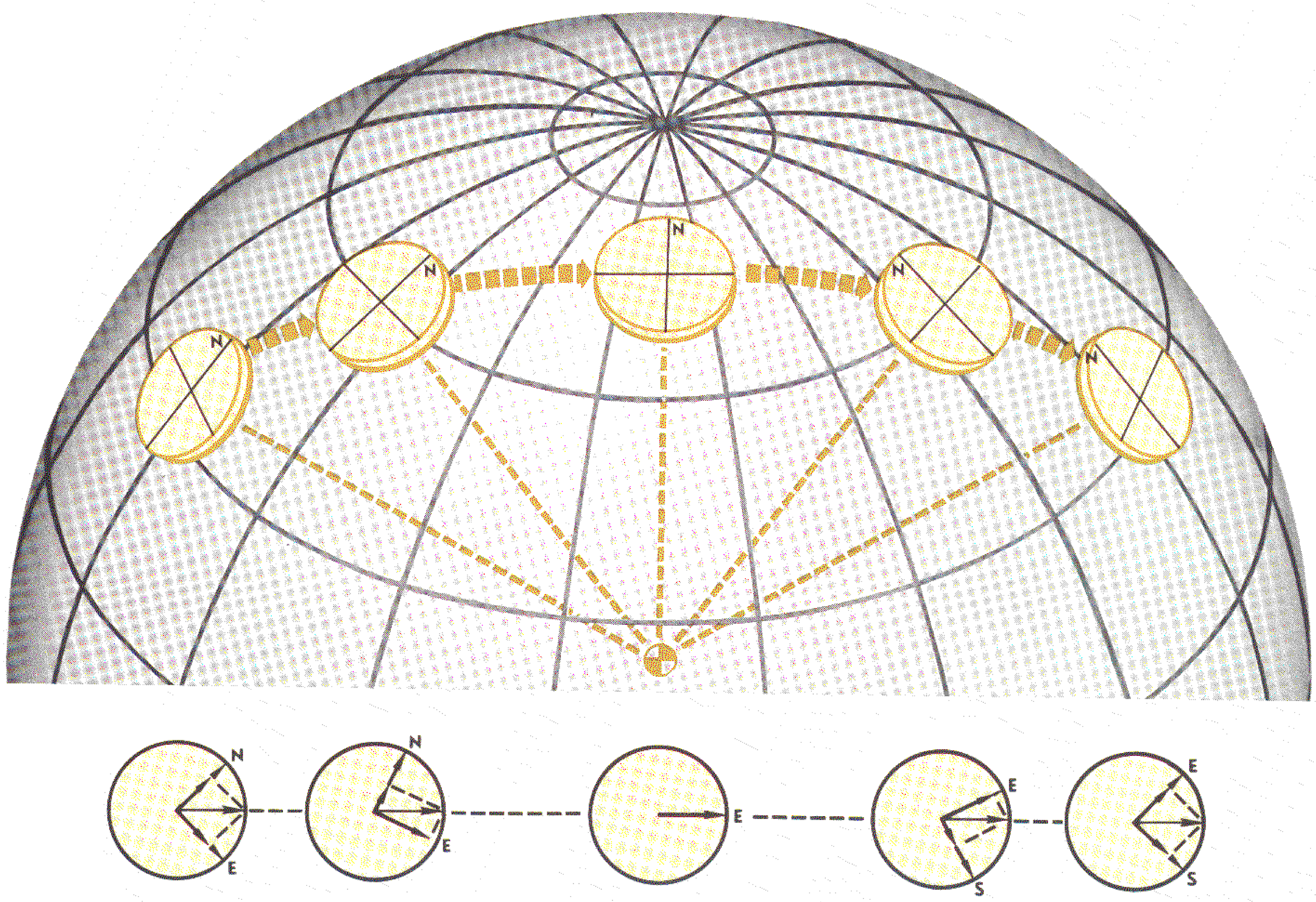


Figure 8b Platform Orientation Along a Great Circle Route Resulting from a N-E Acceleration. Insets show changes in velocity vectors produced by continually recomputed correction signals.

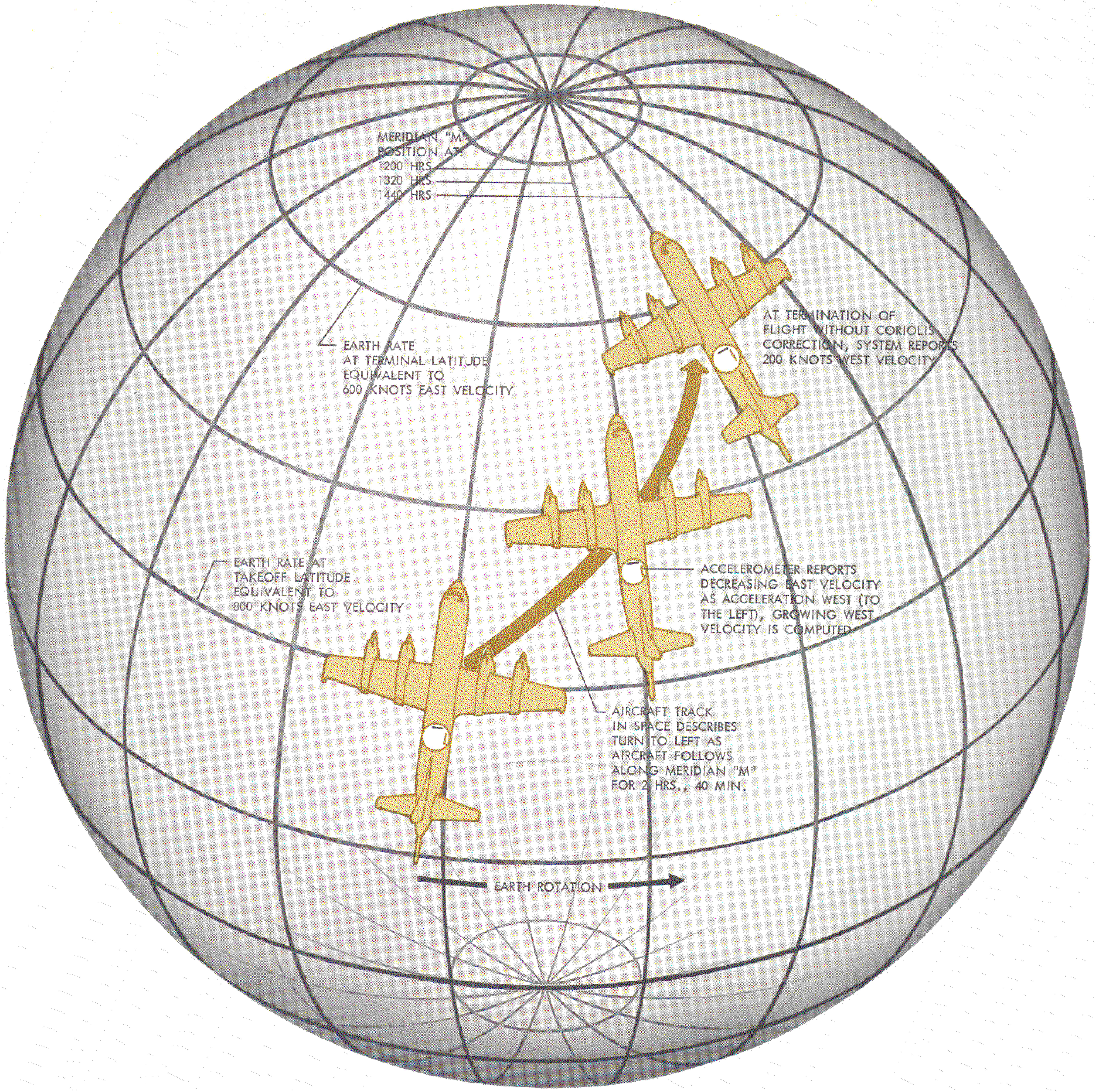


Figure 8c Reaction of Inertial System During Flight Due North Along Meridian "M" Without Coriolis Correction. If Coriolis correction circuits are activated, false west acceleration signal would be voided by artificial east acceleration signal, no east or west velocity would be computed.

Coriolis Correction As mentioned previously, the scope of the centripetal correction is limited to relating the effect of linear acceleration to the resulting track over a sphere, and it makes no allowance for earth dynamics. Since the earth rotates towards the east, all points on the surface possess a constant radial velocity which is maximum along the equator and progressively less at higher latitudes.

Although it is true that the earth has a trajectory in space, this motion is of no importance to the inertial navigation system for it is shared by every point on the sphere. The only variable involved is the variation in earth rotational speed at different latitudes. This variation must be taken into account, and the accelerometers do not automatically make allowance for it.

The need for such an allowance is graphically illustrated by considering an example in which an inertial navigator, totally devoid of any Coriolis corrective mechanism, is put aboard a train in the northern hemisphere, aligned, and transported north. If we assume that the train is located at a latitude where the earth rate is 800 knots east during the alignment, it is obvious that the train's eastward velocity must be constantly reduced as it progresses north. This is tantamount to saying an acceleration is occuring.

Since the train is constrained by the track, a force from the east will be exerted on the wheel flanges. This force is sensed by the E-W accelerometer as acceleration to the west and a growing west velocity is computed. As indicated by Figure 8c, an aircraft flying north directly along a moving longitude meridian will subject the E-W accelerometer to this same force as its course in space is altered to the left to compensate for the decreasing eastward speed of the earth surface.

If we assume that either vehicle travels north and stops at a latitude where the earth rate is 200 knots less than the earth rate at the point of its initial alignment, the velocity computer will continue to report that the vehicle is traveling west at 200 knots even though it is perfectly static.

The Coriolis correction prevents this ludicrous circumstance from occurring by manufacturing exactly enough east acceleration signal to offset the continual west acceleration signal generated by the E-W accelerometer as it travels north. The east and west accelerations cancel, and no change in longitude is reported. Of course, if the aircraft makes the return trip, the platform begins the trip under the delusion that 600 knots east is zero velocity, and as it travels south the increasing earth rate leads it to constantly report east acceleration. In this case, the Coriolis correction also reverses, that is, it manufactures a west acceleration that exactly voids the east acceleration.

Note that, in both of the above cases, *the Coriolis correction has supplied an artificial signal of acceleration to the right side of the actual track. This is the nature of the Coriolis correction, regardless of the direction of travel in the northern hemisphere.* When the track is due east along a parallel of latitude as illustrated in Figure 8d, the centripetal correction offsets the increment of north acceleration generated by the aircraft speed over the earth; the Coriolis correction accounts for the additional force generated by earth rotation. For example, if the earth rate at Latitude "L" is assumed to be 700 knots and aircraft speed is 350 knots, it is obvious that the total speed is 1050 knots. Centripetal correction offsets the acceleration due to centrifugal force of a 350-knot turn of this radius; Coriolis correction offsets the force of a 700-knot turn. The two corrections are additive in this case.

Reversing the course and flying west, the corrections become opposite in polarity—the centripetal signal is still south but Coriolis is north (to the right of the track)—and therefore only a part of the larger correction is effective.

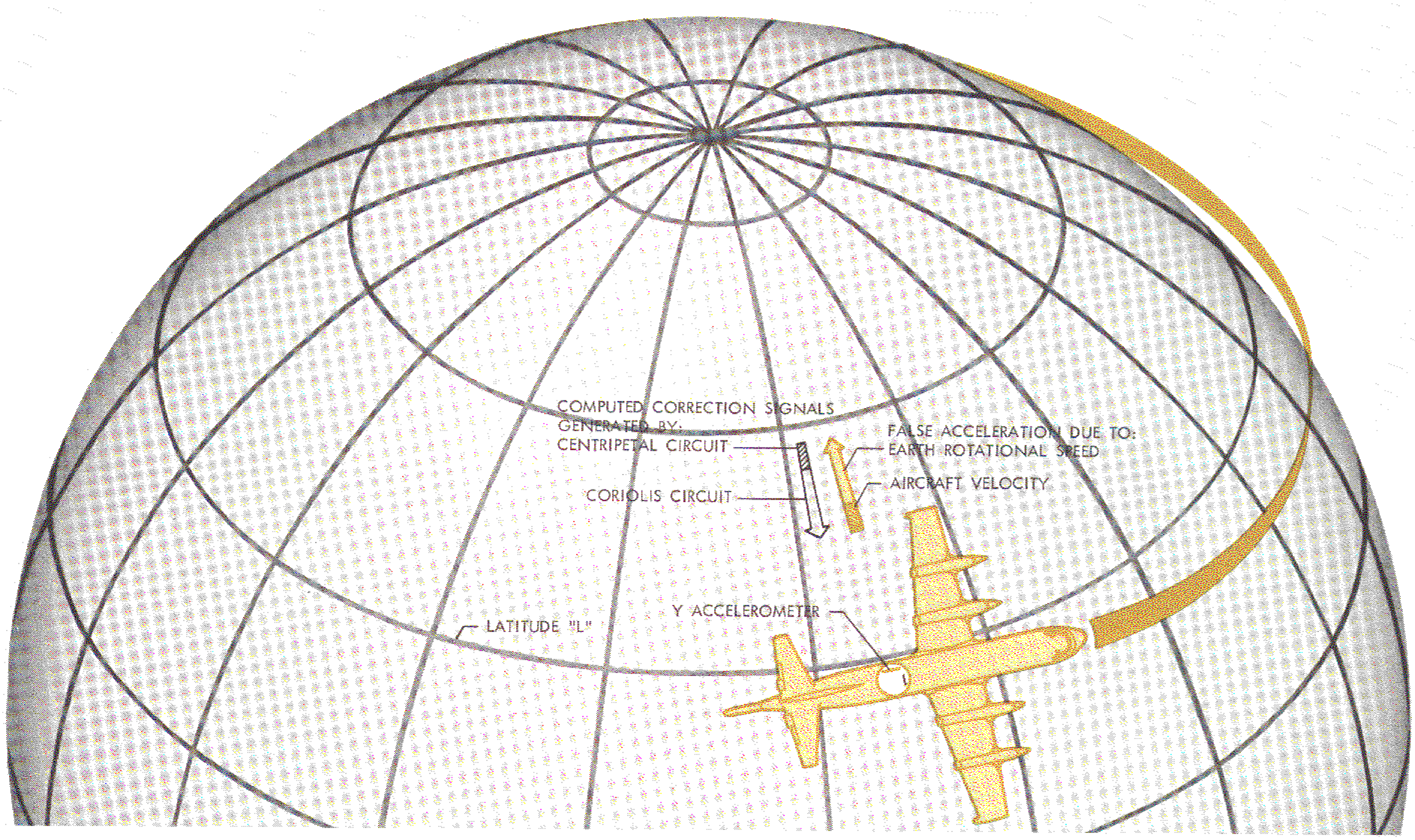


Figure 8d Combined Coriolis and Centripetal Corrections in East Flight. Earth rotation coincident with vehicle track causes Y accelerometer to sense additional increment of false north acceleration; Coriolis correction complements centripetal correction, eliminates total north acceleration error.

The inertial system components reviewed in the foregoing discussion are typical of those presently used in high speed aircraft. The accelerometers are the primary source of information for the system; the remaining components are in existence primarily to establish and maintain a frame of reference for the accelerometers. The attitude and heading information are usually obtained from the system as by-products. From the information supplied by the accelerometers, distance-traveled and position information are calculated in the computer for readout and feedback. Thus, the carrying vehicle is able to instantaneously compute and display its location, velocity, heading, and attitude.

PART TWO

EDITOR'S NOTE

Detailed information as to the theory of operation, alignment, and maintenance of all ASN-42 equipments is given in NAVWEPS 01-75PAA-2-7.3, NAVWEPS 05-35KAA-24 and NAVWEPS 05-35KAA-25 manuals, but it is hoped that the short resume of the more important aspects given in the following pages will provide a better general knowledge of the system, especially for those who are not electricians or navigators.

INERTIAL SYSTEM DESCRIPTION

In its usual application, an inertial system is used only to report the progress of ground track. The P-3 inertial system, however, falls into a somewhat broader category in that its interconnection with other navigational facilities enhances its capabilities.

The precise engineering and careful adjustment that maintains stable platform alignment provides a bonus on the P-3 installation, for the platform provides an accurate reference from which attitude (pitch and roll) and true heading can be measured; signals needed to orient other components not directly concerned with long-range navigation.

All inertial system units are accessible within the P-3 cabin or flight station areas (see Figure 9). Only one control unit, the Inertial Navigator Control, is located in the flight station (near the aft right-hand side of the Pilots Control Pedestal); all other indicators and controls, including the Position Indicator, the Alignment Control, and an auxiliary unit, the Doppler Decoupler control, are situated on the vertical instrument panel at the Navigator's station. The other five system units and one auxiliary assembly, the AU 0400 Converter-Decoupler (Power Supply and Relay Box), are housed in the aft left-hand electronic rack—one of the closed electronic cabinets ventilated by the cabin air exhaust to hold the environmental temperature below 130° F.

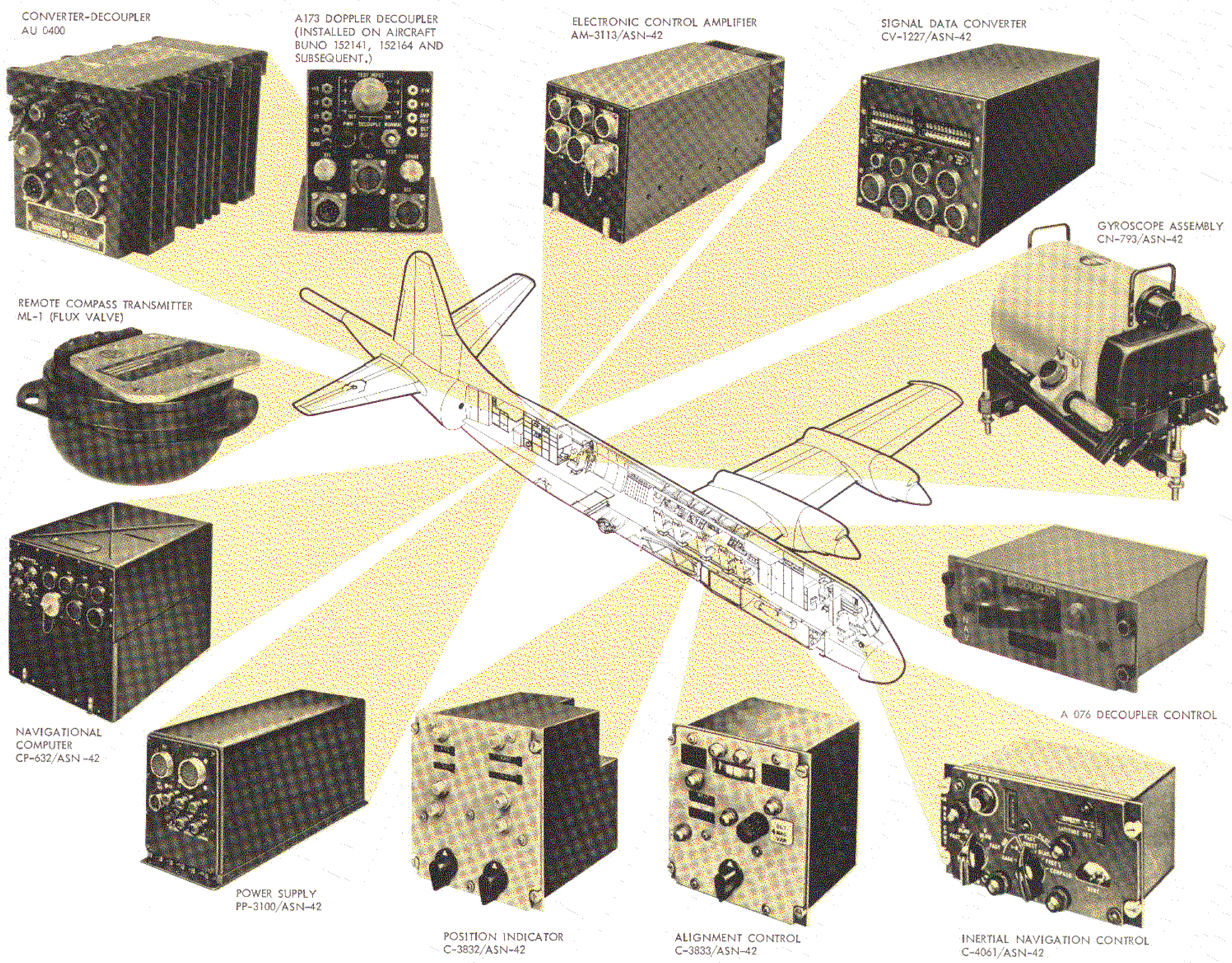


Figure 9 Inertial System Unit Location in Aircraft

The capacities of the Orion's ASN-42 Inertial Navigation System are utilized in two basic operating modes by selecting either NAV/I-D (Inertial-Doppler) or PILOT/FAST ERECT*-SLAVE-FREE-COMPASS at the Co-pilot's Inertial Navigator Control mode selector switch (see Figure 9). This article is chiefly concerned with the Inertial-Doppler capabilities, and while the three subdivisions of the PILOT mode are important added navigational services, we will touch on these only briefly.

The three PILOT modes produce essentially the same type of information provided by conventional attitude heading reference systems, such as the MF-1 used in the P-2 Neptune, and the supplementary ASN-37 or ASN-50 Attitude Heading Reference System (AHRS) used in the P-3 Orion.

The SLAVE mode is primarily intended to supply attitude, magnetic heading, and true heading information for use by the Pilot on local flights or on those cross-country flights where radio navigation aids are available, and a navigator is not aboard. In this mode, magnetic heading is derived from the remote compass transmitter, but the erratic oscillations native to the flux valve signal are not evident, due to an interconnection with the gyro-stabilized inertial platform. True heading is derived by combining magnetic heading with manually-inserted variation.

The FREE mode also provides attitude and heading information but heading is "free" of the magnetic influence of the flux valve. Heading need not be measured in relation to the north pole, for the source is purely gyroscopic and is related to the stable platform's heading orientation in space. Any desired datum for heading can be selected—by the SLEW control—and a course established with reference to that datum. FREE mode will normally be used when employing Grid Navigation techniques at latitudes above 70° (polar regions) or in certain other areas where the earth's magnetic field is too unreliable to use the SLAVE mode.

The COMPASS position of the IN Control switch is an *emergency* mode to be selected when the inertial system is inoperative in its more accurate modes. The single, undamped output — unstabilized magnetic heading—reflects the oscillations of the pendulum-mounted flux valve which are induced by aircraft maneuvers.

**FAST ERECT position of this mode switch is not actually an operating mode. It is used for Pilot mode alignment purposes only, and provides no useful operating information.*

Operation with the mode switch in the NAV I-D position exploits the maximum capabilities of the system to produce continuous indications of latitude and longitude on the Position Indicator—thus performing its prime function as an *aid* for the Navigator in directing the aircraft from point to point by the shortest possible route.

As explained in Part 1 of this article, the inertial platform is a uniquely stable device that is perpetually aligned to the local vertical and oriented in azimuth to the frame of reference—the earth coordinate system of latitude and longitude. Then, since all longitude meridians converge at the true north pole, synchro transmitters attached to the stable platform can electrically measure—and convey to display instruments—the angle between true north and whatever heading the aircraft may take. In a similar manner, transmitters placed between the stable platform supporting structure (gimbals) and the leveled platform measure the aircraft attitude with relation to the plane of the earth's surface. Figure 12 depicts the relative location of these transmitters on the Gyroscope Assembly, and Figure 10 indicates the basic signal flow and illustrates the instruments and avionics to which the inertial system is interconnected.

The inertial system is directly or indirectly interrelated to most of the P-3 avionics systems. To better understand its function and operation, it is helpful to be generally familiar with the nature of the information produced and the form in which it is used or presented.

As indicated by Figure 10, 115-V ac power for the Inertial Navigator system is provided by two multiple-circuit breakers, labeled HEATER and RUN, located on the aft left-hand electronic rack circuit breaker panel. Two additional flight station breakers supply 28-V dc power to the AU 0400 Converter-Decoupler*. One dc breaker is labeled INERTIAL NAV PWR and is located on the Flight Essential DC Bus panel; the second breaker—entitled INERTIAL NAV PWR RELAY—is located on the Extension Main DC Bus.

The items enclosed in the colored area on Figure 10 are the inertial system and its three auxiliary units, and all others are the indicators and systems from

**At the time of this writing, all production aircraft (BUNO 152141, BUNO 152164 and subsequent) are equipped with a Lockheed fabricated unit (A173 DOPPLER DECOUPLER) to replace the AU 0400 Converter-Decoupler. Additional details concerning this unit and the relocation of inertial system circuit breakers are presented near the end of this publication.*

which the inertial system receives, or to which it supplies, information. For the sake of simplicity, aircraft signal-distribution relays and switching are omitted. The distribution illustrated in Figure 10 occurs when the manually operated control switches are as shown: the Pilot's and Co-pilot's HSI HDG switches are set to INERTIAL position, the Co-pilot's Attitude switch is set to NORMAL INERTIAL and the Navigator's NAV SYS mode switch—located on the Doppler/Air Mass Position Indicator—is set to I-D. The NAV SYS mode switch selects the source of attitude, true heading, and distance-traveled information to all tactical equipment.

True heading is measured electrically by two synchro transmitters mounted on the azimuth gimbal within the Gyroscope Assembly. One true heading transmitter normally provides a signal to drive the Doppler-Air-Mass true heading shaft; the second performs the same function in the inertial system Signal Data Converter (SDC). The SDC true heading servo motor drives two repeater transmitters, for the extra capacity is needed to supply the rather heavy demands of the many aircraft avionics systems and instruments. The true heading loads of one repeater transmitter include the compass dials of the individual Bearing-Distance-Heading-Indicators (BDHI, at the Pilot, Co-pilot, Navigator, and Tactical Coordinator stations), the radar station heading indicator, the search radar heading control, the error voltage monitor, and, on later aircraft (BUNO 152140 and subsequent), the Electronic Counter Measures (ECM) indicator. The second repeater transmitter supplies true heading to two units: the heading indicator on the Navigator Ground Track Plotter and the rotatable, illuminated heading arrow mounted on the tracing crab of the Flight Station Plotter.

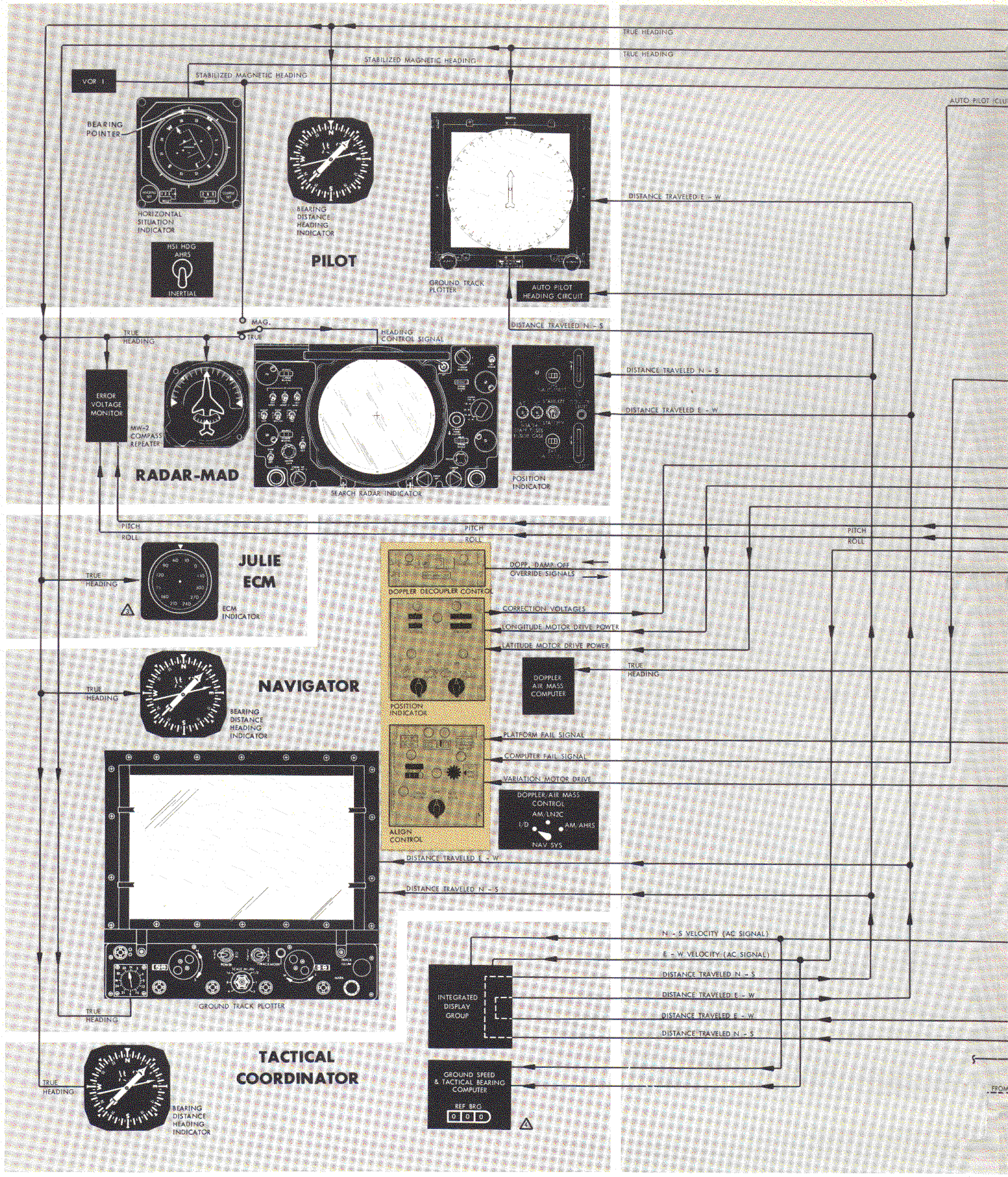
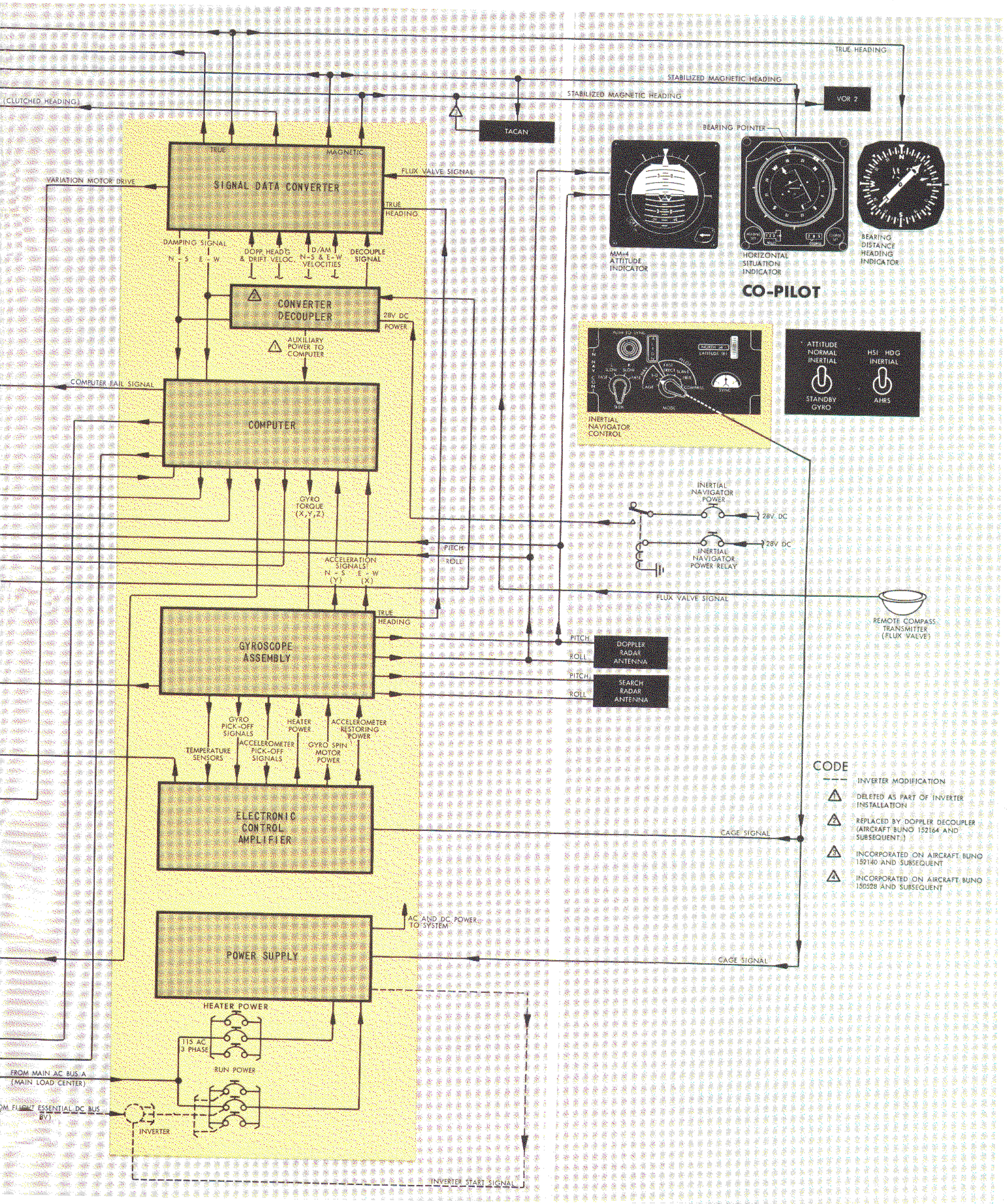
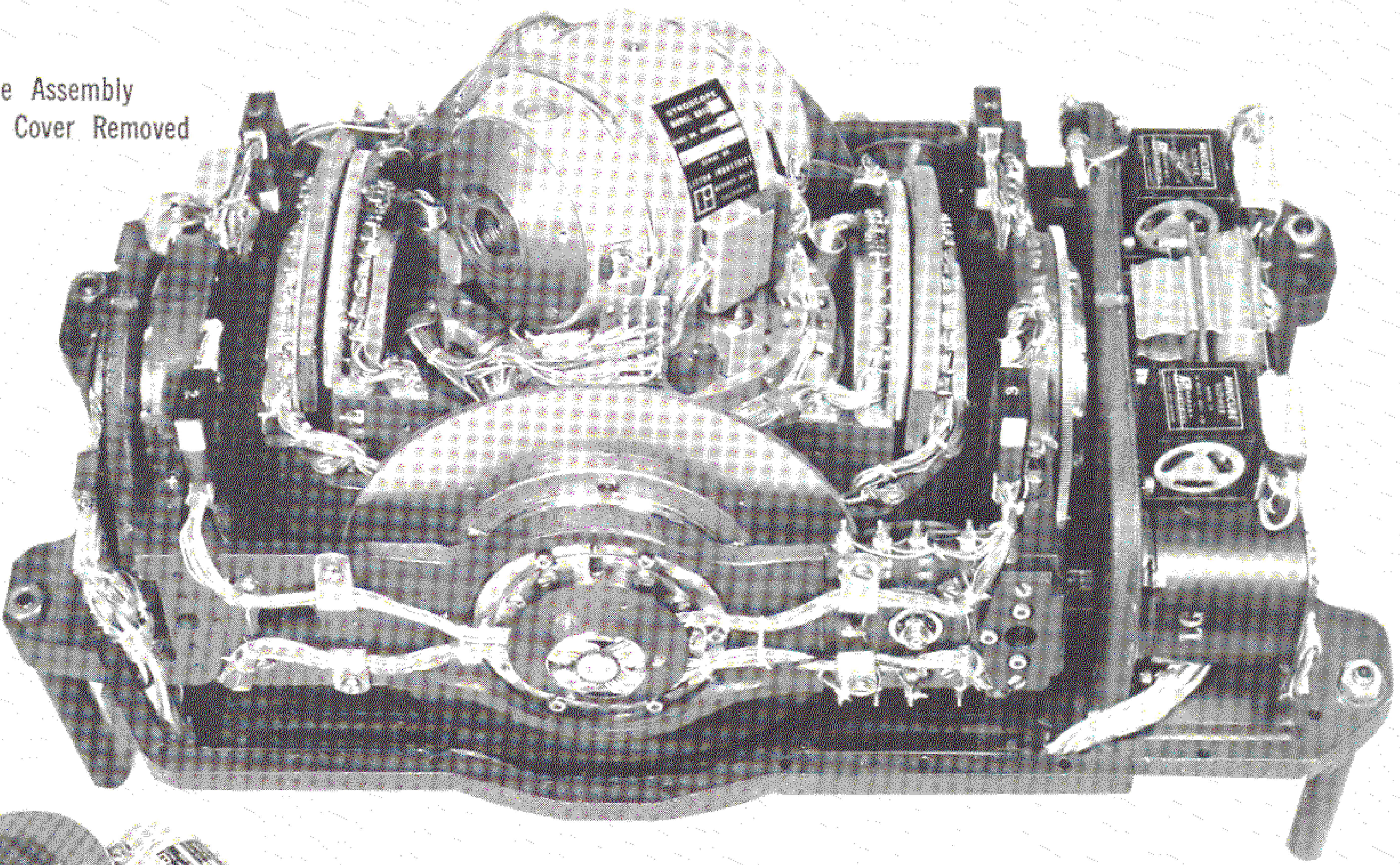


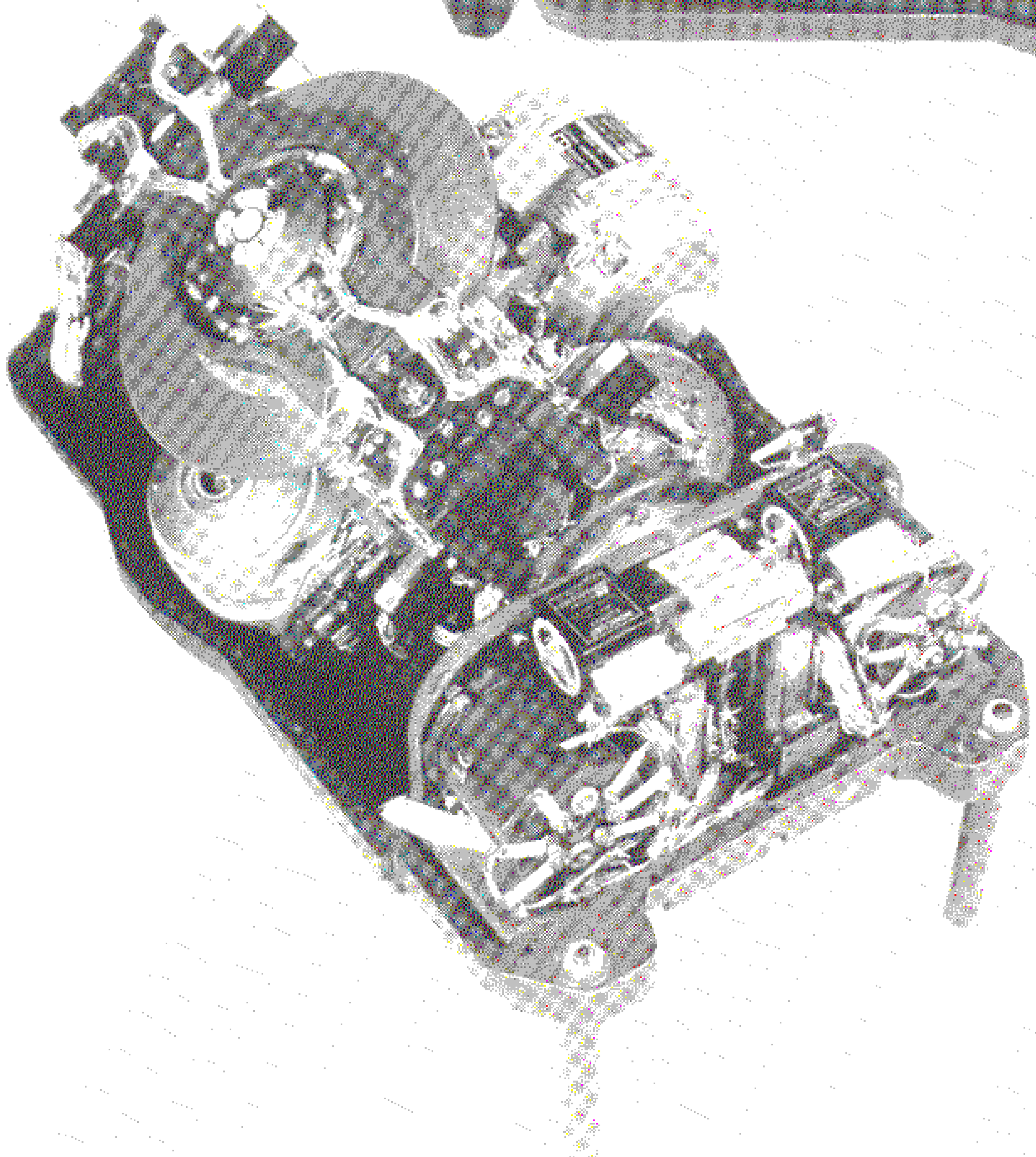
Figure 10 Inertial System Intercon



nnection and Signal Distribution

Figure 11
View of Gyroscope Assembly with Shroud and Cover Removed





As shown on Figure 12 attitude signals are taken from four synchro transmitters (two pitch and two roll, one at each of the gimbal pivot points) through aircraft switching directly to aircraft instruments and avionics.

The output of one pitch and one roll synchro transmitter is channeled through ships wiring to the Search Radar antenna control circuits. When the aircraft maneuvers, these signals can pitch and roll the radar antenna as much as 30° in a direction opposite the aircraft motion to maintain the selected horizontal scanning attitude, thus preventing targets on the radar video indicator from shifting erratically or being lost altogether.

The second pair of pitch-and-roll synchros control the Co-pilot's MM-4 attitude indicator, presenting a constant display of aircraft flight attitude. The same pitch and roll signals stabilize the Doppler radar antenna beams with respect to the earth's surface and are also routed to the Error Voltage Monitor at the Radar-MAD Operator's station where either the pitch or the roll signal can be selected for recording.

From the standpoint of an inertial system, the two most important outputs from the Gyroscope Assembly are the acceleration north-south and acceleration east-west voltages. The Computer processes (integrates) these two inputs to produce N-S and E-W velocities which in turn are the basic signals for deriving *all* of the computer outputs shown on Figure 10 and described here.

The X, Y, and Z* gyro torqueing voltages, indicated on Figure 10 as a single output, represent three separate signals. They are developed by the Computer from the above mentioned velocities—modified by necessary corrections—and utilized in the Gyroscope Assembly to maintain proper orientation of the stable platform with respect to gravity and true north.

Two separate circuits within the Computer integrate north-south and east-west velocities to produce north-south and east-west distance traveled voltages.

**X, Y, and Z gyro torqueing voltages are applied to the gyros to cause them to move* about *that particular axis, i.e., X torque produces rotation of the upper gyro (and the stable platform) around the East-West (X) axis of the system, etc.*

Distance traveled outputs are interconnected to four P-3 avionics systems: first to the ASA-16 Integrated Display Group at the Tactical Coordinator's station —where manually inserted corrections can be added —and then to the flight station Ground Track Plotter, the Navigator's Ground Track Plotter, and the Radar Station Position Indicator.

Each of the two plotters provides a graphical trace showing the distance and direction the aircraft has traveled over the earth's surface (ground track). The total distance traveled in any direction that can be recorded without resetting the stylus is somewhat limited—a fact that generally restricts the use of the plotters to tactical maneuvers rather than long range point-to-point navigation.

The radar operator's Position Indicator does not actually yield a direct indication of geographic position in terms of latitude-longitude. Its front panel counters display the N-S and E-W nautical miles traveled from any geographic point the operator desires, ranging up to 150 NM from a central datum. The counters have manual controls which are used to make the initial setting, but otherwise the data displayed on the counters have little practical importance. The primary purpose of the Position Indicator unit is to convert the distance traveled ac signals into dc voltages that ground stabilize the search radar display to a specific section of terrain. Since the maximum range of each counter is 300 NM in each axis, the Position Indicator is normally used only for tactical situations.

Latitude and longitude are presented on motor-driven counters on the inertial system Position Indicator. The computer processes N-S and E-W velocity through separate integrators and incorporates the power amplifiers and most of the associated circuitry needed to drive the two indicator motors. The two controls located on the Position Indicator front panel allow the operator to set in the proper initial latitude and longitude, or to revise the indications if more accurate information becomes available.

All of the computer outputs described to this point in our discussion are produced from the dc velocity signals. The computer also generates ac signals representative of N-S and E-W velocities which are routed through aircraft relays to the ASA-16 Integrated Display Group, and on later aircraft (BUNO 150528 and subsequent), to the Ground Speed and Tactical Bearing Computer. The Ground Speed and Tactical Bearing system resolves ac velocities into two additional signals: one signal of *aircraft true course* can be displayed on the REF BRG counter located on the face of the Tactical Bearing Computer and is also utilized by the Computer to determine tactical bearings. The second signal is aircraft *ground speed* for use by the armament store Intervalometer.

Figure 10 indicates that a part of the Computer's function involves the use of "damping" signals from the Signal Data Converter. Damping signals are derived from the Doppler Radar System Heading and Drift velocities and are shown as incoming signals to the Signal Data Converter.

The "damping" voltages are mixed (summed) with accelerometer outputs and also with the computed velocity values to decrease the amplitude of Schuler oscillations to a more satisfactory level. To guard against dampening of the inertial system with erroneous Doppler velocities, the Doppler Decoupler automatically disconnects any widely divergent "damping" signals and thereby prevents serious degradation of the inertial system indications.

Although the P-3 inertial system is independent of any magnetic heading reference in the navigate mode, a remote compass transmitter (flux valve) is incorporated into the circuit to provide magnetic heading data and thereby broaden its function. A substantial portion of the P-3's operation involves the use of radio navigation aids, such as TACAN and VOR stations, that transmit magnetically oriented information. To provide the flight station instruments with directly comparable data the inertial system produces magnetic heading outputs in I-D mode as well as in the Pilot modes.

During the first minute or so of the alignment cycle, the electrical signal of magnetic heading from the flux valve is combined in the Signal Data Converter with a signal of variation (inserted manually on the Alignment Control) and the corrected signal is used to "rough align" the platform to an approximate true north. During operation, the Signal Data Converter blends this magnetic heading signal with a true heading signal supplied from the stabilized platform to compute magnetic variation and three stabilized magnetic heading outputs.

The three stabilized magnetic heading outputs originate on a single servo driven shaft and include one to the autopilot for holding the aircraft on a selected heading. A second stabilized magnetic heading signal positions the compass cards on the Pilot's and Co-pilot's Horizontal Situation Indicators (HSI) and is routed to the TACAN radio navigation sys-

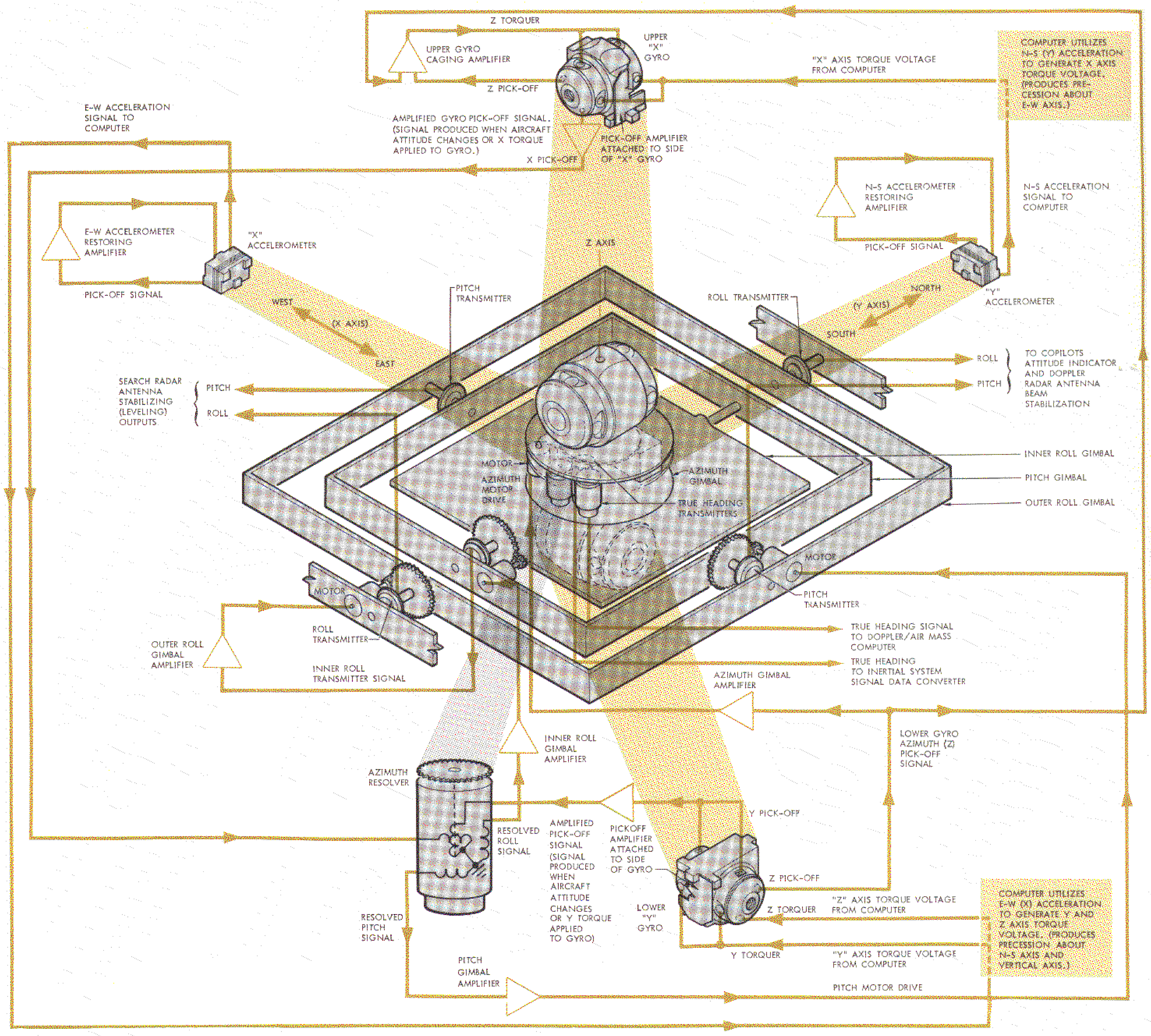


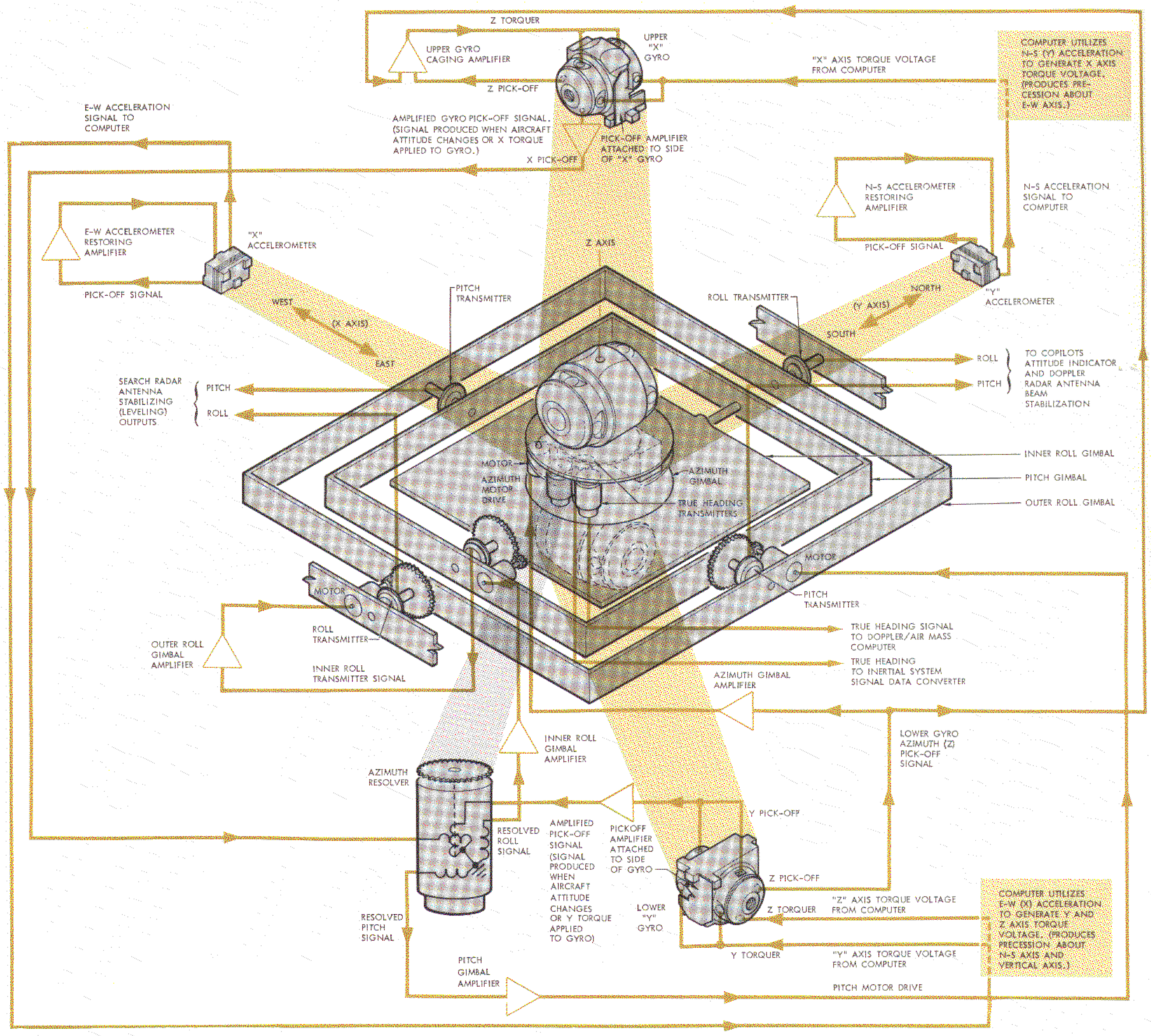
Figure 12 Electro-mechanical Schematic of Gyroscope Assembly and Electronic Control Amplifier

tem* to generate a magnetic-bearing-to-TACAN-station output for display on both HSI *bearing* pointers.

A third transmitter output is combined with both Visual Omni-range (VOR 1 and VOR 2) signals to indicate magnetic-bearing-to-VOR station; either of which can be selected as alternates for TACAN bearing. As indicated on Figure 10, the third transmitter output is also an alternate source for positioning the Search Radar antennas and video display with reference to magnetic north.

The magnetic variation signal is the difference between true and magnetic heading expressed in degrees. Variation is computed in the Signal Data Con-

**On BUNO 152140 and subsequent aircraft, the TACAN source of stabilized magnetic heading has been removed from this transmitter and is connected to the transmitter supplying the two VOR systems.*



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verter, and the derived signal is amplified to drive the variation motor in the Alignment Control. While operating in the I-D NAV mode this computation is automatic, and a constantly up-dated variation signal is supplied to the Doppler/Air Mass (D/AM) computer* and indicated on the inertial system Variation counter. If one of the PILOT modes are in use, the correct value of variation is obtained from navigation charts and must be inserted manually. In the event the inertial system is de-energized, manually inserted variation is still available to the D/AM computer simply by pressing and rotating the SET MAG VAR knob on the Alignment Control to obtain the desired counter reading. In addition to variation, the Alignment Control displays alignment indication (ready meter), failure indications (platform, computer), and provides partial switching for I-D mode alignment.

INERTIAL SYSTEM COMPONENTS

The Inertial Navigator (I-N) Control is used to energize (CAGE) the equipment and to select and control the mode of operation. The CAGE position of the MODE switch is a spring loaded momentary position (selected by raising and rotating the knob) that energizes the Power Supply and starts the automatically controlled timing circuits in the Electronic Control Amplifier and the Computer for system alignment in either I-D or one of the PILOT modes.** Other switches and indicators on the I-N Control are used only in the PILOT modes and are similar in function to corresponding controls on attitude-heading-reference systems.

The inertial system Power Supply converts 115-V ,3-phase, ac power into a total of thirteen ac and dc voltages to supply power to the various operating assemblies (amplifiers, motors, and synchro-transmitters) throughout the system. The power supply includes self-monitoring features to prevent momentary power surges at the aircraft busses from de-energizing the system. In addition, as means of protection for the remainder of the system, all dc voltages developed within the Power Supply are monitored by a special logic circuit to insure that the entire system will be shut down if any *one* of the 6 dc output voltages is missing or out of tolerance.

**The D/AM Computer combines variation with magnetic heading from the Attitude Heading Reference System to compute an alternate true heading signal. This true heading source can be selected for aircraft distribution by placing the NAV MODE switch on the D/AM Indicator to AM/AHRS position.*

***An inertial system inverter energized in CAGE is installed as part of a modification program now in progress.*

Gyroscope Assembly and Electronic Control Amplifier
The Gyroscope Assembly is secured by a special mount (shown attached to the unit in Figure 9) that allows it to be precisely aligned to the aircraft axes. It is housed in the forward compartment of the Aft Left-Hand electronic rack just above floor level. A cooling fan draws air through openings in the unit's fiber glass shroud when the internally sensed temperature exceeds 138°F. The shroud is lined with a mu-metal shield to exclude stray magnetic fields from sensitive internal circuits.

The Gyroscope Assembly outer case is purged, then filled with dry nitrogen and sealed to prevent contamination and moisture formation. When power is applied to the system HEATER bus, thermostatically controlled internal heaters maintain the nitrogen gas temperature in the 134° to 138°F range.

The Gyroscope Assembly and Electronic Control Amplifier (ECA) are closely associated in operation and will be considered as an operating unit. This unit is shown in Figure 12. The illustration is purposely simplified to eliminate the case, cover, heaters, and mechanical details of gimbal structure not essential to a basic operational discussion and is drawn to represent all four gimbals as they would appear when the Gyroscope Assembly is operating in a level attitude and traveling true north.

Some components of the stable platform are illustrated in their normal position and are also shown removed in order to show more clearly the electrical circuitry to these components.

Excepting the X and Y pickoff amplifiers, all the amplifiers shown on Figure 12 are located in the Electronic Control Amplifier. The outer roll gimbal is driven (slaved) to the same roll position as the inner roll gimbal* by a signal from a transmitter at the inner roll gimbal pivot point. Only one drive motor per gimbal is indicated on Figure 12, but two are actually used to minimize the motor size and reduce gear backlash, as well as to distribute torsional loads on the gimbals.

**The inner roll gimbal is redundant except for an extreme pitch maneuver which would result in the outer roll gimbal axis becoming coincident with the Azimuth (vertical) axis. This would essentially eliminate one degree of freedom (produce gimbal lock) and the platform would no longer be free floating in respect to the airframe.*

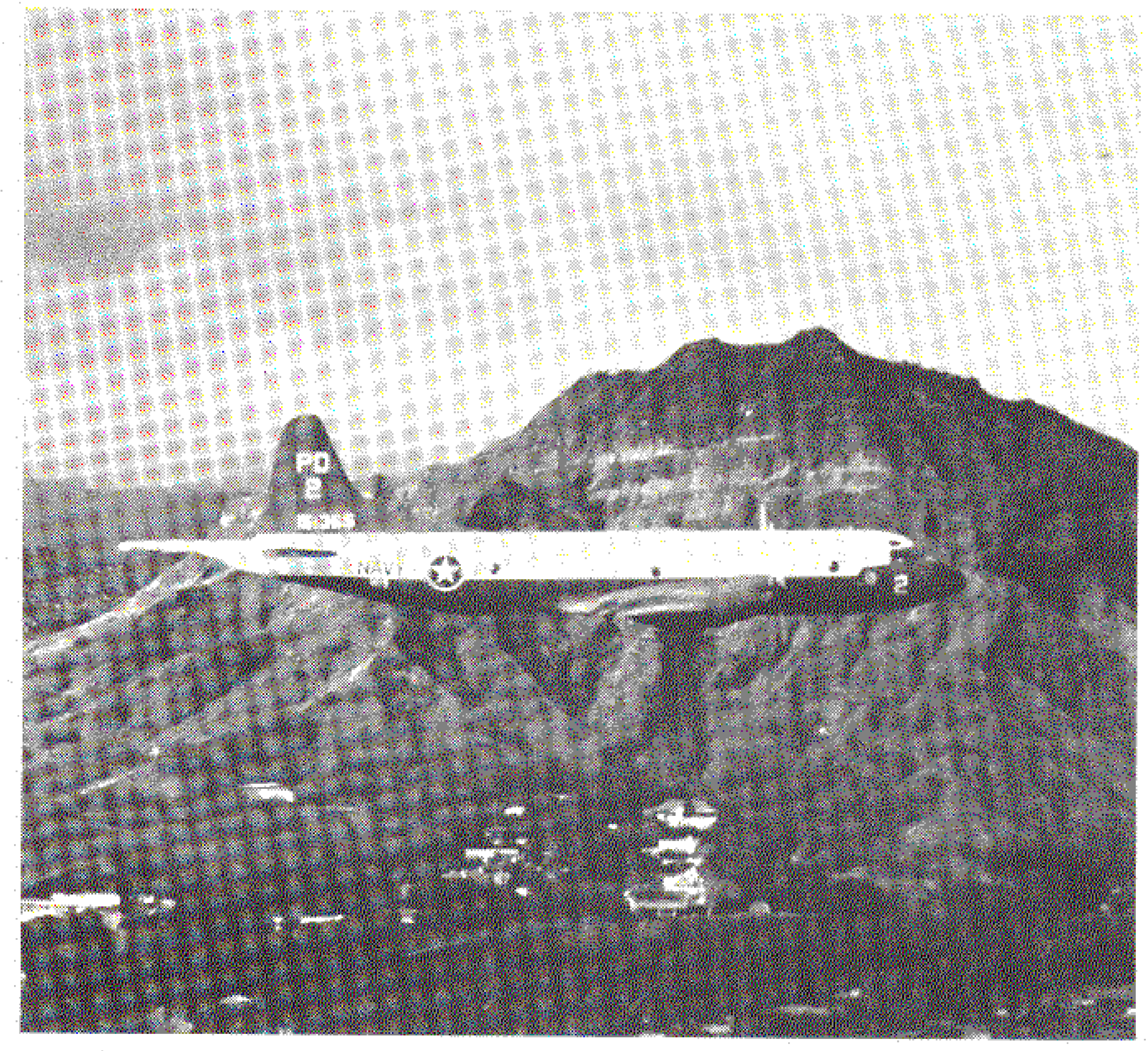
anical stops. Contact with the stops would prevent further motion about the axis, i.e., the gyro would lose one degree of freedom. The pickoff coils which detect vertical displacement forestall this condition by sensing a very small movement of the float. The sensed signal is amplified and used to drive motors which reposition the proper gimbal (pitch, roll, or both). As the motor drives the gimbal, the gyro outer case also moves and the excitation coils fixed on the outer case are recentered about the pickoff coil at the zero-signal position.

The X or Y pickoff coils also produce a signal when the aircraft maneuvers about its pitch or roll axis. For example, if the aircraft pitches nose up while flying true north, the upper (X) gyro, shown on Figure 12, with its spin axis lying along this line of flight, will remain in the original attitude (fixed in space) and the gyro case tends to pitch with the aircraft. The vertically oriented pickoff coils, however, sense the pitch movement immediately and generate a signal which drives the pitch gimbal in a direction (in this case, forward end down) to prevent the pickoff coil ever moving more than a fraction of a degree from its center (null) position.

In the above example, if the aircraft rolls, the lower (Y) gyro pickoffs generate a signal which drives the inner roll gimbal in a direction opposite to that of the roll to keep the pickoff nulled. Of course, when the direction of flight is true east or west the role of the gyros is reversed—the upper (X) gyro is perpendicular to flight, senses aircraft roll, and has full control of the roll gimbal, while the lower (Y) gyro is parallel to flight and senses pitch. For any combination of north-south and east-west travel both gyros are affected by either maneuver, and the two pickoff signals are mixed (resolved) by the Azimuth resolver (see Figure 12) into separate pitch and roll correction signals to overcome the effect of any given aircraft maneuver.

The second (vertically oriented) pair of torqueing coils on each gyro float force the float to precess horizontally about the (Z) azimuth axis. The computer Z axis torque voltage is applied only to coils on the *lower* (Y) gyro. This results in a Y gyro Z pickoff voltage which is divided to drive two amplifiers—the upper gyro caging amplifier (to torque the upper gyro in azimuth), and the Azimuth gimbal amplifier (to reposition the entire Azimuth gimbal thereby renulling the Z pickoffs on both gyros).

We should point out that the upper gyro caging amplifier also has a second input—the upper gyro Z axis pickoff—that effectively "cages" or renulls



its own pickoff by precessing the upper gyro during aircraft turn maneuvers. An aircraft heading change is sensed by the Z axis pickoffs of both gyros but the upper gyro signal is used to prevent the gyro from bottoming on its own stops while the signal from the lower gyro is used to turn the platform in azimuth at whatever rate is required to maintain the accurate north-south east-west azimuth gimbal position.

The preceding discussion has been presented on a step-by-step basis but it should be realized that any or all of the functions can occur simultaneously. For example, if an aircraft flying northeast started a climbing turn, the combination of forces would result in: both gyros sensing vertical as well as horizontal displacement (turn), both accelerometers sensing acceleration, computer generating X, Y, and Z torqueing voltages and all gimbal motors driving simultaneously in response to the applied gyro torque as well as to the maneuver-generated pickoff signals. This combination of events must occur smoothly and without delay to avoid the introduction of transient errors into the accelerometers.

In effect, the gyros provide the stability and sensing necessary to maintain the stable platform properly positioned in attitude and azimuth and to isolate the stable platform from aircraft maneuvers. In so doing, the accelerometers are properly oriented to sense accelerations within the proper frame of reference—latitude and longitude.

The linear accelerometers provide the basic information used in computation of latitude, longitude, and distance traveled. They are located on the stable platform structure between the two gyros.

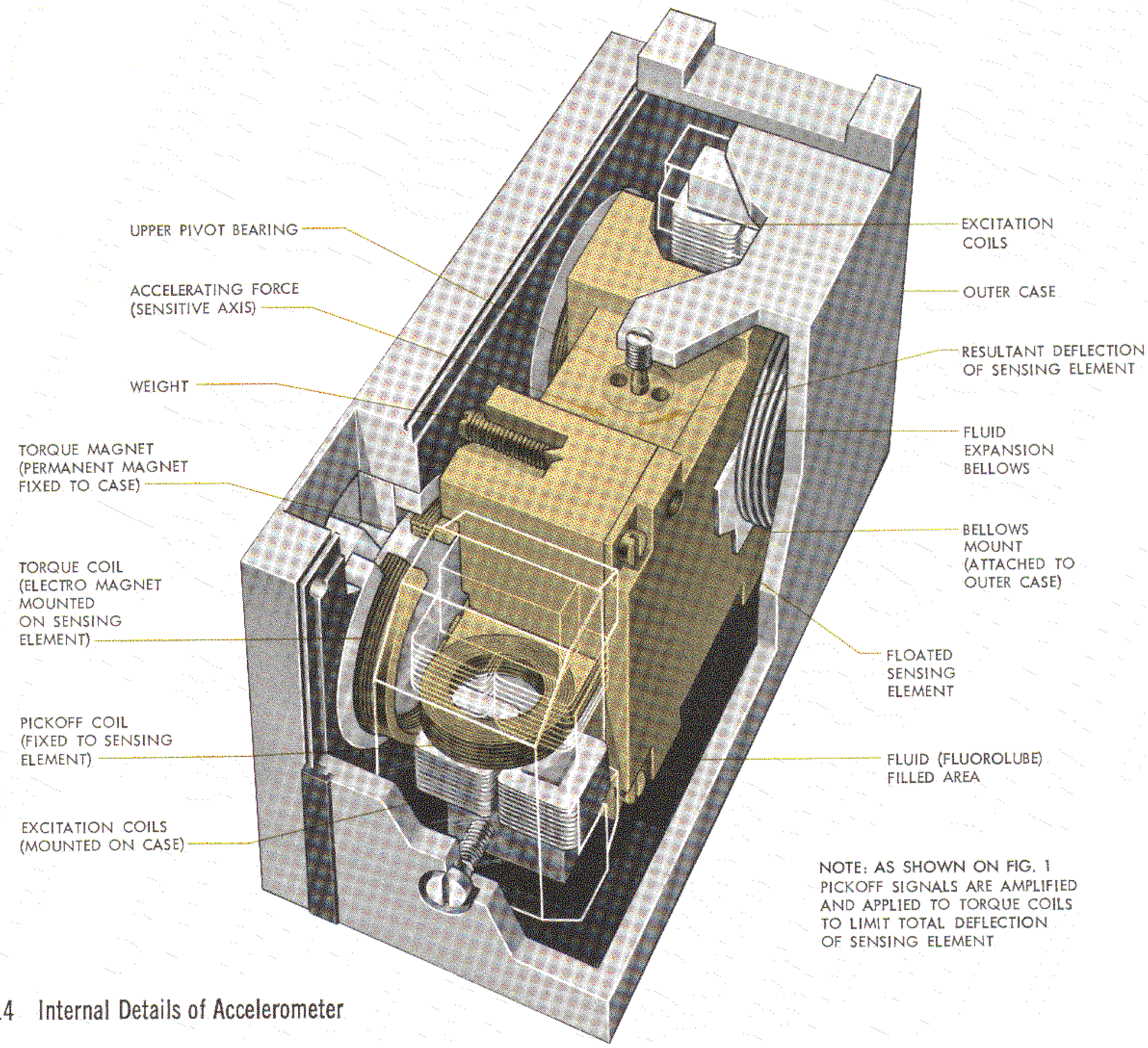


Figure 14 Internal Details of Accelerometer

The accelerometers are identical in construction (see Figure 14) and consist of a floated oblong mass or sensitive element pivoted on two jeweled bearings (top and bottom) affixed to an outer case. The torqueing and pickoff coils are mounted on the sensitive element. The area between the sensitive element and the outer case is filled with the same special fluid (fluorolube) as that used in the gyros and is utilized for the same purpose; to support the weight of the sensing element thereby reducing bearing friction. The entire assembly is maintained at proper operating temperature by heaters attached to the accelerometer case.

The floated sensing element is concentric about its pivotal axis except for a small weight fixed on one end that provides the necessary pendulous action when acted on by a force. The total movement of the sensing element is restricted by mechanical stops to less than half of a degree but, to insure that accelerations are sensed parallel (rather than at an angle) to the coordinate lines during operation, the float is not allowed to move even this small amount. When a force is applied to the sensitive axis (aircraft accelerates or decelerates), the sensing element moves the pickoff coil within the excitation coils and develops a signal which in turn causes the accelerometer-restoring amplifier to produce a dc torqueing voltage. This voltage is applied to the accelerometer torqueing coils, and they react with the case mounted permanent magnets and restrict the sensitive element displacement. The voltage required to limit this displacement is directly proportional to the force being sensed, i.e., the acceleration of the aircraft moving over the rotating earth.

The restoring voltage for each accelerometer is measured through a special scaling resistor (scaled to 2 volts per g acceleration) and applied to the computer as either N-S or E-W acceleration input.

It should be noted that velocity in the E-W direction would result in unleveling *about* the Y axis and platform heading errors *about* the Z axis. Therefore, as indicated in Figure 12, the Computer makes use of the E-W velocity signal to develop the gyro torque required for precession *about* these axes. It is also apparent that N-S velocity would result in an unlevel condition *about* the E-W (X) axis and that the Y velocity signal must be utilized in developing the necessary torqueing voltage.

The process of precise alignment (leveling and gyrocompassing) has been discussed in Part 1, but it is well to point out that during this initial stage the accelerometer outputs are not integrated to obtain velocity. Instead the outputs are amplified by the Computer and applied directly to the gyro torquers to "erect" the platform as quickly as possible.

The Computer and Position Indicator operate together to receive and modify information that originates in the accelerometers.

As described in Part 1 (and shown in Figure 15), the accelerometer signals, modified by Coriolis and centripetal corrections, are the basis for all Computer operations.

In addition to the above major corrections, a damping voltage is added to the accelerometer outputs to reduce the small cyclic errors sensed by the accelerometers as the stable element oscillates about both north-south and east-west axes (simulated Schuler pendulum).

The Computer is divided operationally into two channels (see Figure 15), east-west (X) and north-south (Y). Each channel receives an acceleration signal (plus corrections) at the integrator input summing point. The output of each integrator is *velocity* information represented by two voltages recovered from two motor-driven potentiometers in each channel. A third potentiometer, driven by the same motor and excited by a specially computed signal, develops one of the correction voltages (centripetal) that is applied to the acceleration summing point of the opposite channel. The dc velocity potentiometers supply voltages that are negative (—) for aircraft north and east headings and positive (+) for south and west direction of flight.

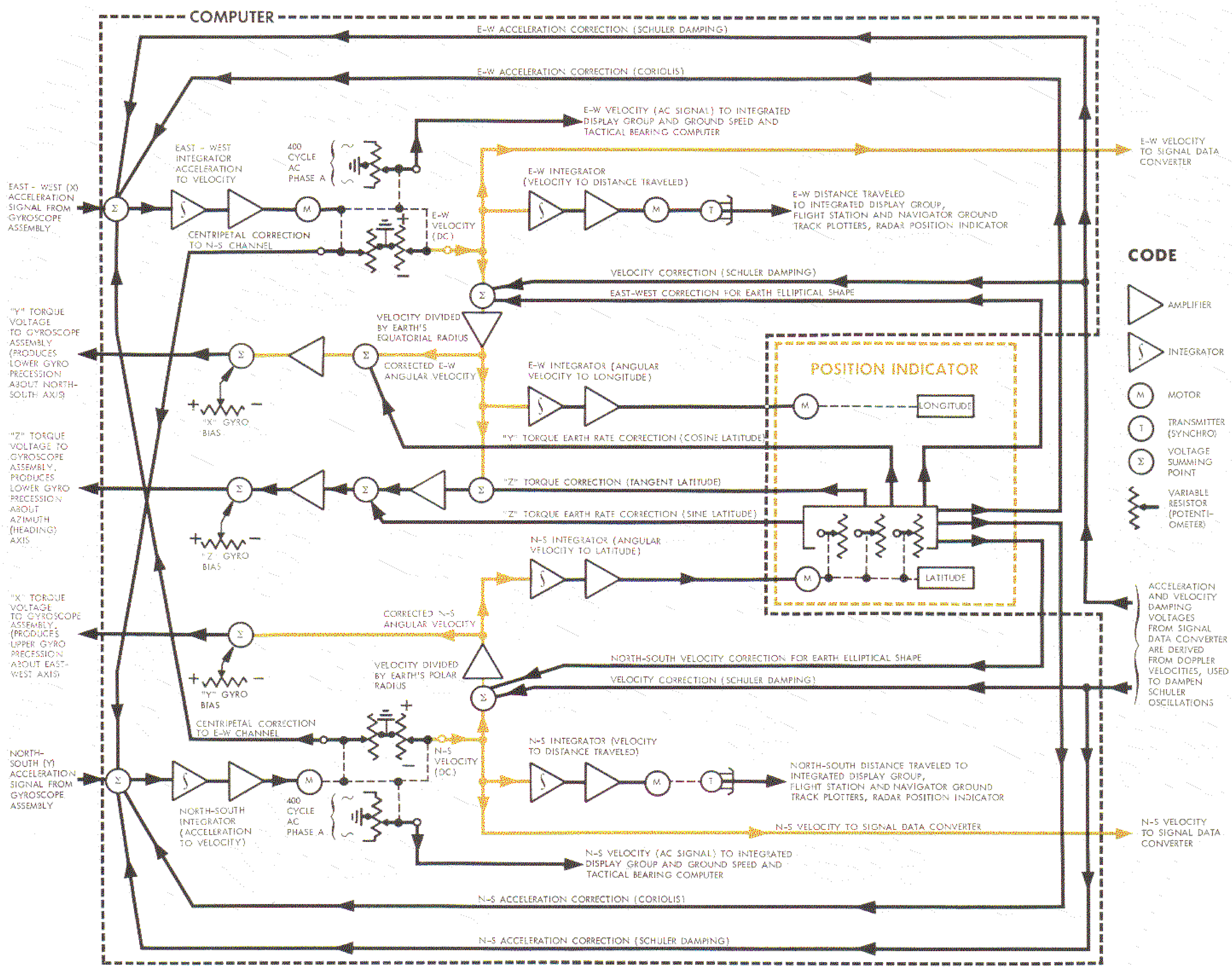


Figure 15 Functional Schematic of Computer and Position Indicator

The ac velocity potentiometers supply a signal that is either in phase or 180° out of phase with the reference signal (aircraft 115-V ac Phase A) to distinguish between one cardinal heading and its reciprocal.

When the aircraft attains a constant velocity, the accelerometer outputs drop to zero. Assuming no correction signals are present at the Computer input summing points, the integrator-driven potentiometer motors will stop quickly. The voltage at the pickoff arms of the potentiometers now represents velocity along that particular flight axis. Normally, however, correction signals will be applied to the input summing points. The motors will continue to reposition the velocity potentiometers as necessary to indicate the actual velocity along each axis.

The dc velocities are the basic signals for all remaining Computer functions. No velocity corrections are added to the north-south and east-west velocity-to-distance-traveled integrators. Each integrator output is amplified to drive a motor that turns a synchro transmitter. The number of revolutions of this transmitter determines the distance traveled output (one revolution of the transmitter is equal to two nautical miles) supplied to the Ground Track plotting units.

Before using the dc velocity signals for computing gyro torqueing and latitude-longitude, two additional corrections are added to the velocity summing points of both channels—those necessary to compensate for the elliptical (flattened at the poles) shape of the earth and those for damping the movement of the gyros to diminish latitude-longitude cyclic errors that result from Schuler oscillations. The earth's elliptical shape is such that the polar radius is approximately 70,000 feet shorter than the radius at the equator. If no allowance is made, this variation results in an error of approximately 11 minutes (11 NM) of latitude indication when the flight path crosses the 45th parallel. Damping voltages will be discussed in more detail in connection with the Signal Data Converter unit where they originate.

The input signal for the latitude integrator is the corrected north-south velocity voltage divided by R_1, a constant value representing the earth's polar radius, thereby changing linear velocity to angular velocity for computing the angle of latitude. The integrator output is amplified to drive the latitude motor in the Position Indicator. The motor positions the latitude counter and moves the arms of specially designed potentiometers (variable resistors). These potentiometers generate the sine, cosine, and tangent functions of the angle of latitude which are subsequently incorporated as Coriolis, centripetal, elliptical, and earth rate corrections in the Computer. The center position of the potentiometers represents 0 degrees (equator) latitude.

The corrected east-west velocity is divided by a constant R_2 voltage representing the earth's equatorial radius to produce the angular east-west velocity, and this value is integrated and amplified by separate circuits to position the longitude motor and indicator. No correction potentiometers are required in this channel.

The corrected angular velocity voltages also become a part of X, Y, and Z torque signals for the two gyros. Earth rate corrections are added to aircraft velocities at Y and Z torque summing points to obtain the *total* leveling and the *total* azimuth torque drive required to fix the stable platform to earth rather than space orientation. Since no earth rate movement takes place *about* the east-west axis (the north-south plane), earth rate correction is not applied to the X torque summing point.

Despite all precautions taken in design and construction of the gyros, mechanical unbalance and bearing friction cause them to drift from their starting position. The constant or predictable portion of this drift is measured during initial adjustment of the system and "biased" out by application of a dc voltage at the torque summing points (see Figure 15) to exactly cancel the drift. The gyro bias potentiometers* are accessible on the computer front panel by removal of a small cover plate. However, no attempt should be made to correct their settings except as noted in Maintenance Instruction Manual, NAVWEPS 01-75PAA-2-7.3.

Signal Data Converter, Converter-Decoupler, and Align Control Two major and several minor functions take place within the Signal Data Converter (SDC).

As previously stated, one of the SDC functions is to detect differences between reported inertial vel-

**Some confusion exists concerning the labels on X and Y GYRO BIAS potentiometers on the Computer front panel. Y GYRO BIAS is the designation on the control which corrects for drift indicated by an output from the Y (North-South) accelerometer. However, this output indicates a vertical movement (drift) of the upper X gyro which causes the Y accelerometer to be unlevel and to sense a gravity force. The Y GYRO BIAS might be more realistically labeled "Y AXIS BIAS."*

The same reasoning is applicable to the X GYRO BIAS designation. This is bias applied to the lower "Y" gyro to counteract drift (vertical movement) of the system X axis.

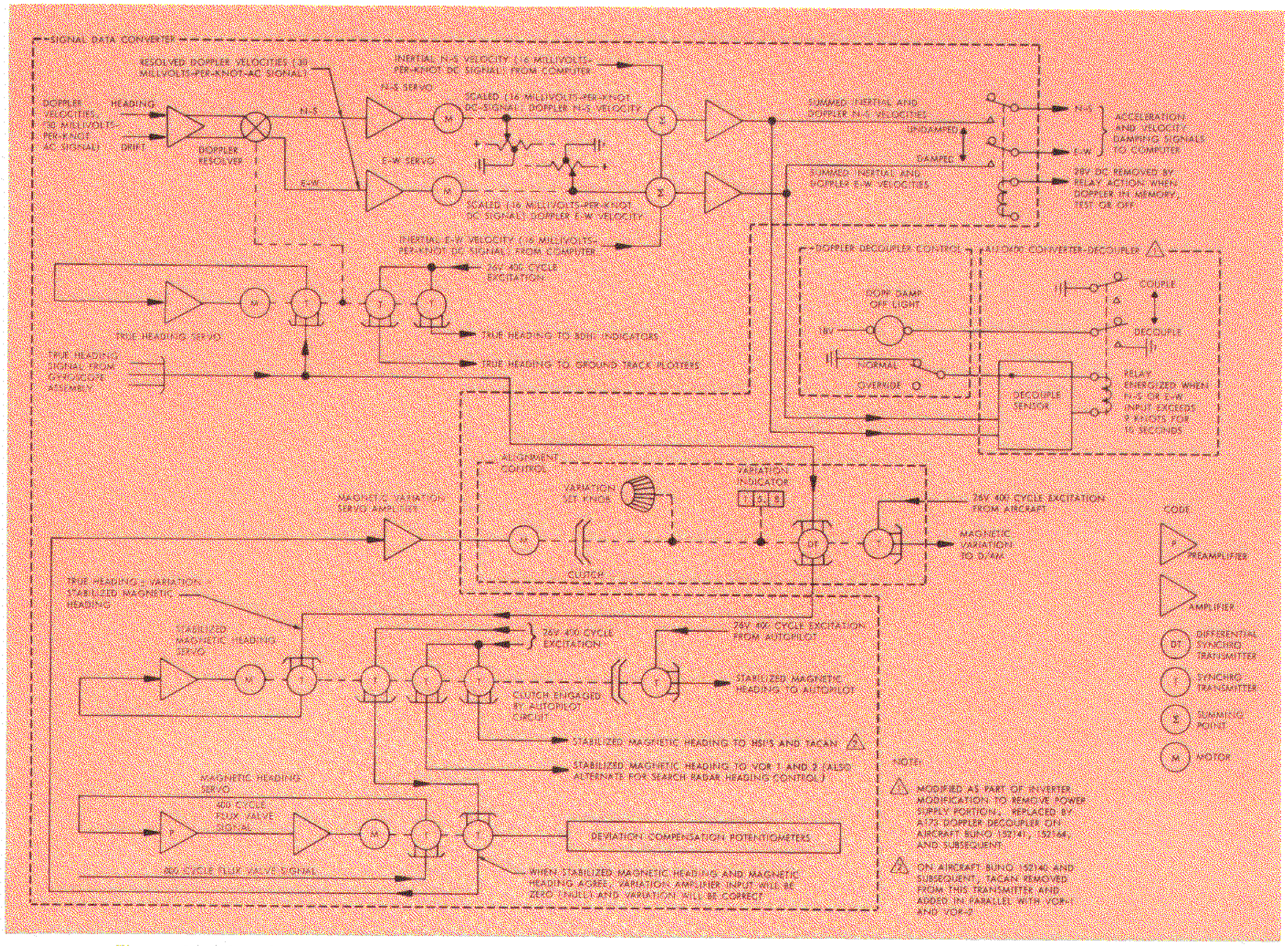


Figure 16 Circuit Depicting Major Functions of Signal Data Converter, Converter-Decoupler, and Alignment Control

ocities and Doppler velocities to be utilized by the Computer in creating Schuler damping signals. Since the Doppler outputs are ac voltages (30 millivolts per knot) separated to represent speed forward and drift (speed sideways) without regard to global coordinates, they cannot be compared directly with the inertial systems dc voltages (approx. 16 millivolts per knot) representing velocities north-south and east-west. The ac Doppler signals are first converted into N-S and E-W ac voltages by the Doppler velocity resolver which is driven by the true heading shaft. The new ac signals are then changed to representative dc values by the servo driven potentiometers, and finally scaled to 16 millivolts-per-knot by a summing amplifier (not shown on Figure 16).

The Doppler N-S and E-W dc velocities are then compared (summed) with N-S and E-W inertial dc velocities supplied by the Computer. If the corresponding velocities are equal, they will appear at the summing points with an opposite polarity and cancel out. When they differ, the difference value will be supplied to the computer as a damping signal for that particular channel. In the event the Doppler signals become unreliable or when the Doppler system is in TEST, MEMORY, or OFF, damping signals are automatically disconnected from the computer by relays in the Signal Data Converter.

The Computer damping voltages are monitored by the Doppler decoupler portion of the AU 0400 Converter-Decoupler when the system is operating in I-D mode. If the value of N-S, E-W, or both inputs exceeds approximately 150 millivolts* (equivalent to 9 knots) for a period of 10 seconds, all damping is removed by the decouple signal and the DOPP DAMP OFF indicator is illuminated. Damping is restored to the Computer when the level of both monitored signals is reduced to, and remains at, 150 millivolts or less, for 10 seconds. Suggestions for use of the Doppler damping NORMAL-OVERRIDE switch are presented in the General Operating Information section of this article.

This value is changed to 225 millivolts (equivalent to 13.5 knots) on aircraft equipped with the A 173 Doppler Decoupler in place of the AU 0400 Converter-Decoupler.

The second major function of the Signal Data Converter is computation and distribution of heading information—both true and magnetic. One true heading output from the Gyroscope Assembly is shared within the system by two loads: the true heading servo input synchro, and the Alignment Control variation differential synchro. The true heading input synchro provides the input signal for the true heading repeater servo system* amplifier which in turn drives a motor, the input synchro, two true heading repeater transmitters, and the Doppler velocity resolver on a common shaft. Since the signal distribution from the two repeater transmitters was covered earlier in this article, they will not be discussed here. Note that the Doppler velocity resolver is the device mentioned previously that converts Doppler heading and drift velocity inputs into N-S and E-W velocity terms.

True heading is altered — magnetic variation added or subtracted — by the variation synchro shaft position and the modified signal is indicated on Figure 16 as stabilized magnetic heading. Since this signal is not adequate to supply aircraft magnetic heading requirements, the signal is utilized to position the Stabilized Magnetic servo which drives three repeaters.

Stabilized magnetic heading distribution to aircraft avionics and instruments was discussed previously, but some additional description is due the autopilot heading transmitter which is a special type not permanently coupled to the driving shaft. The shaft incorporates a clutch that is engaged and disengaged by control signals from the autopilot. When the autopilot is engaged, the clutch is also engaged and the synchro output is routed to the autopilot aileron channel to hold the aircraft on a selected heading. During an autopilot-controlled turn, the clutch is disengaged, the spring loaded transmitter returns to a zero signal position, and is reclutched to the shaft after the aircraft is stabilized at a new heading.

The flux valve 800-cps magnetic heading signal is converted to a 400-cps signal by a preamplifier and used to position a magnetic heading servo. Magnetic heading data is taken from a transmitter attached to the shaft, and is corrected by a deviation network to cancel out the effects of aircraft structure on the earth's magnetic lines of force (compass rose compensation).

Variation is automatically computed by comparing true heading, taken from the inertial platform, with the magnetic heading sensed by the flux valve. The two phases of computation are virtually simultaneous but are best explained separately. First, as discussed previously, stabilized magnetic heading is computed from true heading and magnetic variation which is directly dependent on the second phase for its accuracy. The second phase involves the two interconnected synchros shown on Figure 16 which compare computed magnetic heading and flux-valve magnetic heading. When the interconnected synchros detect a divergence between these signals, a corrective signal appears at the variation servo input. As the variation servo drives, it in turn brings stabilized magnetic heading into agreement with magnetic heading to complete the sequence. Since the variation shaft turns very slowly due to a gear reduction, it will not reflect the rather erratic output of the flux valve and the stabilized magnetic heading repeaters will present stable information to aircraft instruments.

With the exception of the servo amplifier, the entire variation servo system is contained in the Alignment Control. The variation shaft clutch indicated in Figure 16 is engaged in I-D mode. When engaged, the counter presents computed variation to the nearest tenth of a degree and also drives the output transmitter connected to the variation servo in the Doppler/Air-Mass Computer. The clutch is disengaged in all PILOT modes to permit manual insertion of variation.

The True Heading servo is representative of a basic servo system and consists, in this case, of an input synchro (control transformer), an amplifier, and a motor. When signals are applied to the input synchro, its output causes the amplifier to drive the motor. As the motor turns it repositions the input synchro rotor until its output decreases to a minimum. The amplifier input and output is reduced to zero and the motor stops. The new position of the motor shaft (and that of transmitters geared to it) represents an angular mechanical change equal to the electrical signal input.

ADJUSTMENT AND ALIGNMENT

The inertial system is designed to minimize necessary adjustments after aircraft installation is made. With two exceptions, *deviation compensation* during compass rose operations and final *gyro biasing,* all adjustments are normally made in the shop. The technique of "swinging" the inertial system on a compass rose is very similar, if not identical, to that commonly used for attitude heading reference systems and might well be the topic of an entire article. We mention it here only to point out that for maximum magnetic heading accuracy the inertial system Signal Data Converter deviation potentiometers should be recalibrated, using an MC-2 Calibrator or a compass rose, when *the Signal Data Converter is replaced* or the *flux valve is either replaced or removed and re-installed.*

Gyro biasing is, however, the most critical adjustment required on an installed system. Improper system biasing is the cause of more out-of-tolerance notations on flight "yellow sheets" than any other inertial system discrepancy.

Past experience proves that gyro-biasing problems are the cause of much unnecessary work and many costly delays. The records of a group of aircraft which experienced inertial system discrepancies on 38 flights show that:

1) In eight instances the discrepancies were rectified by gyro bias adjustment without removing components from the aircraft.

2) Twelve systems were removed to the shop, where it was proven that ten of these were removed unnecessarily, for they required only gyro-bias adjustment to one or more controls.

Thus, the gyro-bias adjustment was responsible for almost exactly 50% of the troubles experienced by these aircraft, a situation that warrants a heavy emphasis on this aspect of maintenance.

It is difficult to set in proper gyro bias after installation without the use of a Gyro Bias Test Set (Gyroscope Test Set TS-1987/ASN-42). The theodolite-north-line procedure works well under shop conditions but Gyroscope Assembly location in the aircraft makes it impractical after installation.

Although very careful gyro-biasing in the shop will often provide a system that is sufficiently accurate, field experience shows that it is advisable to make a simple check of the gyro-bias adjustment after installation to obviate flight discrepancies due to this cause. Carrying out at least 1 hour of the 2-hour test procedure set forth below will generally reveal the fact if a bias adjustment *is* necessary but, of course, a test flight must be made to determine whether the system is acceptable in all respects. Note that this is a test of accuracy only. It does not supply sufficient data to guide the biasing adjustment, and any attempt to make the bias adjustment without the gyro-bias test set will probably worsen the situation instead of improving it.

1. Connect angle indicator (AN/USM-165) to Signal Data Converter test plug 3J8; S1 lead to pin 3J8-K, S2 lead to 3J8-q, and S3 lead to 3J8-J.

2. Energize the system in I-D mode, set in local latitude and longitude on the Position Indicator, record the setting and align the system.

3. When alignment is complete, record true heading indicated by the USM-165 indicator.

   NOTE

   The actual reading of the true heading indicator will be determined by the aircraft heading and is not important to this test since only the drift from the initial reading is desired.

4. Switch to OPERATE and record time, latitude, longitude, and true heading at 10-minute intervals for 120 minutes or longer if time permits.

   NOTE

   The data recorded will be used to determine whether the equipment is performing satisfactorily during the ground drift test. The maximum allowable errors are:

1. Latitude—2 minutes of latitude drift per hour.

2. Longitude — determined by computation (2 divided by the Cosine local latitude). Example: Cosine 40° latitude is .766.
   2 divided by .766 = 2.6 minutes of longitude drift per hour allowed at 40° latitude.

3. True heading drift not to exceed value computed for longitude.

If the above maximums are exceeded during the drift test, the test should be discontinued and the system rebiased with the Bias Test Set.

Of course, proper gyro biasing as well as overall operational readiness of the entire inertial system can best be determined in flight over a pre-planned course where heading, latitude, longitude, and distance between check points (VOR station, TACAN station or visual landmark) are known.

Checks of this nature, whether incorporated as part of local training flights or in connection with other mandatory aircraft check flights outlined by BuWeps INST. 4700.2, provide a practical operational check prior to actual missions. Such flights should be conducted as indicated in the "intertial system accuracy tests" included as part of the Periodic Maintenance Requirement Manual, NAVWEPS 01-75PAA-6. For the information of readers who do not have access to this publication, the pre-planned course over which the flight tests are to be conducted should meet the following minimum requirements:

a. Include at least eight check points. The same check points may be used more than once but at least two different points are needed.

b. Provide approximately 15 minutes flying time between check points. Flight speed and/or course length should permit at least 2 hours for tests, longer if possible.

c. Course should contain components of all cardinal directions. Ideal course would be box or L shaped, oriented to true cardinal headings with each leg as long in time as possible.

d. If VOR or TACAN stations are to be used for "on top" position checks, flight altitude should not be more than 15,000 feet.

Figure 17 is a replica of a typical graph (included in NAVWEPS 01-75PAA-6) on which latitude and longitude check point differences (errors) are plotted. The maximum limits—above and below the zero reference—are based upon a tolerance of 4 nautical-miles-per-hour (4NM/HR) for the first forty minutes of operating time and 3NM/HR thereafter.

The latitude limit (black) lines on the graph apply for tests performed at *any* latitude, because 1 minute of latitude is always equal to 1 NM. Note, however, that longitude limits (colored lines) are greater —1 minute of longitude is equal to 1 NM only at the equator—and must be computed for the particular test area. Variations from the known values at check points are then read in minutes of latitude and longitude error on the Position Indicator and plotted directly as nautical miles on the graph.

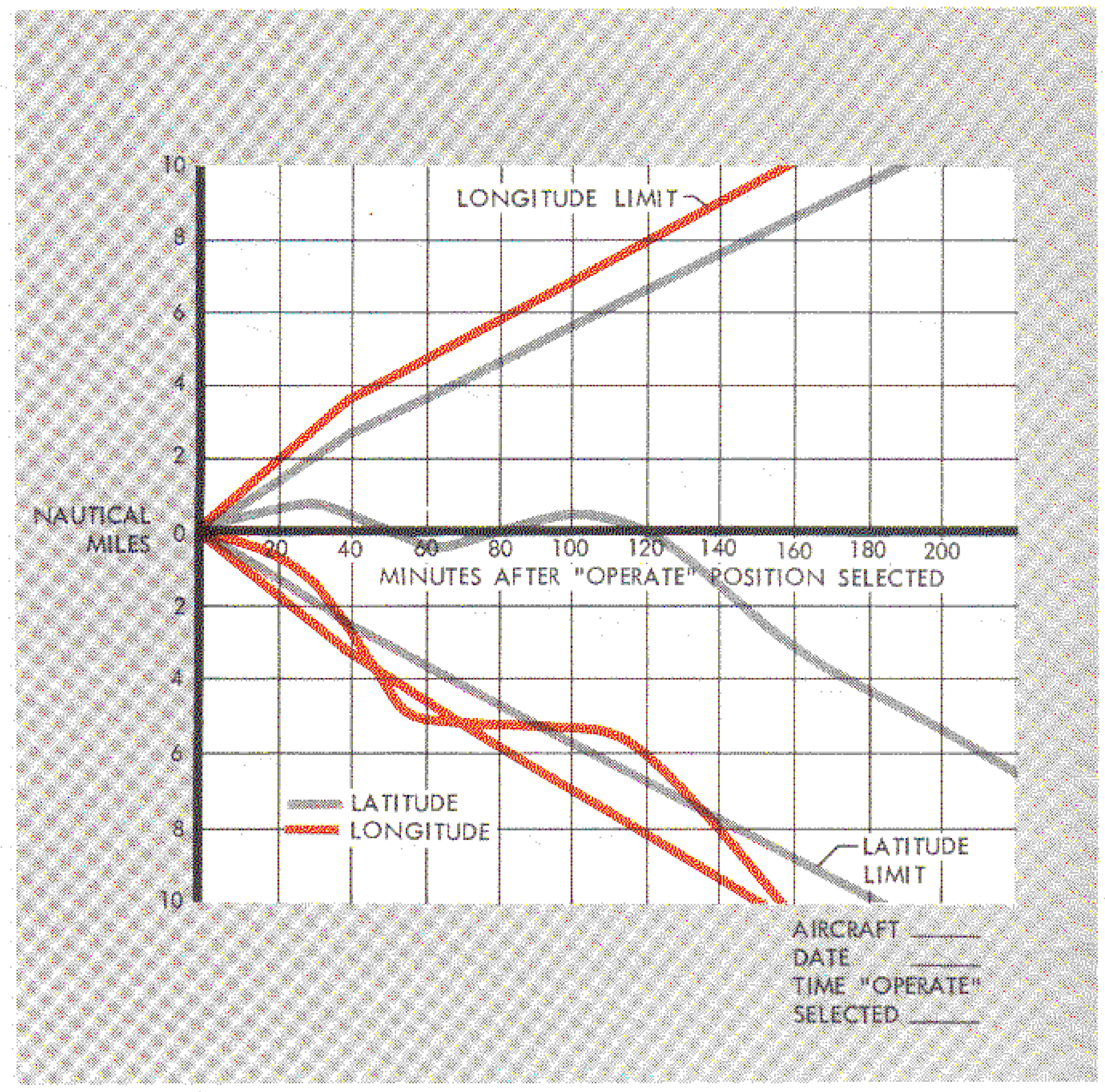


Figure 17 Graph showing a Longitude Out-of-tolerance Error Traced to Incorrect X Gyro Bias Setting. Incorrect Z gyro bias was not apparent in latitude indication before 2 hours.

Specific details for graph layout are given in connection with the typical graph depicted in NAVWEPS 01-75PAA-6, but briefly, longitude limits are computed by these two formulas:

1. First 40 minutes, limit lines diverge from true at the rate of 4 NM/HR divided by Cosine local latitude.

2. Beyond 40 minutes, limit lines diverge from true at the rate of 3 NM/HR divided by Cosine local latitude.

Figure 17 depicts the graph used at Lockheed's Burbank facilities (34° 11.8" latitude, 118° 21.2" longitude) and is a plot of the latitude and longitude errors that were traced to incorrect "X" and "Z" gyro bias settings.

Experience gained from production flying of the inertial system indicates reliable repeatability from flight to flight. For example, if a system has a 2 NM/hour latitude drift rate on its first flight, the direction and the rate of drift will normally be the same on subsequent flights. Although this drift rate is within the specified tolerance, it is apparent that the exact performance data is essential to the Navgator if he is to derive maximum benefit from the Inertial System capabilities. Flights conducted over a known course as just described will not only provide this information but will indicate the functional accuracy of the other navigation systems as well.

This insight is vital to the Navigator and/or the Tactical Coordinator when making a decision concerning the proper operating mode for use in a tactical situation.

The data plotted on the graph can also be useful in trouble analysis. For example, gross errors are seldom due to misadjusted gyro bias, and an error in excess of 5 NM/HR in either latitude or longitude for the first hour in OPERATE position usually indicates a fault other than incorrect bias adjustment.

The Figure 17 graph is typical of gyro bias trouble, and the specific nature of the trouble may be analyzed as follows:

Longitude lapsed out-of-tolerance between the 30 minute and one hour marks. Longitude is directly related to the X axis, and re-adjustment of the X GYRO BIAS control is indicated.

Latitude appeared to be quite accurate for 2 hours, thus exonerating the Y Gyro-Bias setting, but thereafter a precipitous error appeared. Such a pattern is generally due to a gradually developed azimuth error, and a re-adjustment of the Z GYRO BIAS is indicated.

GENERAL OPERATING INFORMATION

Incomplete ground alignment can also be the direct cause of system errors. Although the system is basically automatic in operation during this phase, one area depends directly upon the patience and judgement of operating personnel. A premature selection of "Operate" position can start the mission with an error that will become progressively worse as the flight continues. To avoid this, care should be taken to allow the "ready meter" needle to come to rest in the center green area and remain there for a minimum of 30 seconds. "Ready meter" needle action is highly damped and it cannot be deemed to positively indicate that gyrocompassing is well stabilized unless it remains in the green meter area for the specified time.

Figure 18 indicates the pattern of latitude and longitude errors which generally result from inaccurate "gyrocompassing" of the system. As shown, when the errors are plotted as specified in NAVWEPS 01-75PAA-6, they increase in opposite directions on the graph. Of course, the latitude and longitude plots will be interchanged in some instances.

Ground power units improperly maintained or of the wrong type often cause difficulty in ground alignment. The extremely small signals involved, particularly during the "gyrocompassing" phase are indistinguishable to the high gain amplifiers from noise or voltage variations in the power supply. Persistent alignment difficulty, even with a system known to be in good operating condition, is reason to suspect the ground power unit. The NC-12 and NC-12A provisioned for the P-3 are more than adequate when properly maintained. In most cases substitution of other lower-powered, wider-tolerance ground power units will result in extended time of alignment, inaccurate alignment, or both, and their use should be avoided.

The amber AU 0400 NO-GO light located on the Converter-Decoupler front panel should be inspected periodically during flight* and if the light is illuminated measures should be taken to re-activate this auxiliary power supply. The AU 0400 provides exceptionally stable power to critical Computer circuits. When it is unable to function properly, possibly due to a single abnormally long delay in a bus-transfer operation, the NO-GO light illuminates and the circuits normally carried by the AU 0400 are diverted to the Inertial System Power Supply. Over an extended period of operation the small transients which normally occur in the bus power will be re-

**Modification of the AU 0400 Converter-Decoupler (or replacement with the A 173 DOPPLER DECOUPLER unit) as part of the inverter installation will eliminate this troublesome chore.*

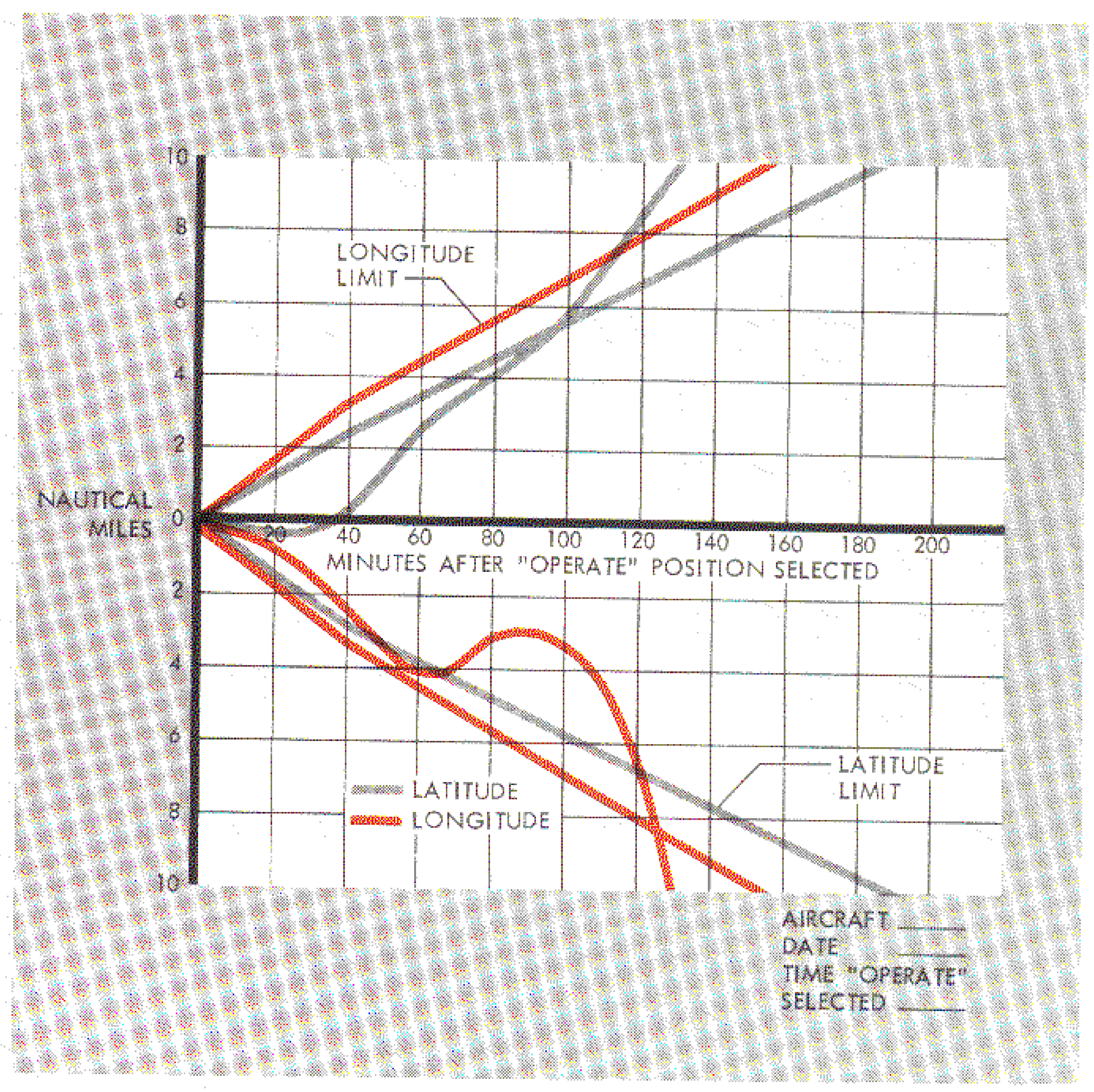


Figure 18 Graph Depicts Typical Latitude-Longitude Errors due to Incomplete Gyrocompassing

flected in accumulated errors, especially in the latitude-longitude indications. If the light is checked periodically, and a special check is made when there is reason to believe bus power has been interrupted briefly, the flight crew can protect the fidelity of their inertial navigator, avoid writing unnecessary flight discrepancies, and facilitate maintenance work if that is, in fact, necessary.

The Doppler damping OVERRIDE switch on the navigator's panel incorporates a guard which maintains the switch in NORMAL position. When the guard is lifted, the OVERRIDE position can be selected. The question is: When and why should this switch be placed in OVERRIDE position? It is a practical impossibility to cover every situation that might occur, but the following discussion, based on a careful study of the circuitry involved, outlines some of the considerations that will enter into the Navigator's decision.

In the I-D navigate mode, velocity signals from the Doppler system are continually used to "damp" the false velocity signals induced by the Schuler oscillations provided that Doppler and inertial velocities do not differ by more than 9 knots (13.5 knots when the new A 173 Doppler Decoupler is installed). However, the Decoupler may disconnect these corrections at a time when they are needed most; during the Schuler period when the inertial velocities are approaching their maximum cyclic error point.

Activating the OVERRIDE switch at this time will reduce the amplitude of this cyclic error, but *caution* must be exercised as the decouple function is bypassed and damping is continuous. In other words, the decouple function may have occurred as a result of some malfunction of the inertial system Doppler input circuitry, or of the Doppler system itself, and may not be the result of a large cyclic error. Under these conditions the Navigator should monitor inertial and Doppler ground speeds to determine which appears to be most nearly correct, and thereby be certain he is damping rather than inducing large errors into the Computer.

To determine which system, Doppler or inertial, is in error can prove somewhat difficult, especially while flying over water in an area where no visible landmarks, LORAN, or ground based radio navigation stations can be relied upon for a "fix". In this case, a comparative check of ground speed can be made on the TD-441/A Intervalometer Ground Speed Indicator by switching from inertial to Doppler input to this unit. When damping is removed (DOPP DAMP OFF lamp illuminated), and ground speed differential is small (within the predetermined limit monitored by the decoupler), the Doppler Decoupler unit or the Doppler input circuitry to the Signal Data Converter is generally at fault. In this case the OVERRIDE feature should *not* be used as the defective circuits can introduce large errors. However, should the ground speed differential prove to be somewhat larger (14 to 18 knots), and the Doppler ground speed indication compares favorably with that plotted from true air speed and wind, the inertial system is probably in error—larger than normal Schuler oscillation—and its accuracy will be increased by placing the Doppler Decoupler switch to OVERRIDE.

In short, a cautious check should be made before overriding the decoupler. If the Navigator overrides a decoupler that was triggered by any discrepancy other than an abnormally large Schuler oscillation, there is every chance that the inertial system performance will be seriously degraded.

MAINTENANCE

Maintenance time for the inertial system can be considerably reduced if flight crews recognize and report system discrepancies and/or unusual electrical power conditions that occur prior to and at the time of failure. In many cases the report of such discrepancies, especially the chronological order of events, will aid maintenance personnel in locating the trouble. In a relatively large number of cases, a seemingly unrelated item such as a power transfer relay may be at fault rather than the system. Removal of a part or all of a system to the shop is time consuming and may very well introduce troubles where none existed.

Were the "out of tolerance" errors generated early or late in flight? Were the "discrepancy" indications gradually built up or did they occur rapidly after three or more hours of flight? The answers to these and other questions, regularly tabulated, indicate such faults as improper bias, small offsets in dc amplifiers, or complete failure of some component in a specific area.

For example, an incorrect value of earth rate voltage from the Position Indicator will cause a rapid rotation of the longitude indicator before it becomes noticeable as an obviously incorrect true heading output. Both areas will be affected, but the time difference indicates a common error source, the earth rate potentiometer in the Position Indicator.

Another instance is that of a flight discrepancy report of rapid change in longitude while flying east and normal change while flying west—indicating a partially defective A-24 summing amplifier in the Computer.

Numerous other examples could be cited, but the above two serve to illustrate the value of noting the sequence and specific nature of the discrepancy to aid maintenance personnel in diagnosis.

Identifying the most likely cause of a reported symptom is a technique acquired from experience. A brief record of repair histories should be maintained in the form of a log which states the original discrepancy and the steps taken to correct it. A file of this type of information is useful even to experienced personnel, and will be an invaluable training aid for new personnel coming into the shop.

The information presented below was compiled from records maintained by Lockheed Flight Operations and Lockheed Quality Control, and may be helpful as a guide for establishing similar tabulations at AMD facilities.

ASN-42 SYSTEM DISCREPANCY	EFFECTIVE REPAIR ACTION
LONGITUDE COUNTERS WILL NOT SLEW.	REPLACE 4A8 POWER AMP. IN COMPUTER.
LONGITUDE COUNTERS NOT DRIVING AND PLOTTERS NOT MOVING E-W.	REPLACE 4A7 INTEGRATOR SHAFT ASSEMBLY.
LATITUDE COUNTERS OUT-OF-TOLERANCE AFTER 55 MIN. FLIGHT	RESET "Y" GYRO BIAS.
INERTIAL SYSTEM WILL NOT ENERGIZE IN CAGE.	REPLACE AM-3113 ELEC. CONTROL AMPLIFIER.
LATITUDE OUT-OF-TOLERANCE AFTER 3 HR. FLIGHT.	RESET Z GYRO BIAS.
GROUND TRACK PLOTTERS RUN EAST-WEST WITH NO VELOCITY INPUT.	REPLACE 4A9R119.

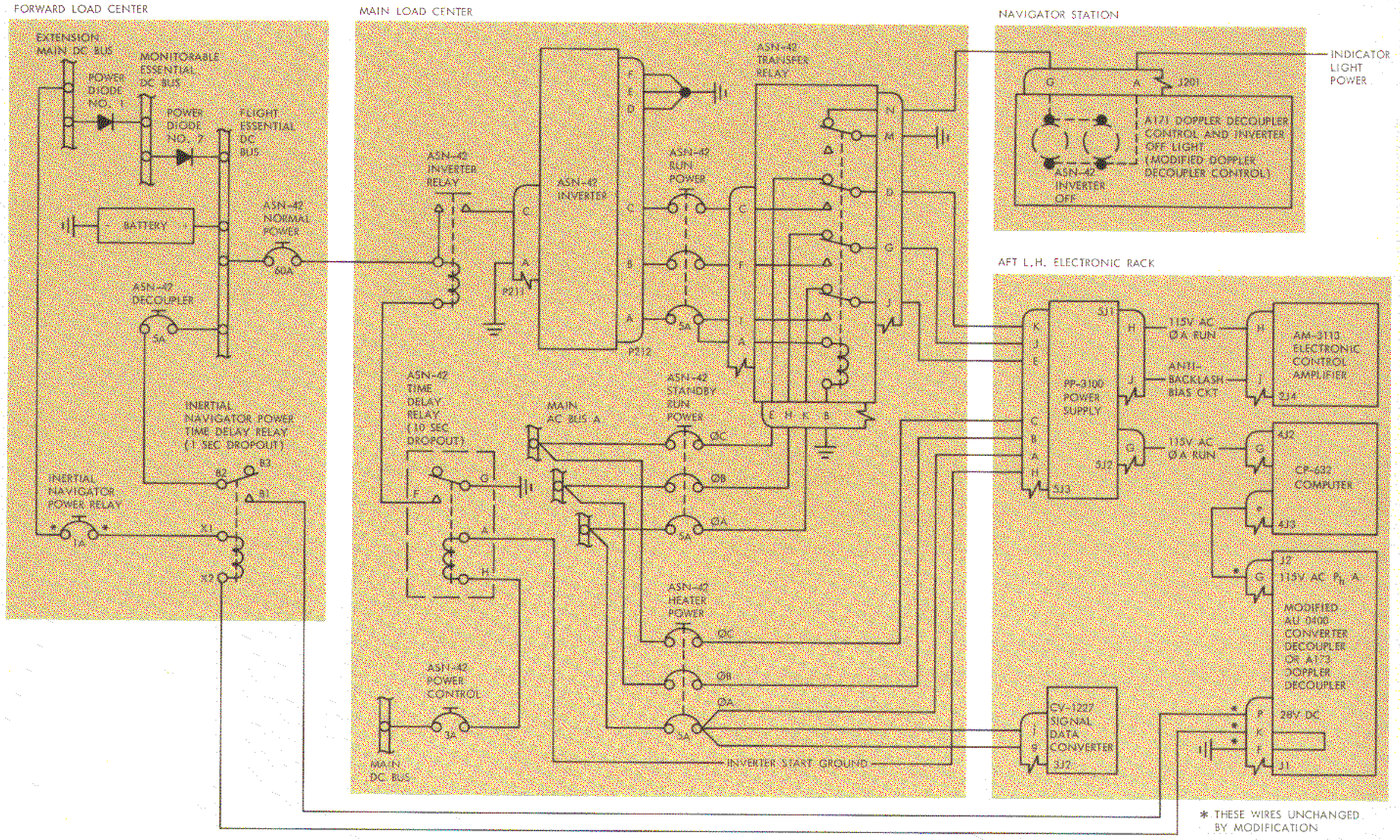


Figure 19 Inertial System Inverter Power Circuitry

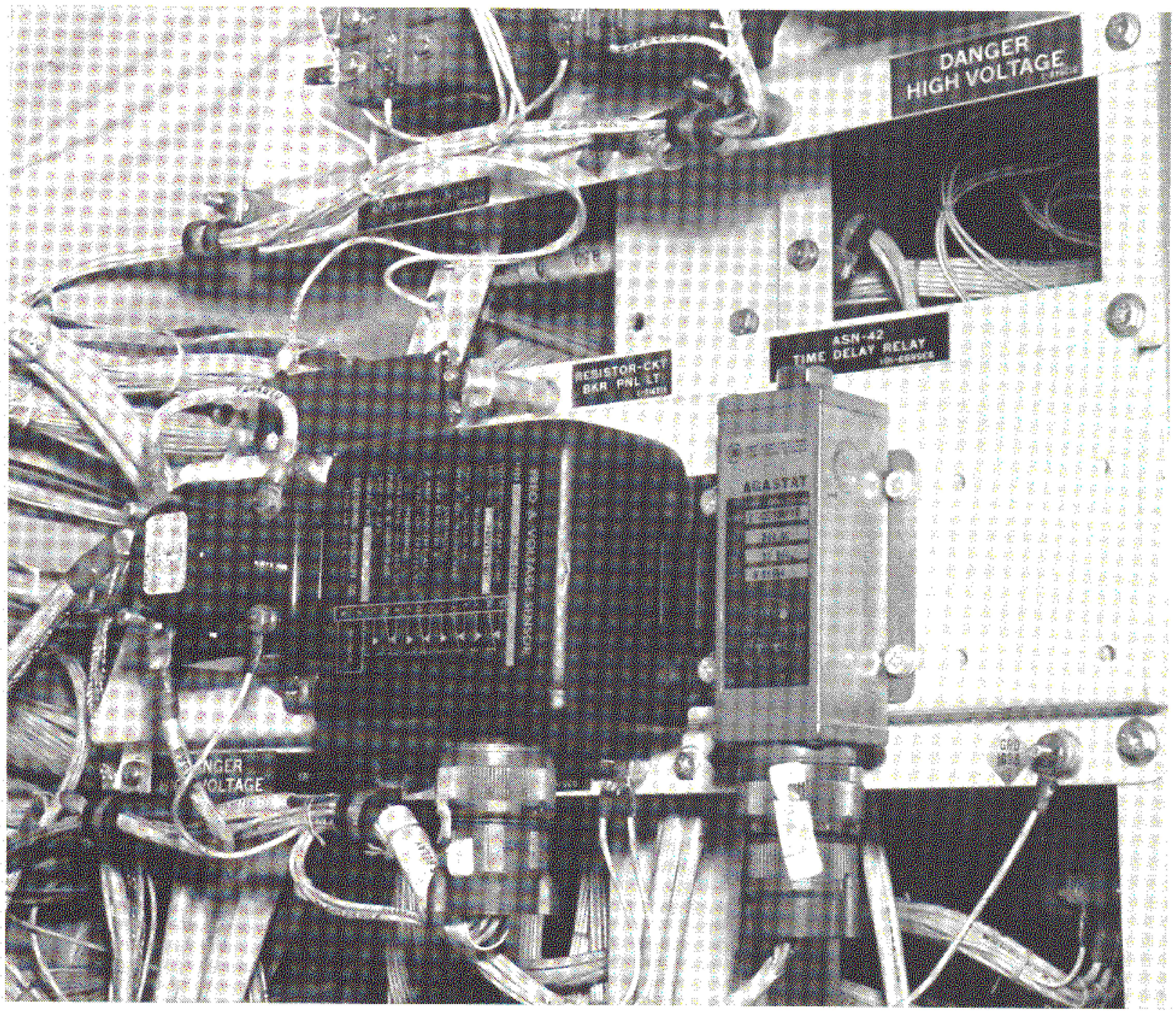


Figure 20 Inertial Navigator Inverter Control Relays — Main Load Center

SYSTEM MODIFICATION AND RELATED CHANGES

Past experience has proven that the implements that introduce a new concept in any field must undergo modifications and improvements to produce the desired results. The P-3 inertial navigation system is no exception.

Included as part of a current up-dating and modification program is the installation of a rotating type inverter to supply the inertial system 115-V ac RUN bus. The results of concurrent flight testing of the inverter (and related changes) by the Navy and Lockheed Flight Test give reason to believe that two previously discussed trouble areas are virtually eliminated by this change. Specifically, poor regulation of ground power units will no longer seriously impair or delay inertial system ground alignment, nor will aircraft ac and dc power transfers and transients introduce velocity errors into the Computer during ground or flight operations.

The simplified electrical circuit depicted by Figure 19, indicates that 28-V dc power to run the inverter is to be supplied directly from the Flight Essential DC Bus by way of a circuit breaker (ASN-42 NORM. PWR, shown also on Figure 21). The aircraft battery effectively filters any transients that might otherwise appear on this bus.

The inverter control relays on Figure 19 are shown in Figure 20 as they appear installed on a panel above the top forward shelf of the main electrical load center. The shock mounted inverter is secured to the floor at the forward end of the Main Load Center proper.

Figure 23 directs attention to the new location of the inertial system circuit breakers at the Main Load Center panel (relocated from Aft L.H. Electronic panel).

CAGE position of the Inertial Navigator Control energizes the inverter power relay (supplies a ground) via a PP3100 Power Supply relay and a time delay relay. The time delay relay energizes when the Power Supply "start" ground is applied but will not de-energize for 10 seconds after 28 V Main DC Bus power is lost or removed.

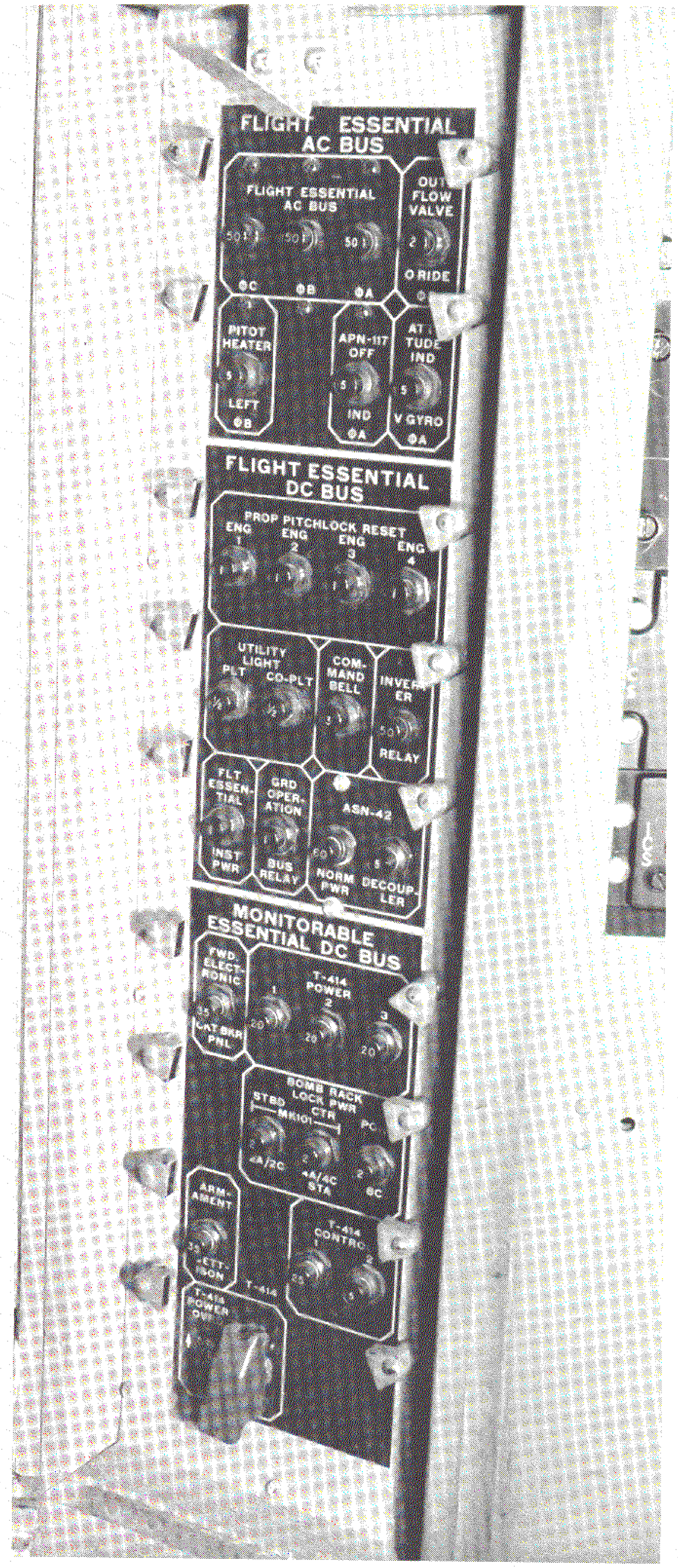


Figure 21 Inverter and Doppler Decoupler 28V DC Circuit Breakers at FLIGHT ESSENTIAL DC BUS — Forward Load Center

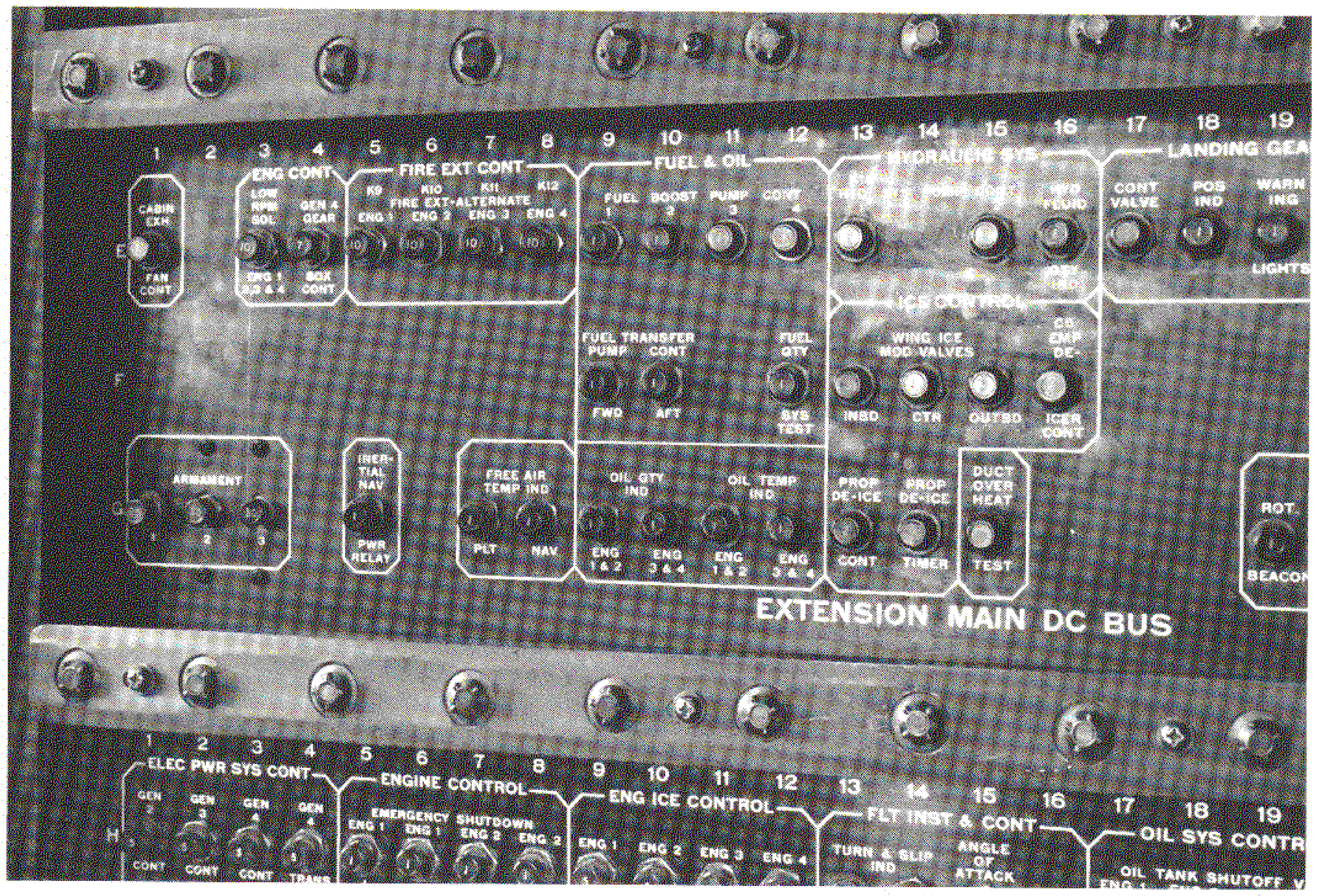


Figure 22
EXTENSION MAIN
DC BUS — Forward Load Center

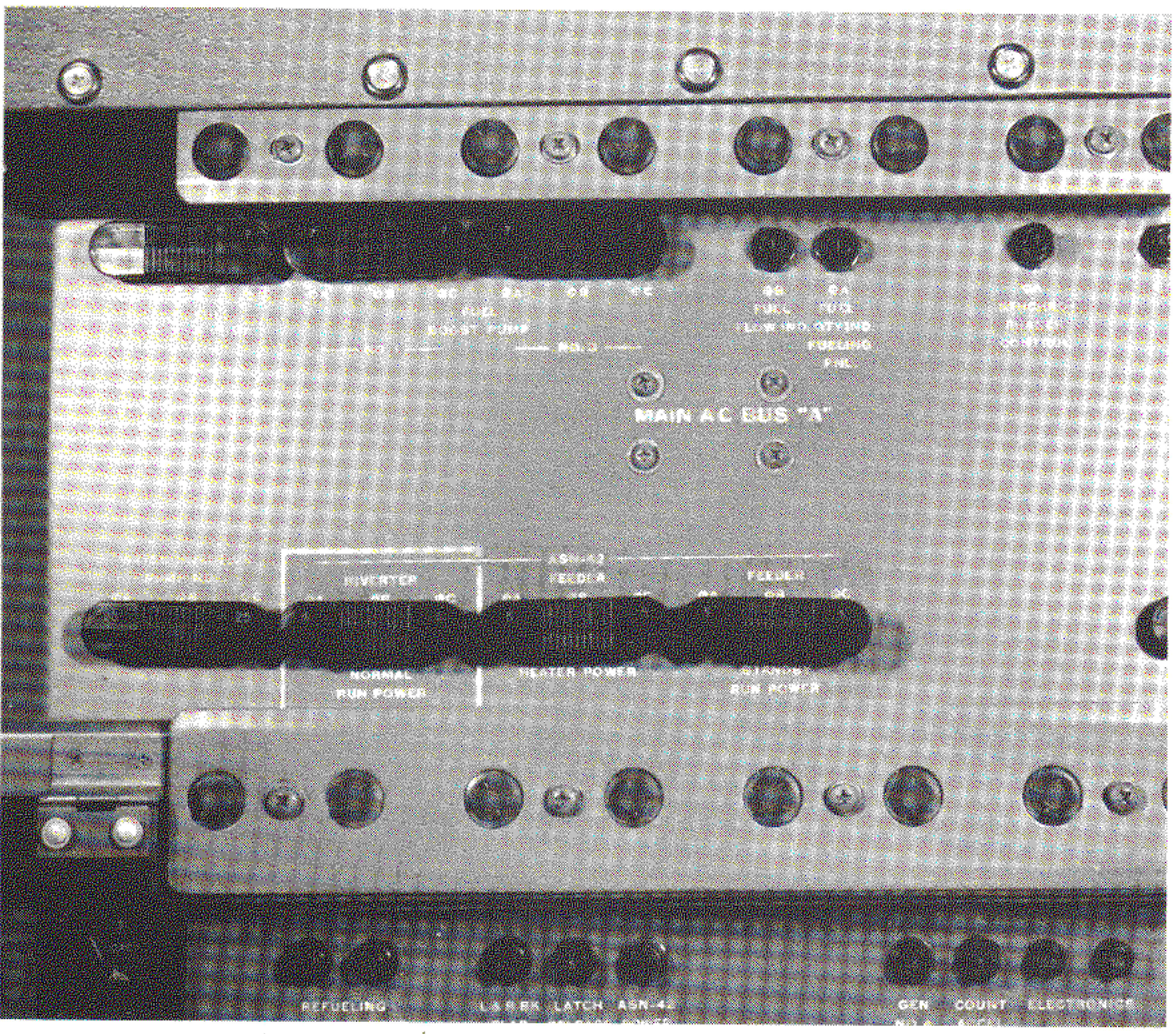


Figure 23 Inertial System Circuit Breakers — MAIN AC BUS "A"

In the event the inverter output drops to approximately 100 volts or to 375 cps, the inverter under-voltage/under-frequency sensitive relay transfers the system RUN bus to aircraft 115-V, 3 phase, ac power. The same relay also completes the ground circuit for the Navigator's INVERTER OFF warning light located on the modified DOPPLER DECOUPLER CONTROL unit.

All critical circuits within three inertial system units are rewired to the 115-V ac RUN bus input. The affected units are the Computer, the Electronic Control Amplifier, and the System Power Supply. Since the need for the auxiliary dc power supply feature of the AU 0400 Converter-Decoupler is eliminated by the inverter modification, this unit is modified to retain only the decoupler function and the unit is reidentified and reinstalled.

On aircraft BUNO 152141, and on BUNO 152164 and subsequent the inverter and associated changes are incorporated on a production basis. The AU 0400 unit is replaced by a Lockheed manufactured A 173 Doppler Decoupler unit incorporating the external test points and self test features in evidence on circuit diagram Figure 24. In addition, the input signal level is adjustable (150 to 300 millivolts) and is set to decouple at approximately 225 millivolts (equivalent to 13.5 knots). All aircraft wiring to the Doppler Decoupler, including the power circuits shown on Figure 19, is unchanged and the A 173 Doppler Decoupler circuit is designed to be directly interchangeable with the modified AU 0400 Converter-Decoupler. One circuit breaker nameplate at the Flight Essential DC Bus panel is changed from INERTIAL NAV POWER to ASN-42 DECOUPLER.

Any change in aircraft equipment invariably brings up the question of interchangeability. In the interim period while the inverter modification is in process, unmodified inertial systems can safely be used in modified aircraft. Conversely, modified sys-

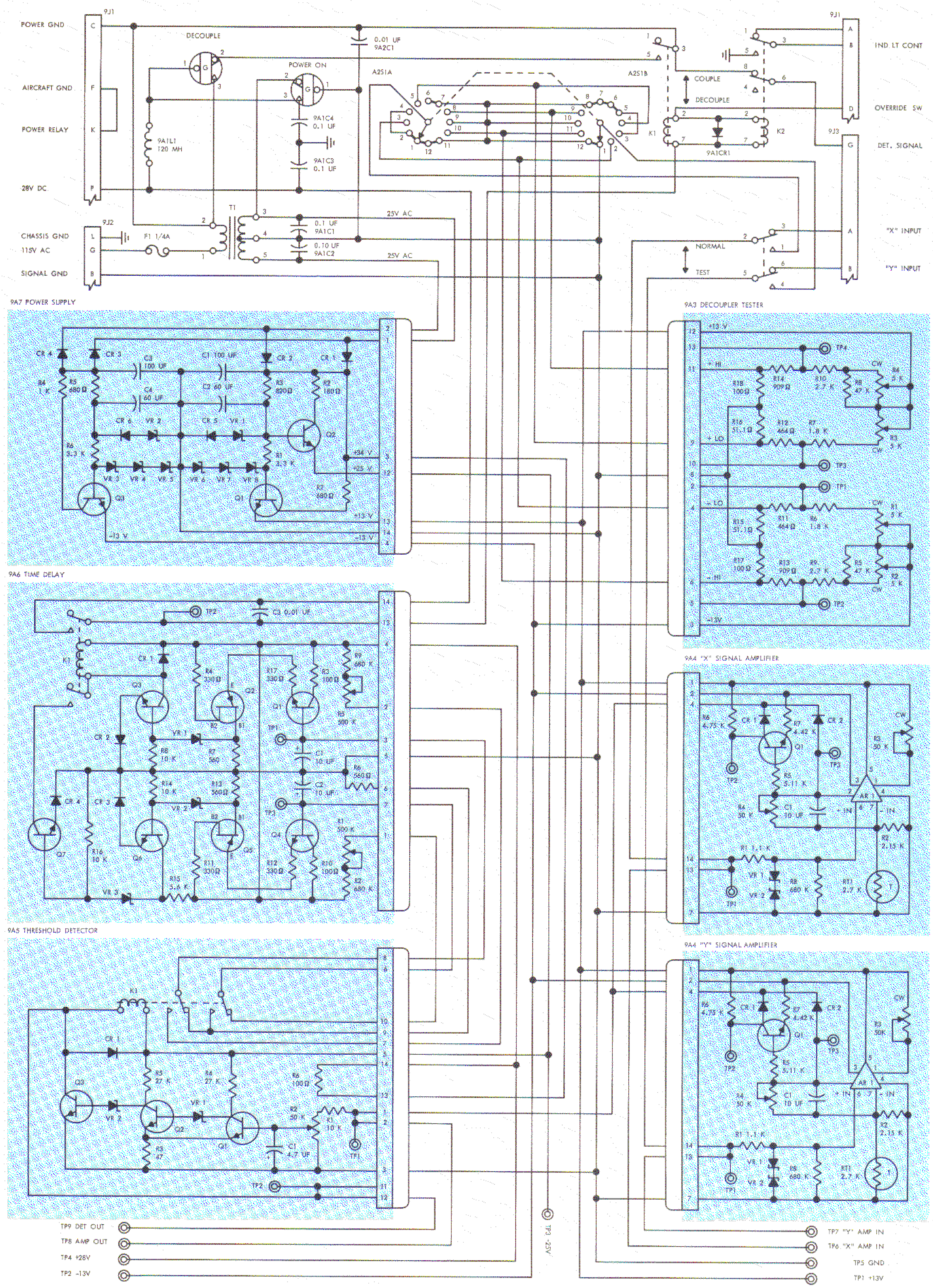
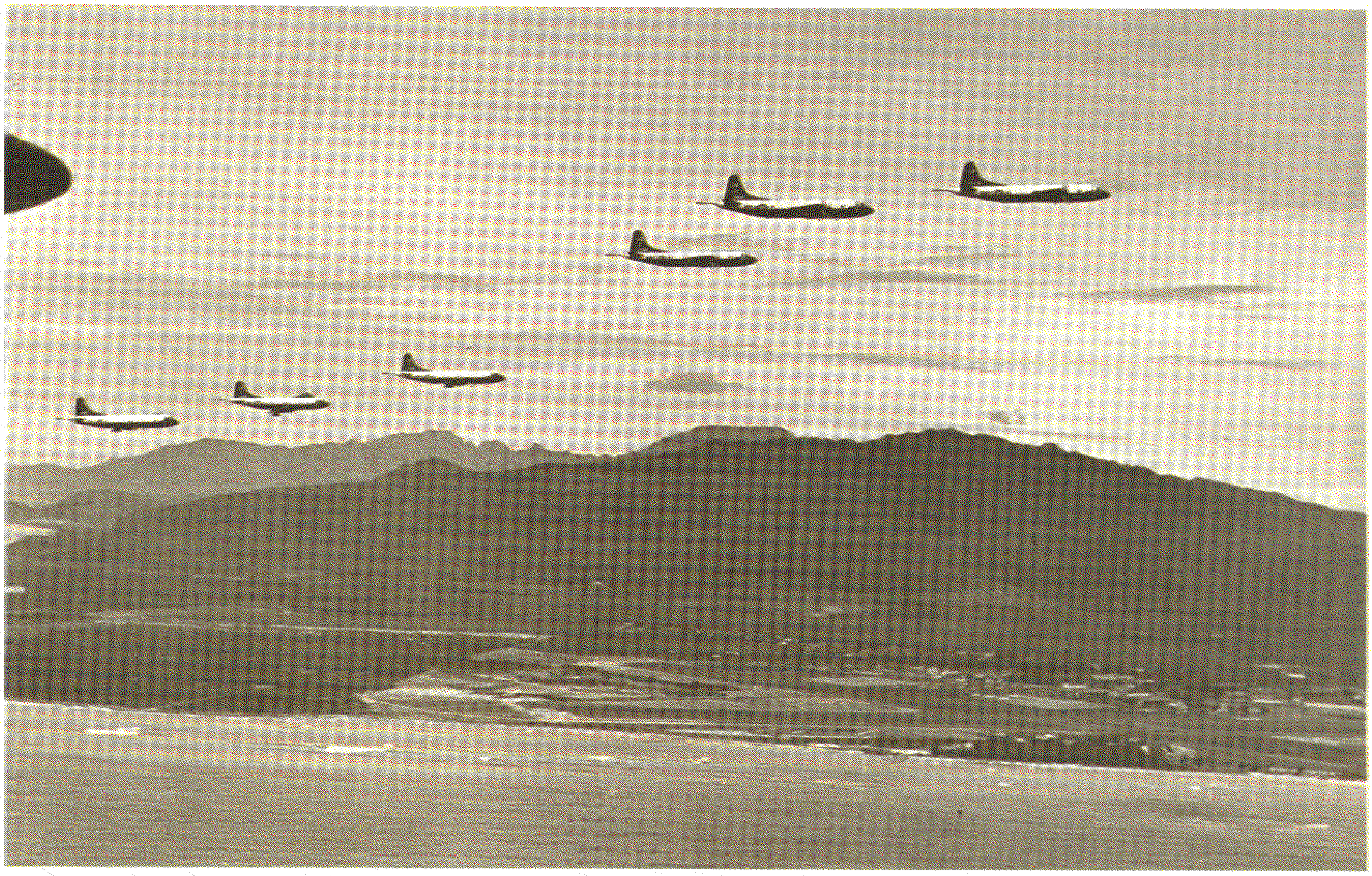


Figure 24 A 173 Doppler Decoupler Schematic



tems can be used in unmodified aircraft *provided* the unmodified AU 0400 Converter-Decoupler is retained. However, except for the AU 0400 just noted, modified and unmodified units should *not* be operated as a part of the same system.

Included as a part of the current updating program is the High Aircraft Motion (HAM) modification. This revision involves wiring changes in the inertial system Computer to materially reduce the effect of aircraft motion—caused by wind and/or personnel movement within the aircraft—during the "fine" alignment (leveling and gyrocompassing) phase. Basically, the change alters the Computer relay logic to include the existing "In-Flight Align"* relays as a part of the "Ground Align" circuit. These relays modify the feedback path of the Operational Amplifiers used for leveling and gyrocompassing, thereby changing the gain of the amplifiers as well as altering their response time to the motion-induced transients sensed by the accelerometers.

Some operating experience with modified aircraft has been accumulated. One squadron, operating updated P-3's in Arctic regions under the handicap of extreme high wind and low temperature conditions, reports a decided increase in mean-time-between failures as well as a distinct improvement in enroute accuracy with terminal errors reduced nearly to zero. The improved systems, coupled with an aggressive and optimistic approach to all phases of utilization and maintenance, resulted in this squadron reporting "no problem area" insofar as the inertial system is concerned.

**The facility for system alignment while airborne was never developed to the operational stage. The relays originally provided for this will now be used in Ground Alignment; selector position IN-FLIGHT ALIGN remains inoperative.*

ALTERNATE DATA SOURCES

In line with the practice of providing an alternate or secondary source of vital information for the Orion, it is well to mention here that the same philosophy is followed in providing latitude and longitude position indications. The alternate P-3 navigation source involves more than one avionic system, each of which would require considerable explanation. Therefore, we will say only that both sources are provided simultaneously, and that a second Position Indicator (ID-888) presents latitude and longitude computed by the Doppler/Air Mass (D/AM) computer. The D/AM computer, in turn, receives ground velocity information from one or the other of two sources: Doppler Radar velocity heading and velocity drift, *or* True Air Speed (TAS) system heading velocity and a manually set N-S and/or E-W wind speed value inserted on the D/AM Position Indicator. The D/AM computer, however, requires true heading for its computation which is normally supplied by the inertial system. In the event the inertial system cannot provide true heading, the D/AM computer combines magnetic heading with manually inserted variation to produce its own source of true heading, a process somewhat less accurate than that used by the inertial system. The accuracy of latitude-longitude indication will be affected accordingly. ▲▲

The Lockheed Product Support Organization

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SERVICE REPRESENTATIVES, P-3 ORION

LOCATION	NAME	MAILING ADDRESS	TELEPHONE
Patuxent River NATC, Maryland	D. H. Horadam, Regional Rep.	P.O. Box 218, NATC Patuxent River, Maryland 20670	Volunteer 3-3111 Ext. 645 or 251
Brunswick, Maine	F. Hampton, Resident Rep.	P.O. Box 241 Brunswick, Me. 04011	207-725-5561 Ext. 513
Norfolk NAS, Virginia	A. Barber, Resident Rep.	P.O. Box 8127, Ocean View Station Norfolk, Virginia 23503	703-444-8023
Jacksonville NAS, Florida	J. Gipson, Resident Rep.	P.O. Box 1300, Yukon, Florida	904-389-7711, Ext. 8724 NAS Jacksonville
Key West NAS, Florida	S. Brown, Resident Rep.	P.O. Box 1087, Key West, Florida 33040	305-294-4417, Key West
Moffett Field NAS, Calif.	F. Hays, Resident Rep.	Lockheed P-3A Office Unit One P.O., General Delivery NAS Moffet Field, Calif.	961-1717 Mountain View, California
North Island NAS, San Diego	R. B. Hedin, Resident Rep.	P.O. Box 525 Coronado, California 92118	435-6611, Ext. 1272 Coronado, Calif.
Honolulu, Hawaii	R. C. Spooner, Regional Rep.	P.O. Box 409 FPO San Francisco, Calif. 96611	430-66126 Honolulu, Hawaii
Honolulu, Hawaii	T. Snelling, Resident Rep.	P.O. Box 409 FPO San Francisco, Calif. 96611	430-64257 NAS, Barbers Point, Hawaii
Burbank, California	R. E. Kananen, Supervisor P-3 Project, Customer Service	Lockheed-California Company P.O. Box 551, Burbank, California 91503	847-4680 Burbank, Calif.

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