R-544

USE OF BODY-MOUNTED INERTIAL SENSORS IN AN APOLLO GUIDANCE, NAVIGATION AND CONTROL SYSTEM

by

J. McNeil, J. E. Miller, J. Sitomer

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USE OF BODY-MOUNTED INERTIAL SENSORS IN AN APOLLO GUIDANCE, NAVIGATION AND CONTROL SYSTEM

ABSTRACT

The general configuration of the Apollo guidance, navigation, and control system is briefly reviewed, Details are presented concerning the inertial measurement unit subsystem. A description is provided of a strapped-down *or* gimballess inertial measurement unit utilizing single-degree-of-freedom integrating gyroscopes. The basic pulse rebalanced sensor subsystem is described. Attitude algorithms and techniques for their computer mechanization are briefly discussed. An advanced design gimballess IMU is described and explained, utilizing the 18 size pulsed inertial reference integrating gyro and integrated circuit guidance electronics. An illustrative sensor package having a volume of 0.2 cubic ft and weighing less than 20 lbs is presented as part of a discussion of the potential of body mounted inertial sensors. The general topic of replacing the present IMU with a gimballess (strapped-down) IMU is discussed. Functional, mechanization, and interface changes are explained to define the impact of such upon an Apollo type system. Finally, topics related to the complete redesign of a GIMU equipped Apollo type system are included.

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J.E. Miller
J. Sitomer

April, 1966

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CHAPTER I

INTRODUCTION

Strapped-down, gimballess, or body-fixed guidance is an often proposed means of mechanization of inertial measurement units (IMU) for aerospace guidance, navigation, and control systems. To place this type of mechanization in proper perspective, one should acknowledge the following:

a. Almost any inertia reaction measurement function can be implemented with a properly instrumented gimballed IMU, and an associated computer.

b. The strapped-down sensor IMU is no more than a different mechanization of a means to acquire acceleration and angular velocity information.

c. The proper choice of axes when making measurements and processing the information obtained may significantly influence the mass, volume, reliability, and cost of the entire vehicle guidance, navigation, and control system.

The Apollo guidance, navigation, and control system utilizes information from inertially stabilized accelerometers and body-mounted optics and radar. Since the time of Apollo's configuration, computer and pulse rebalanced inertial component technologies have advanced enought to permit the fabrication of a true body-fixed IMU. This paper discusses two questions:

a. What value might there be in just replacing the gimballed inertial reference with a gimballess inertial measurement unit (GIMU) in an Apollo type system?

b. What value might there be in designing a future Apollo type guidance, navigation, and control system around the GIMU sensor mechanization?

CHAPTER II

THE APOLLO GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

The Apollo guidance, navigation, and control system is utilized both in the Apollo command module (CM) and in the lunar excursion module (LEM), (see Ref 1). Both sets of equipment are nearly identical except for the optical unit and the essential controls. The general description of the system will relate particularly to the command module. Shown in Fig. 1 is the location of the CM guidance, navigation, and control system. The system's role in the lunar mission is as follows:*

a. Navigation - translational measurement and control in free fall

b. Attitude control - rotational measurement and control in free fall

c. Guidance - translational measurement and control during acceleration

d. Thrust vector contrcl - rotational measurements and control during acceleration.

The system consists of a set of sensors, one of which is the IMU used for maintenance of the inertial reference coordinate frame. The accelerometers are used for measurement of the integral of specific force of the vehicle measured in the inertial reference frame.

The scanning telescope (SCT) is a two-degree-of-freedom (60 degree field) optical sensor used to find targets required for alignment and navigation measurements. The Sextant (SXT) is a 1.8 degree field two-line-of-sight sensor used to make the precision alignment and navigation measurements.

The PSA, power and servo assembly, are the analog electronics for operation of the IMU and its sensors. The AGC, Apollo guidance computer, occupies a central role in the CM guidance system. It serves as a processor for all sensor data, computation of commands, and deliveries of these commands to thrust or torque producing jets and to give a display of information to astronauts. In addition, the astronaut may communicate with the AGC via a keyboard and the AGC may communicate with the astronaut via a display. The communication between the IMU for the gimbal angles and for the AGC is via the CDU (coupling display unit). This element of the system converts the sine and cosine outputs of

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*See Fig. 2 .

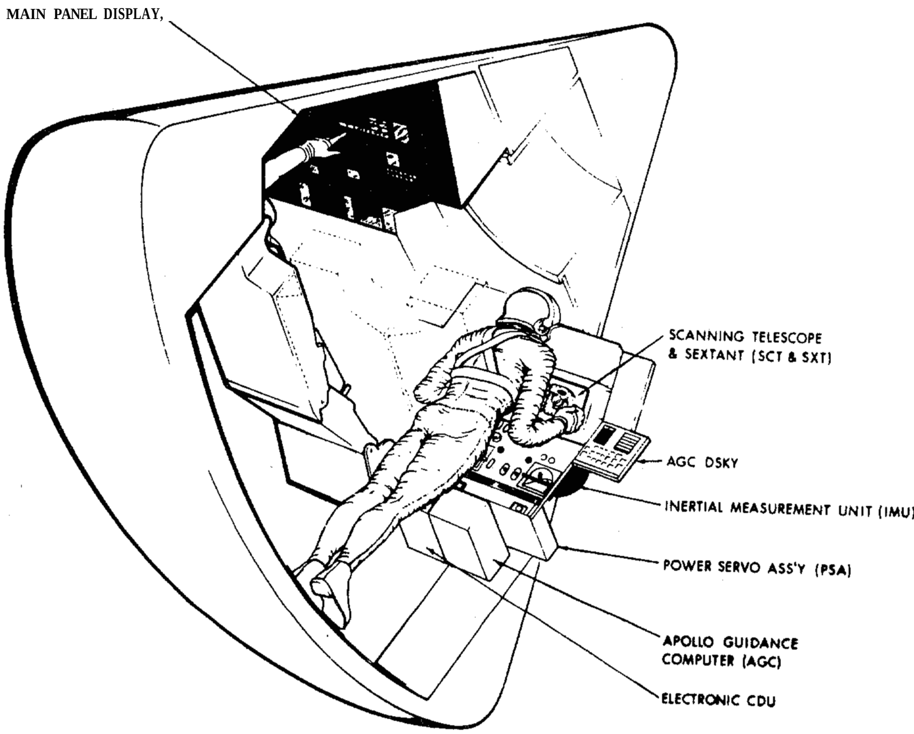


Fig. 1 Location of the guidance and navigation system in the command module.

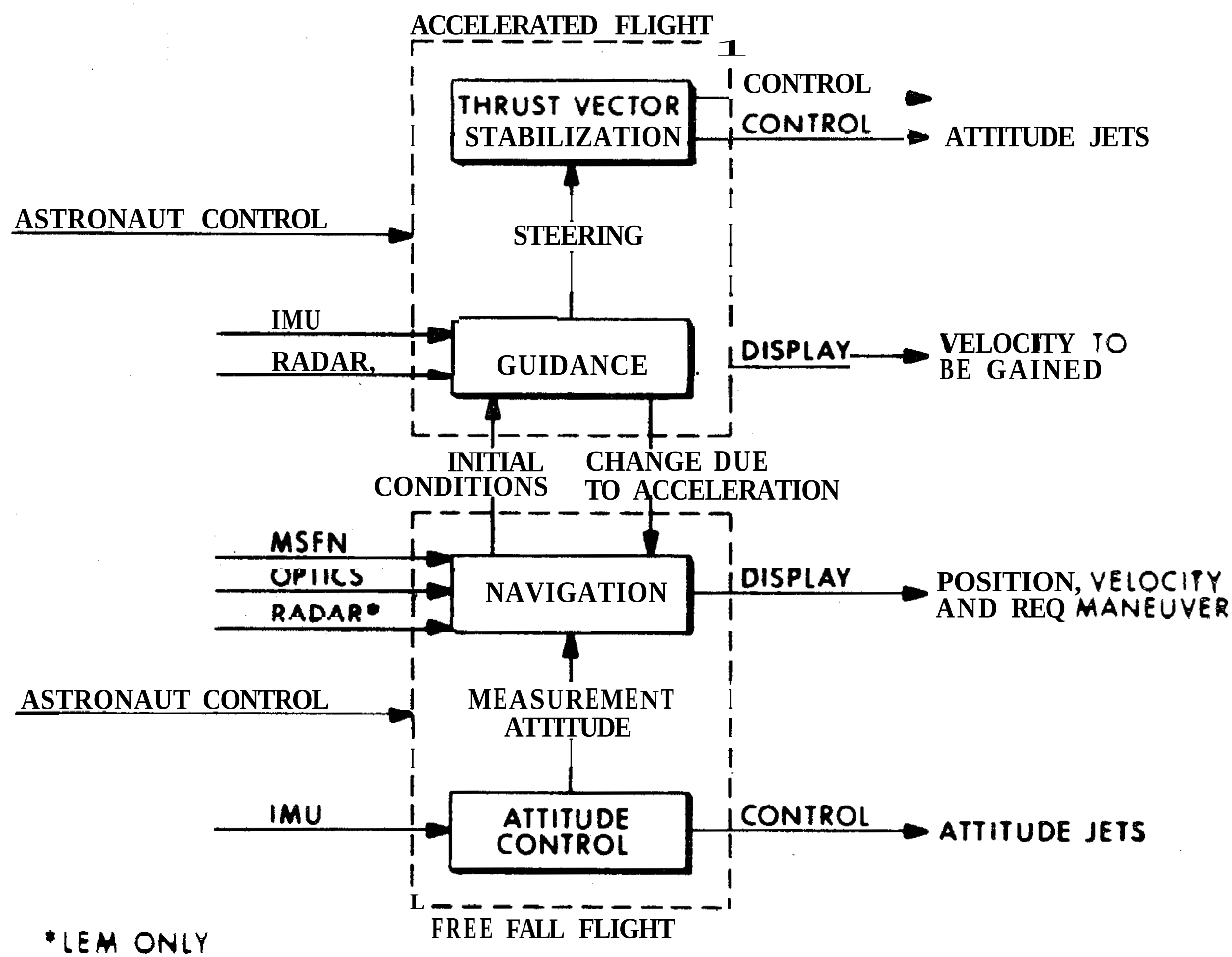


Fig. 2 **Apollo** guidance and navigation function flow.

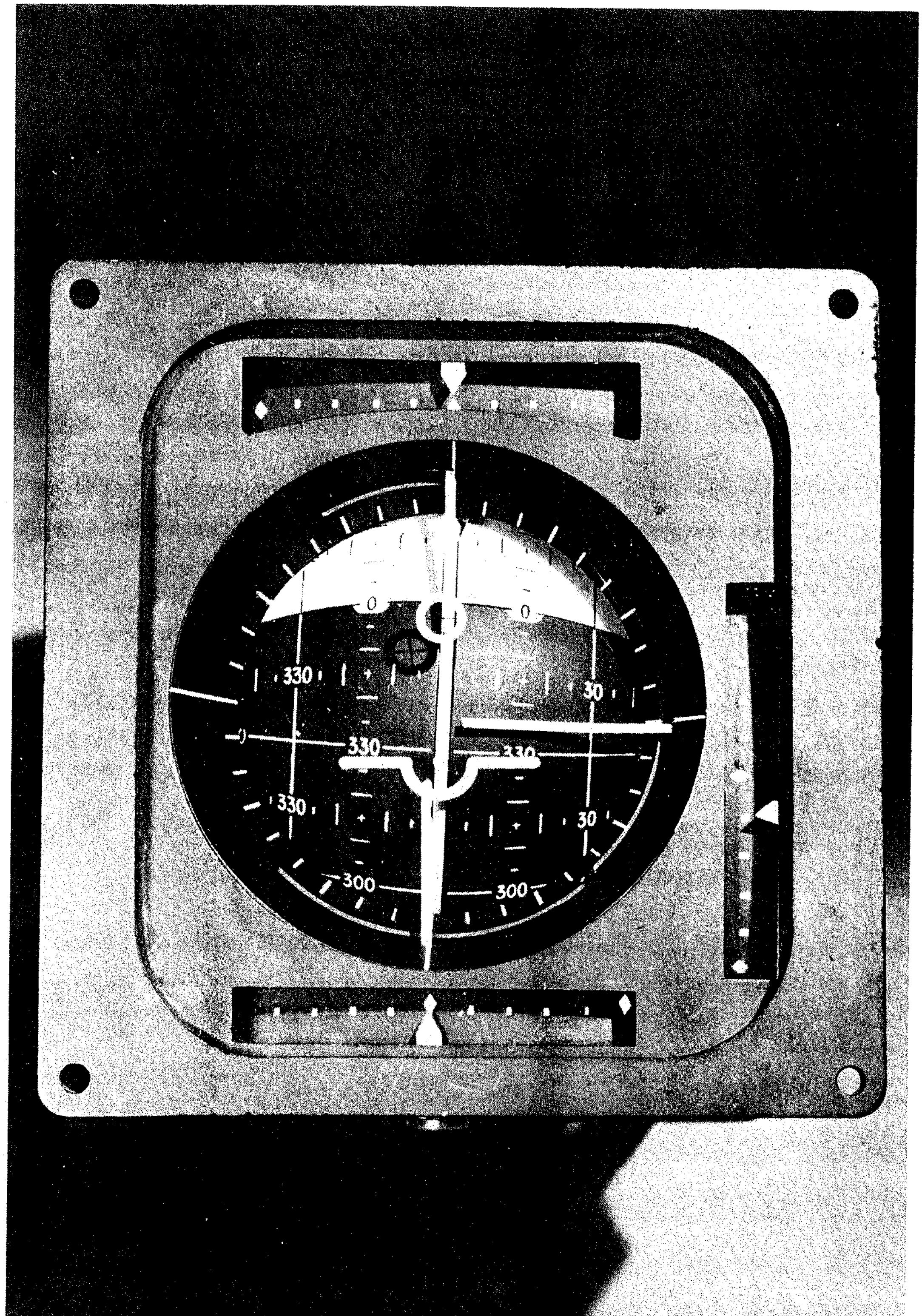


Fig. 3 Ball attitude indicator.

of the IMU gimbal angle resolver to a whole angle stored in a counter in the CDU. The contents of this counter are also stored in the AGC; thus, the two counters are incremented together, and the IMU gimbal angles are known by the AGC. The CDU provides the additional function of measuring the orientation of the optics line of sight in the same fashion and providing the AGC with information. The CDU also provides a digital to analog conversion for display.

The attitude information of the stable member axes of the IMU with respect to the CM axes is computed via a matrix transformation utilizing the three IMU gimbal angles. The accelerometer information is available in inertial coordinates and the output of the AGC for thrust vector control is in body or CM coordinates. Again, the same matrix conversion must be applied.

In addition to the displays of the AGC and those of the CDU commanded by the AGC there are a set of status and warning lights to alert the astronaut as to the status of the guidance, navigation, and control system. These are displayed on the same display as that of the AGC digital information. There is one other display that is of primary interest to the astronaut. This is the total attitude display or flight director attitude indicator (FDAI), commonly called "eight ball". Visual display is the orientation of the spacecraft with respect to the IMU stable member axis. Shown in Fig. 3 is the ball attitude indicator. The needles along the sides are the attitude error needles driven by the CDU.

CHAPTER III

A GIMBALLESS INERTIAL MEASUREMENT UNIT

3.1 The Single-Degree-of-Freedom Gyro Concept

The gimballess concept requires that angle information be obtained in a body coordinate frame, and that the information be processed in a computer which will continuously possess information as to the orientation of the body axes with respect to a set of imaginary stabilized (*or* reference) axes. Body angle information may be obtained from laser, electro-static, single-degree-of-freedom platform, acceleration, angular rate, and angular rate integrating sensors. This paper will only consider the use of the rate integrating gyro.

Figure 5 shows the Apollo II IRIG, the gyro presently being used in the Apollo guidance, navigation, and control system, It is a single-degree-of-freedom integrating unit equipped with a ducosyn signal generator and "V" connected ducosyn torque generator. A detailed description is given in Ref 1. In its present form, it is suitable for use as a pulse rebalanced sensor in low angular velocity environments. Two modifications, namely an increase in the torque limit of the electromagnetic suspensions, and the replacement of the ducosyn torque generator with a permanent magnet D'Arsonval type, make it suitable for a gimballess IMU to be used in space vehicles.

Present day magnetic materials permit the fabrication of permanent magnet torque generators with a drift in sensitivity insignificant in an Apollo type application,

3.2 Pulse Rebalanced Inertial Components

Figures 6 and 7 are block diagrams of the 16 PIP accelerometer used in Apollo. Figure 8 shows the pendulum. The description which follows for a binary device is basic for the pulse rebalanced accelerometers and gyros as designed at MIT. A simple change in the electronics converts it to a ternary device.

Basic operation - the inertial sensor is a single-degree-of-freedom pendulum with a pendulosity of 1/4 gm cm.

The pendulum is mounted on the stable member and is non-rotating with respect to inertial space. The signal generator is a variable reluctance device termed a signal generator microsyn. It is excited from a sinusoidal source which is synchronized and phase locked to the guidance computer. The signal generator output is voltage modulated by the angle of the pendulum float with respect to case (A_{c-f}) and the float angular velocity, (A_{c-f}). The voltage may be considered to contain only float angle information. This signal, amplified, is used in an interrogator. The interrogator is a peak detecting device used to determine sign and magnitude of the float angle at discrete times. These discrete times are the computer clock times, (AT), synchronized with sinusoidal excitation, Only float angle sign is determined and the operation is binary. The interrogator output is a command to torque the float angle to null. The current switch directs a constant current source into either the odd poles or even poles of an eight pole torque microsyn. The torque generator creates a torque either positive or negative proportional to the square of the current in the windings. Current is controlled constant by comparison of a precision voltage reference with the voltage drop across a precision resistor in the current loop.

3.3 Attitude Algorithm

There are numerous ways of representing the orientation of body axes with respect to the stabilized (or reference) axis. The most obvious are Euler angles and direction cosines. The former usually involves products of transcendental functions, and the latter, a simple function as the means of locating an attitude relationship between two axes. In addition, direction cosines permit all attitude and simple matrix multiplication mechanizations. The nine direction cosine matrix is currently being used in strapped-down systems at MIT/IL.

Algorithms to represent a technique of updating the direction cosines appear in technical literature as first, second, and fourth order equations. This is because, for practical space computers, no absolute attitude solutions exist. As an illustration, a first order derivation is presented below:

$$C_{ij} = \overline{I}_{Si} \cdot \overline{I}_{Bj}$$

$$\frac{d}{dt} C_{ij} = \overline{I}_{Si} \cdot (\overline{W}_B \times \overline{I}_{Bj}) + (\overline{W}_S \times \overline{I}_{Si}) \cdot \overline{I}_{Bj}$$

Where:

$\overline{W}_B$ = angular velocity of body frame

$\overline{W}_S$ = angular velocity of stabilized frame

$$\underbrace{\dot{C}_{ij} = C_{i,j+1} W_{Bj+2} - C_{i,j+2} W_{Bj+1}}_{\text{Body Axis Rates}} + \underbrace{C_{i+1,j} W_{Si+2} - C_{i+2,j} W_{Si+1}}_{\text{Stabilized Axis Rates}}$$

Fig. 4 Transformation Computer updating equations.

Fig. 4 (cont.)

Where:

$$\Delta C_{ij} = C_{i,j+1}\Delta\theta_{j+2} - C_{i,j+2}\Delta\theta_{j+1} + C_{i+1,j}\Delta\psi_{i+2} - C_{i+2,j}\Delta\psi_{i+1}$$

$$\Delta\theta = W_B \Delta t, \; \Delta\psi = W_S \Delta t$$

Example:

$$\Delta C_{xx} = C_{xy}\Delta\theta_z - C_{xz}\Delta\theta + C_{yx}\Delta\psi_z - C_{zx}\Delta\psi_y$$

Several important facts should be noted:

(1) The inertial sensor performance will usually contribute more to performance errors than to the algorithm.

(2) The algorithm should be tailored to the mission so as to minimize computer mechanization complexity.

(3) The mechanization of the algorithm in the digital space computer is often more important than the quality of the algorithm itself. (See ref 2).

3.4 Algorithm Mechanization

The Apollo type system will utilize a digital computer. Attitude information will have an uncertainty due to quantitized gyro angle and computer information. During the solution time required to update the cosines, the vehicle angular velocity may change, thus introducing an integration error of the vehicle angular velocity. The order and technique of solving the updating equations will control the attitude error growth even for a constant angular velocity input. See Ref 2.

Incremental updating mechanizations tend to produce attitude errors as a function of total (neglecting sign) angle activity. These may not be suitable for angular vibration environments. Whole number updating tends to produce attitude errors as a function of time. The collection of incremental angles filters out oscillations.

However, the phenomenon of non-commutivity of angles must not be ignored, nor must the information bandwidth required for a particular mission. A large thrust vehicle with high maneuver rates requires frequent updating of the cosines to insure the proper transformation of accelerometer information. In contrast, the Apollo type system will thrust at constant or slowly changing attitude, thus permitting the use of a higher order algorithm, updated upon demand, and statistically, at a low frequency.

Future vehicles will have a central computer (or brain) with sensors and actuators attached to it. The use of the independent transformation computer occurs when there is a real time traffic problem at the central general purpose computer, and when computations are to be done in special purpose subsystems.

A proper repertoire of instructions and proper interface design should insure the compatibility of the GIMU with the expected spacecraft computer.

3.5 An Advanced Design GIMU

Using the concepts previously discussed, a high performance strapped-down sensor assembly can be designed with the use of the most recent development of inertial sensors and electronics. The following are the design goals of such a package:

(1) Performance necessary to meet the specifications of an Apollo type mission

(2) High reliability
(3) Small size, volume and weight
(4) LOW input power requirements
(5) Simplified in-flight repair
(6) Ease of calibration
(7) Simplification of the physical interfaces
(8) Low cost, consistent with the above requirements

An inertial package consisting of three 18 PIRIGs and three 16 PM PIPs is shown in Figs. 9 and 10. This package is a complete inertial assembly consisting of all the pulse-torquing electronics, the ac and dc power supplies, temperature controllers, and clocking logic. All that is necessary to operate the package as an inertial assembly is the +28vdc from the fuel cells or batteries. The outputs of the package are body-referenced integrated angular rate from the gyros and integrated specific force from the accelerometers. These sensors have the performance capabilities of meeting the spacecraft requirements.

The pulse-torquing electronics for both instruments is identical. A typical form that it may take is shown in Fig, 11. The use of miniature circuits accounts for the small volume attained. Generally, two basic building blocks are used, a digital "nor" logic in the computer and a miniaturized linear amplifier which would be used in all other circuits where signal gain is necessary. The use of common building blocks cuts down on the amount of different components used, and, therefore, increases the reliability of the system.

The entire package is contained within a volume measuring 6 x 6 x 10 inches or about 0.2 cu ft. The six sensors are mounted on a component block which serves as the main member of the GIMU. If the component block is made of aluminum,

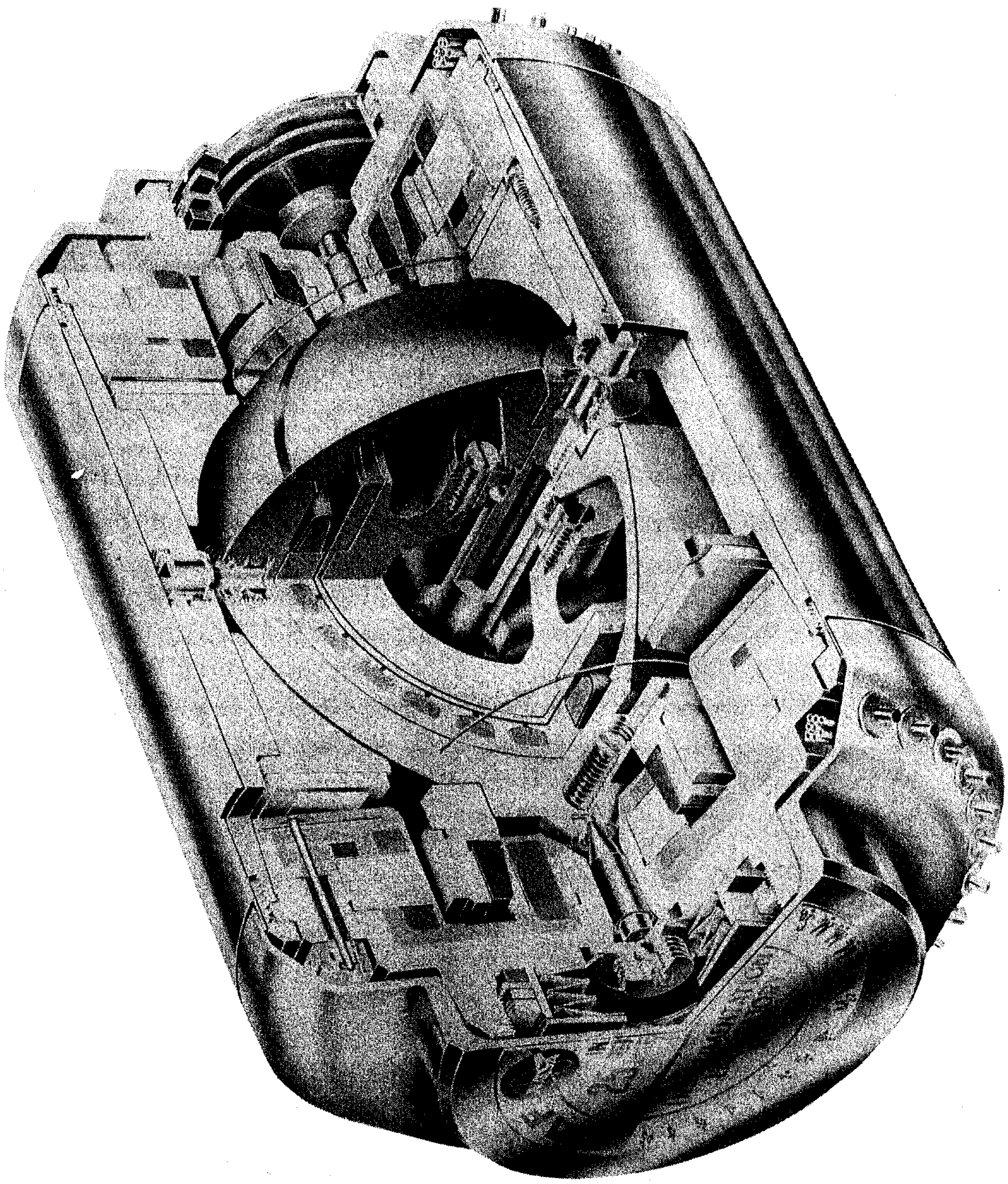
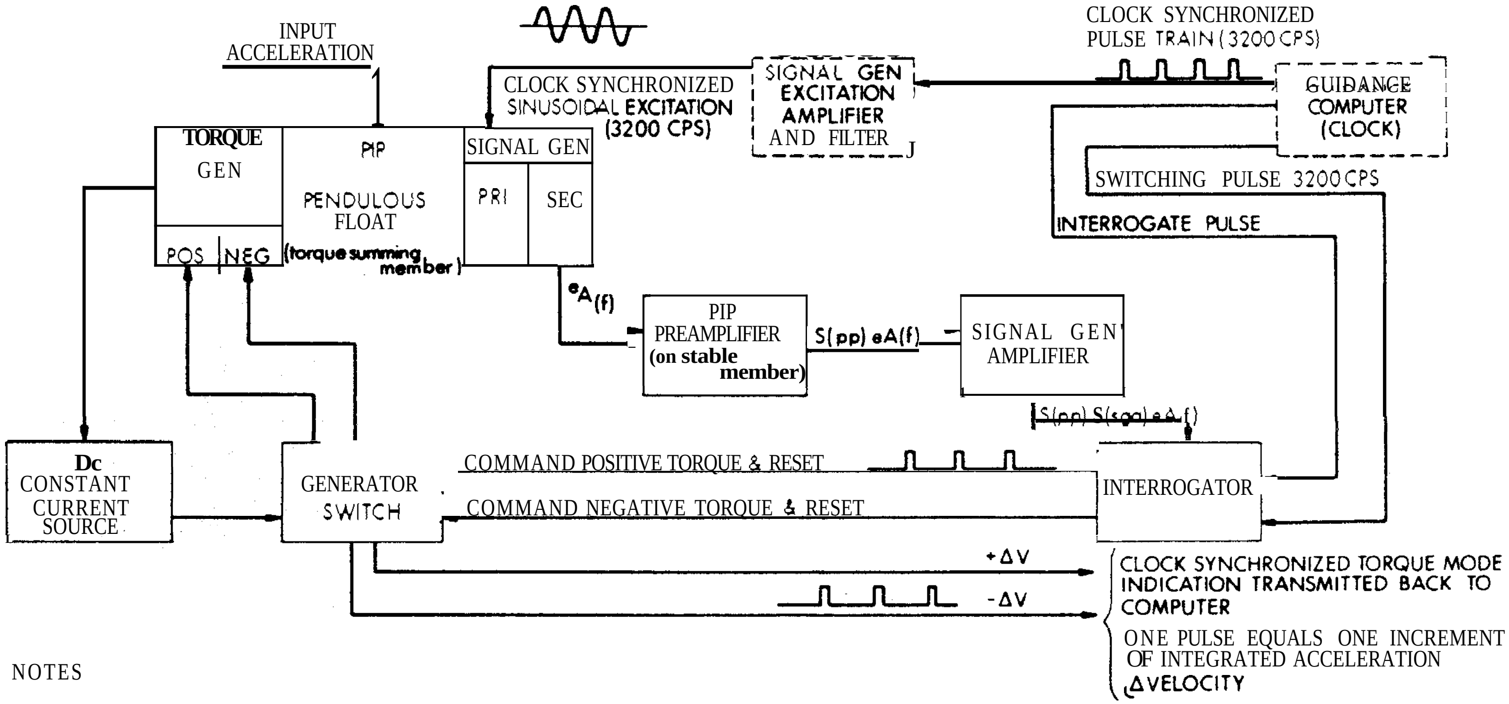


Fig. 5 Apollo II IRIG.



NOTES

1. PIP (PULSED INTEGRATING PENDULUM) 16 SIZE SINGLE DEGREE OF FREEDOM PENDULOUS UNIT
2. WAVE FORMS SHOW CONDITION FOR NEGATIVE INPUT ACCELERATION OF PIP CASE

AV COMMAND MODULE 5.85 CM/SEC/PULSE

ΔV LEM 1.0 CM/SEC/PULSE

Fig. 6 PIP accelerometer block diagram.

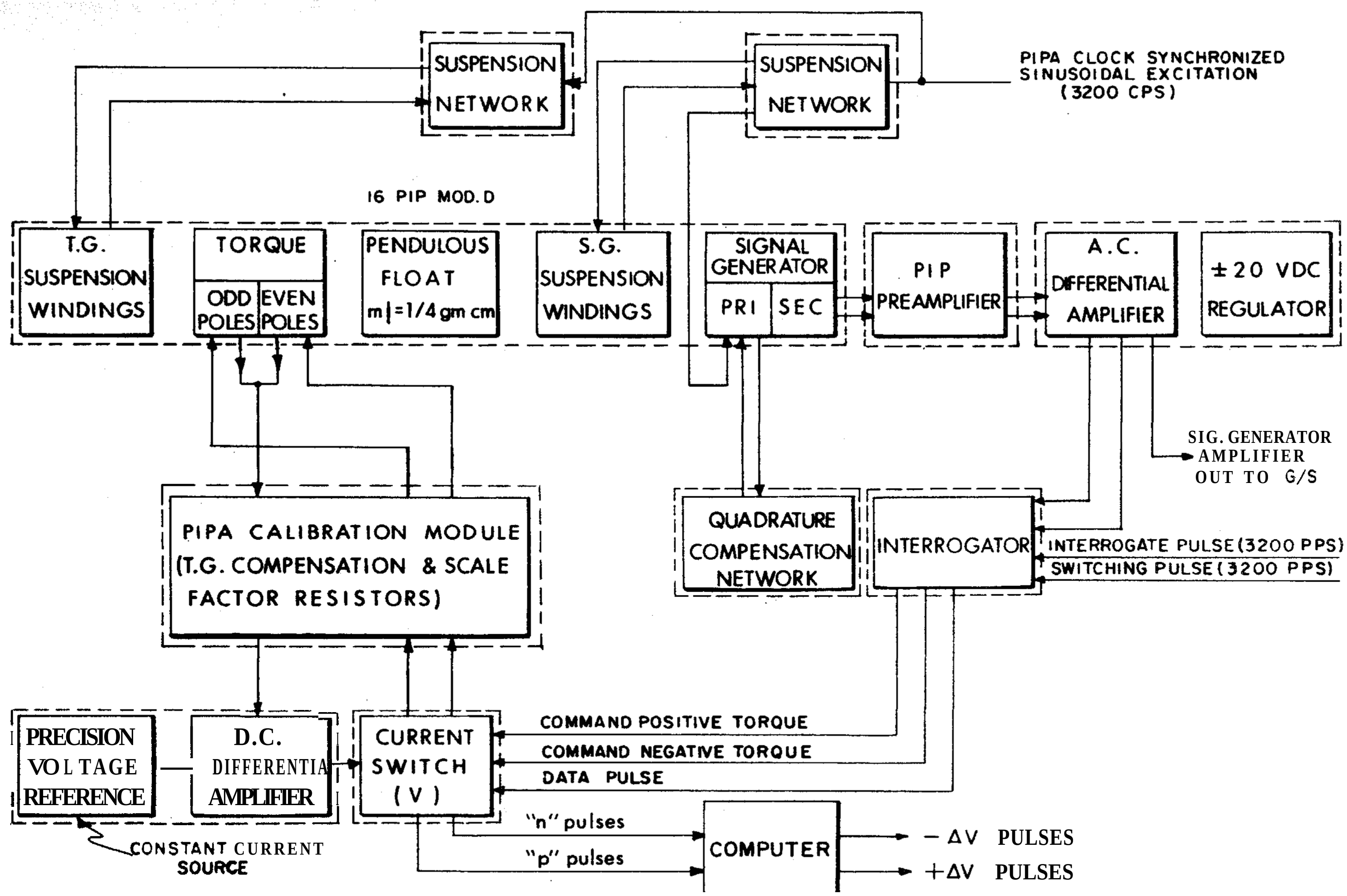


Fig. 7 PIP accelerometer module block diagram.

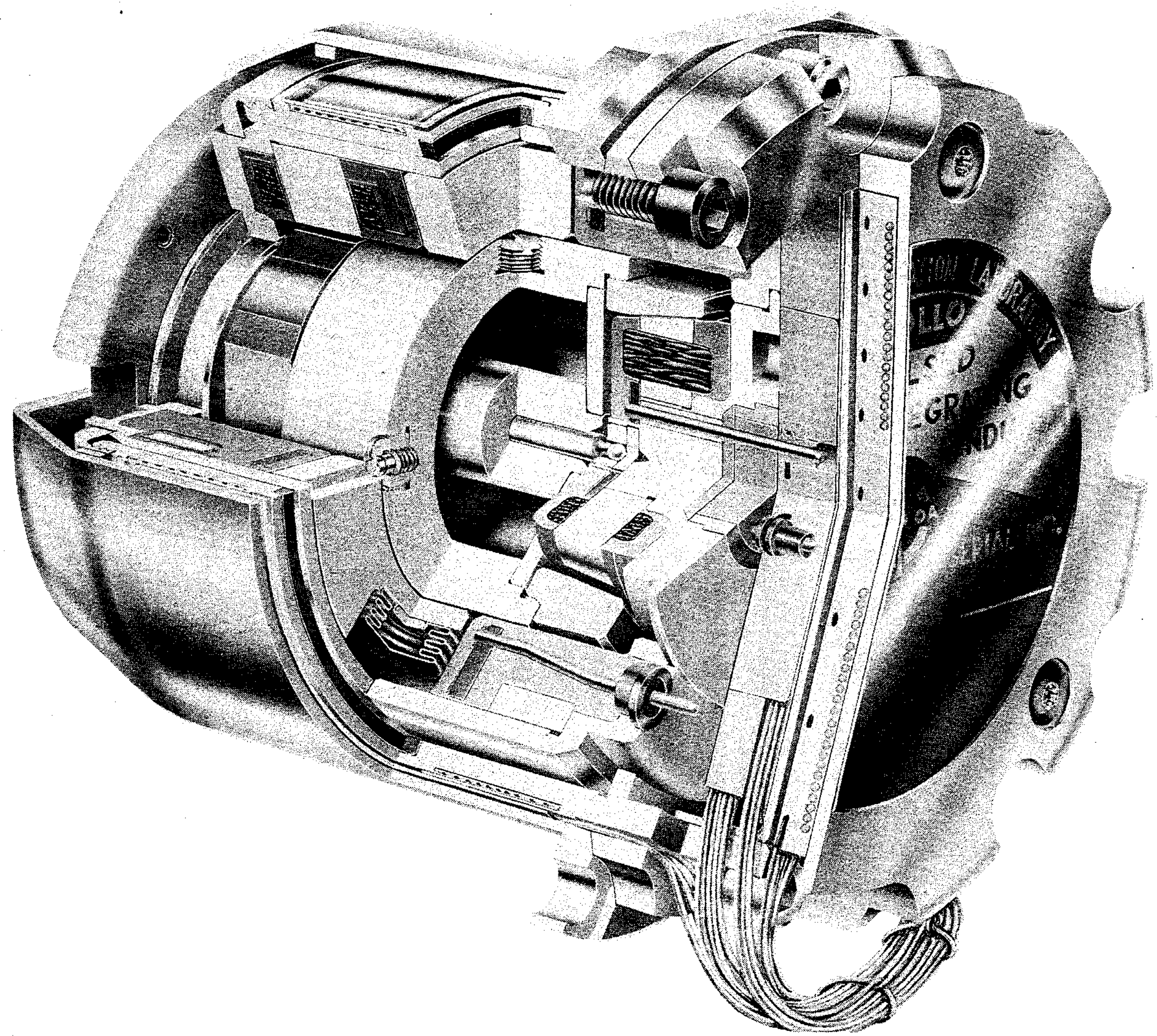


Fig. 8 16 pulsed integrating pendulum, mod D.

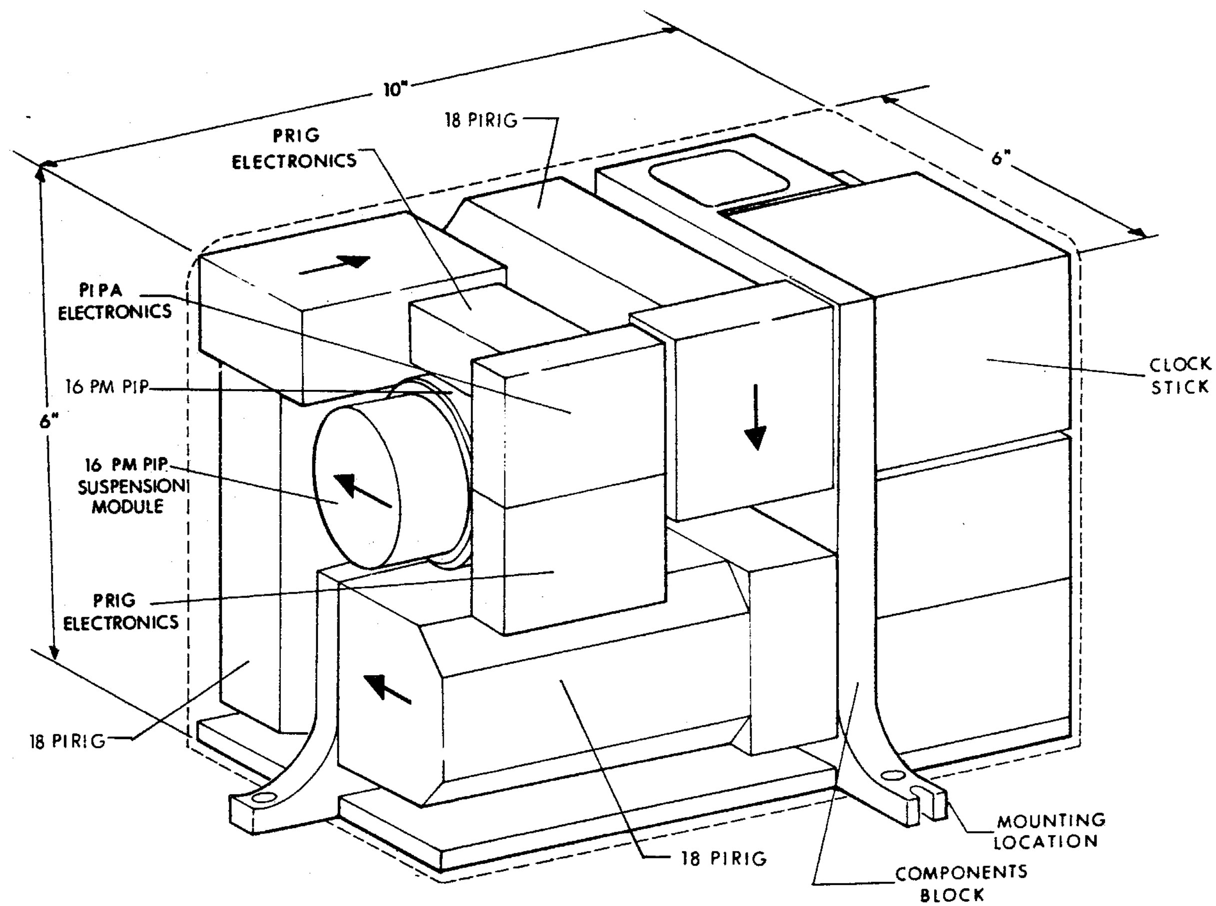


Fig. 9 GIMU assembly front.

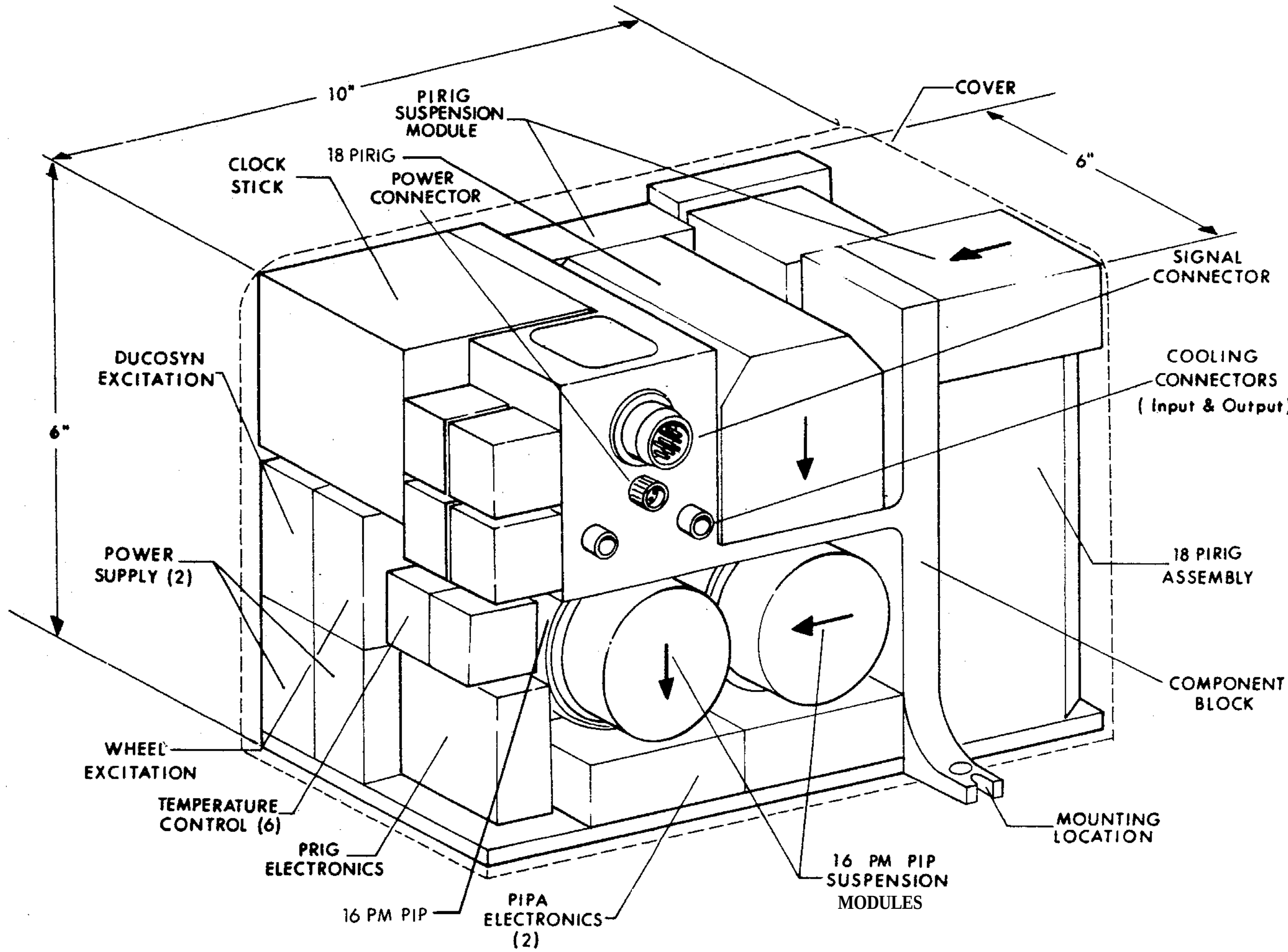


Fig. 10 18 GIMU assembly rear.

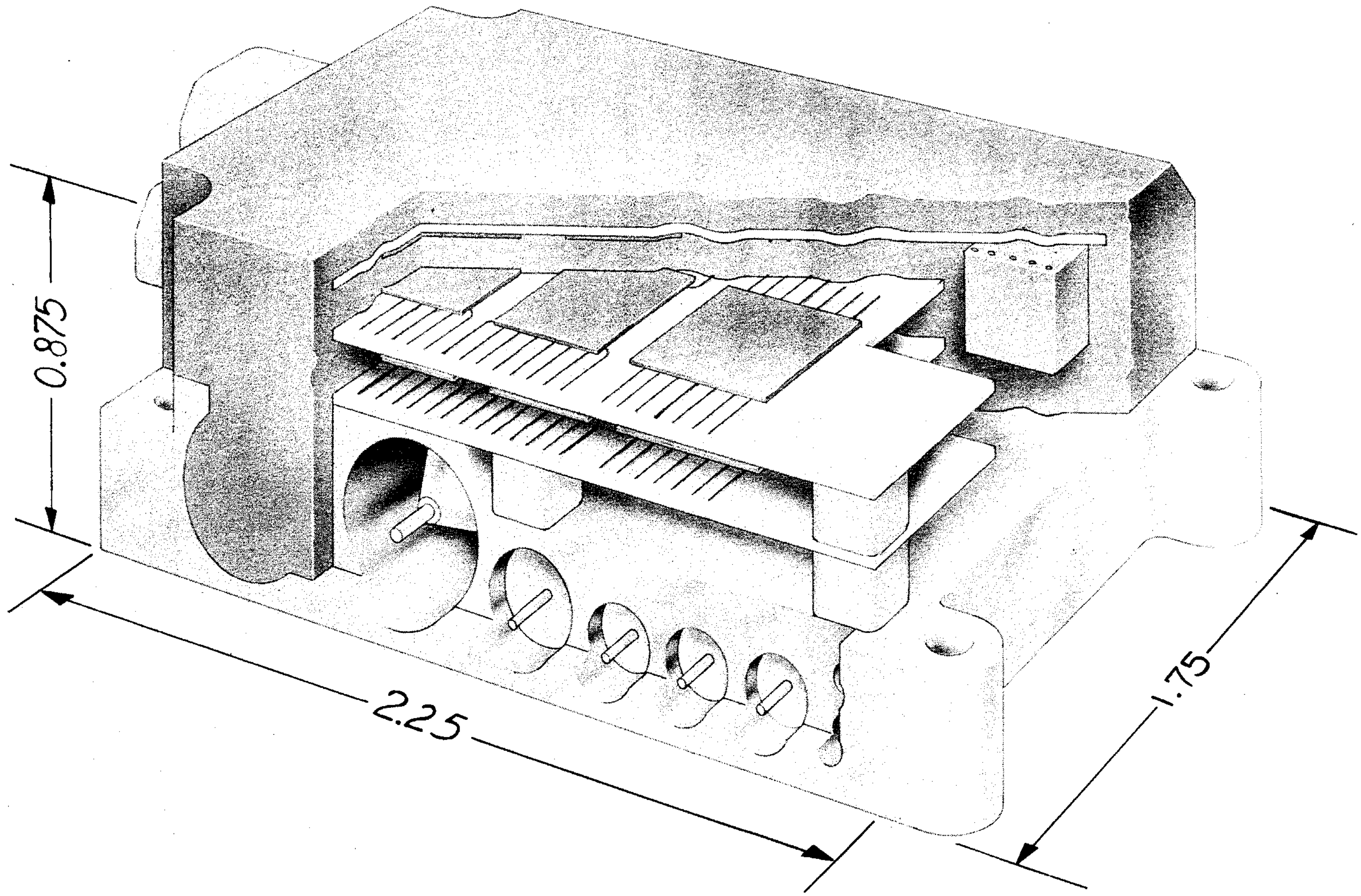


Fig. 11 PIP electronics module.

the weight of the assembly would be about 20 lbs including electronics. By using beryllium, the weight of the package could be reduced 3 lbs to 17 lbs.

The power dissipated within the package would be about 80 watts not including any heater power. The package is designed so that integral cooling passages may be cast throughout the component block.

The inertial components are all precalibrated and prealigned. The gyros are mounted within a housing or doghouse with the input axes of all three gyros oriented in the same direction relative to the housing, This means the gyro assignment need not be determined before prealignment. The accelerometers use an alignment ring scheme to mount to the component block--again not needing any pre-determined orientation. These features allow for ease of maintainability, the use of only one calibration procedure, simplified test procedures, etc., or more simply--ease of producibility.

Because the package is self-contained, the number of signal wires or cables is greatly reduced. In some Apollo applications the separation of certain guidance functions has occurred and, therefore, long lead lengths become a necessity. These long leads tend to put constraints upon the equipment because of the EMI problems that occur when these signals are led in common cables with other spacecraft signals. This packaging concept allows for a greatly simplified system integration within the spacecraft.

The package is enclosed by two covers, one over the electronics, and the other over the inertial components. Both covers create an environmental seal. Field or in-flight repair can be easily accomplished since all modules are plug-in modules and the inertial components are precalibrated and prealigned with every component or module being readily accessible and removable.

Ease of production generally leads to lower cost and maintainability. It also leads to lower servicing costs. Certain tradeoffs can be considered such as the choice between berylliumand aluminum. The aluminum is less expensive, but weighs more. There are many other tradeoffs that could be considered, but each one must be weighed in terms of a specific design. Generally, the design has been conceived to be as inexpensive as possible, commensurably with the goals of such an Apollo type mission.

CHAPTER IV

AN APOLLO TYPE SYSTEM WITH A REPLACEMENT GIMU

This chapter considers the Apollo instrumentation as described in Ref 1 as an illustration of a spacecraft guidance and navigation system. We shall examine the impact upon such a system of the replacement of the gimballed IMU and associated apparatus with a GIMU.

4.1 Functional Changes

The principal functional change is that the spacecraft would possess an all attitude inertial reference. Prepositioning of the reference axes because of anticipated spacecraft maneuvers would be eliminated. Reference axis orientation would be for efficient guidance and navigation computations.

4.2 Mechanization Changes

The use of a GIMU permits implementation of the IMU subsystem function with the following advantages:

a. Lower weight

b. Smaller volume

c. Reduced power consumption

d. Increased reliability through reduction in type and number of parts and elimination of certain low reliability components associated with gimbal assemblies

e. Attitude information is generated in digital form, thus eliminating A/D conversion hardware

f. In-flight or field maintenance easily accomplished with fewer parts without need for special environment or facilities (tools)

   Inertial components, complete with microelectronics, prealigned and calibrated, may be installed in flight

g. More rugged, less susceptible to damage resulting from handling or system malfunction.

Referring to Fig. 13, the IMU coupling data units would be eliminated. Stabilized axis orientation is accomplished by slewing internal to the computer, (see Fig. 4). The function and mechanization of the optics CDUs remains the same. However, if so desired, one algorithm (matrix) in the computer, driven by optics and gyro angle increments, will give continuous line of sight information in inertial coordinates.

The requirement for total attitude display is identical with that of a gimbal system in that both systems must drive FDAIs. However, the form of the information available differs in the two systems. With a GIMU, attitude information is contained in the nine direction cosines. Unfortunately, the nine direction cosines cannot determine the FDAI gimbal angles unambiguously even though gimbal angles uniquely specify the C-matrix. The reason is simply that three Euler angles together with the signs of the angles represent six "bits" of information. It can be shown that, using the same definition of information bits, the C-matrix contains only five bits. This checks with the observation that, given certain groups of five cosines, the matrix can be completely specified. Thus, one more bit of information is needed. A convenient choice is knowing on which side of gimbal lock the middle gimbal lies. The scheme outlined below determines the Euler angles uniquely if given:

a. three particular cosines (C_{12} , C_{13} and C_{23})

b. the signs of two other direction cosines (C_{11} , C_{33})

c. on which side of gimbal lock the middle gimbal lies.

Suppose the coordinate relations are shown in Fig. 12

$$[C] = \begin{bmatrix} \cos A_{og} \cos A_{mg} & \sin A_{og} \cos A_{mg} & -\sin A_{mg} \\ C_{21} & C_{22} & \cos A_{mg} \sin A_{ig} \\ C_{31} & C_{32} & \cos A_{mg} \cos A_{ig} \end{bmatrix}$$

Fig. 12 FDAI updating.

where only the elements to be used are written out. Then a simple algorithm, (Fig. 14), will yield the correct set of Euler angles. Obviously, if any of the angle conditions are satisfied inherently by a particular mechanization, then the algorithm can be simplified accordingly.

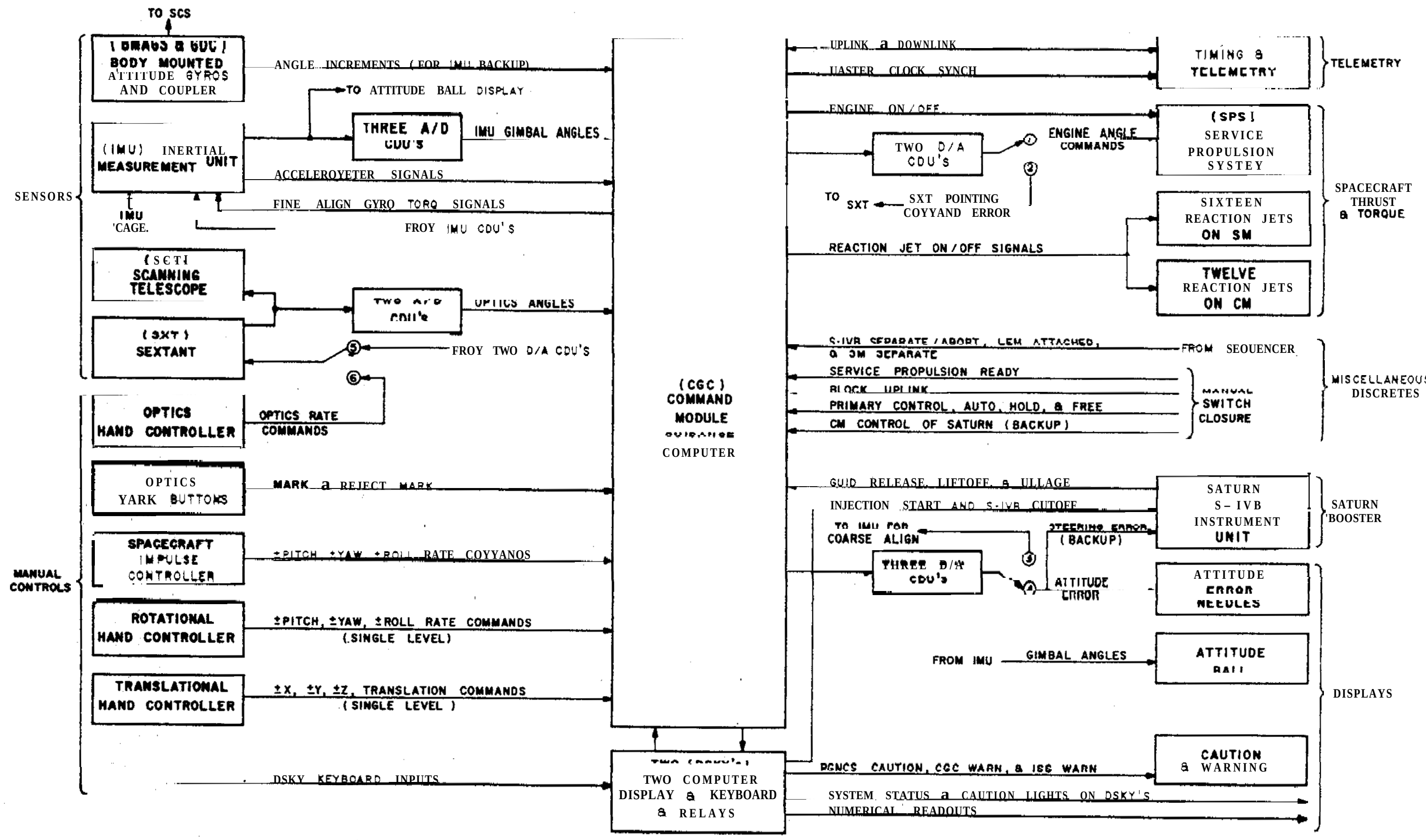
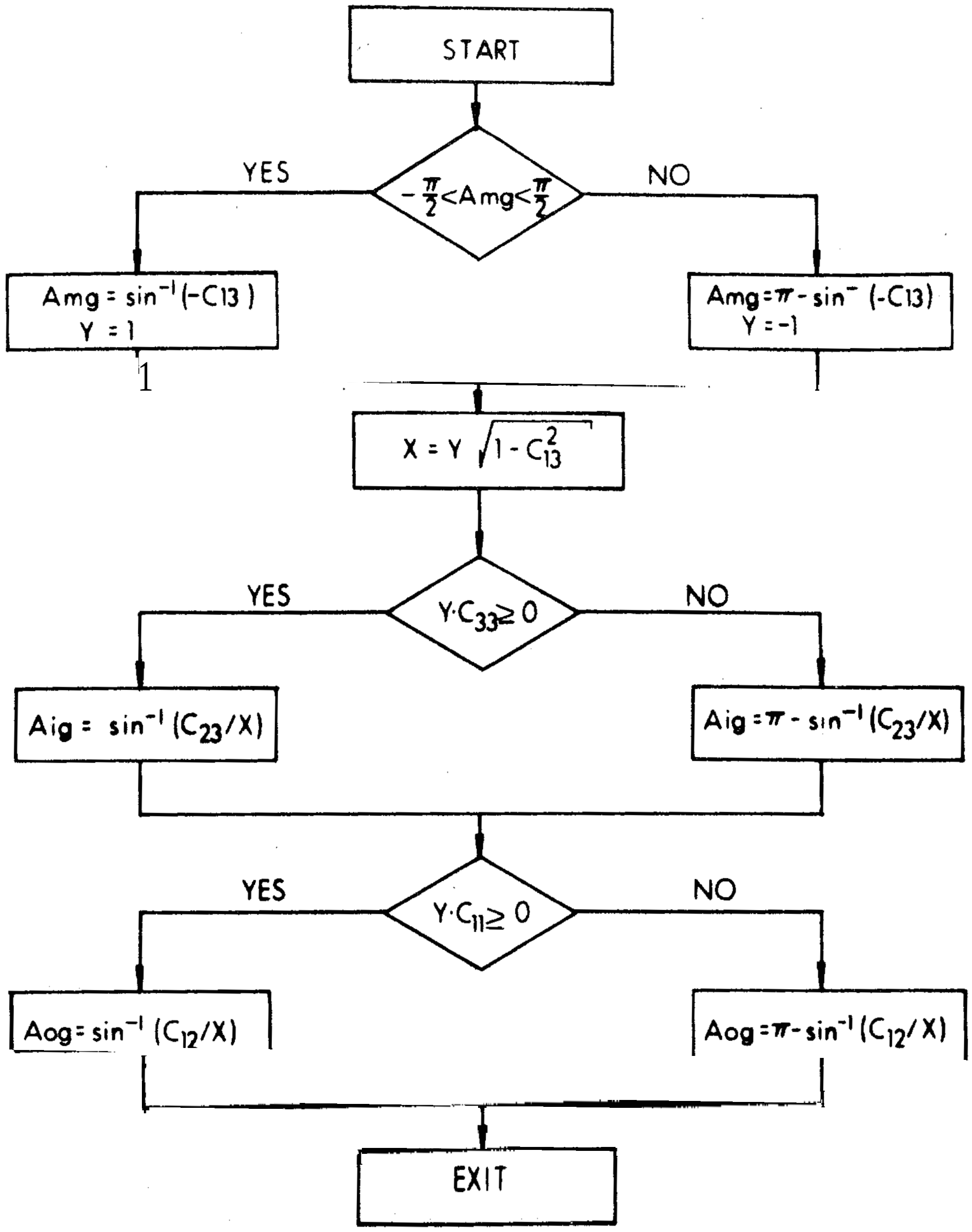


Fig. 13 Guidance, navigation, and control interconnections in command module.



NOTES:

(1) **Amg** $= n \cdot \frac{\pi}{2}$ **CORRESPONDS TO GIMBAL LOCK**

(2).ARCSINES LIE **BETWEEN** $\pm \pi/2$

Fig. 14 Algorithm for FDAI updating.

With a replacement GIMU in order to acquire FDAI gimbal angle, analog signals would require the addition of digital to analog converters.

Spacecraft attitude error resulting from guidance or navigation computations is readily obtained. Attitude or rates generated in stabilized axes are converted to body axes by the use of the transposed cosine matrix, then are counted and displayed. Analog body rate information would be provided by the GIMU. Digital body rate information would be calculated simply in the computer from gyro incremental angle information.

4.2.1 Environmental Differences

The most fundamental difference between the IMU and GIMU lies in the area of mechanical vibration. The IMU is always described as a system of mass, springs, and usually low damping. This yields high peak oscillation at the region of the natural frequency or the first peak resonance frequency. This first resonant frequency is in the region of 100-200 cps. The transmissibility for a vibration model of the IMU is shown in Figs. 15 and 16. The nonlinearity of the structure can be seen from the two transmissibility curves for different input accelerations. The GIMU, on the other hand, is a stiff structure with probably a unity transmissibility out to high frequencies. These sensor assemblies are mounted on navigation bases with low resonant frequencies and low transmissibility at the higher frequencies (greater than 1000 cps.).[/] The ratio of the input vibration may be roughly compared (not knowing the navigation base characteristics) by first examining the input vibration spectrum for the navigation base. These inputs are shown for the Apollo command module and the LEM in Fig. 17 . The ratio of the areas of transmissibility for the IMU to GIMU would be a measure of mean squared g input vibration. Without consideration of the navigation base characteristics, this ratio is 1.3:1. If one considered the navigation base characteristics to have very low transmissibility above 350 cps and unity below 350 cps, (the latter is unrealistic, but realisms would increase the ratio), then the ratio of g rms of IMU to GIMU would be in the neighborhood of 2.2:1. If, indeed, the input vibrations were contained in the higher frequencies, (above 350 cps), and the navigation base transmissibilities were unity, then the input vibration to the inertial components for the GIMU would be much greater than that for the IMU. As is usually the case, unfortunately, the design is complete before the system transmissibilities along with their input vibration spectrums are accurately known. If such information were known early in the design stage, the whole configuration could be more nearly optimized for the vibration environment.

4.3 Interface Changes

Figure 13 is the Apollo command module guidance and navigation block diagram. Interface changes associated with a replacement GIMU are discussed below.

1. IMU - will be replaced by the GIMU. The attitude ball, gimbal angle, fine align, coarse align, and IMU cage connections are eliminated. Gyro signals to the computer are added.

   The present IMU mounting could be used. However, if the navigation base configuration could change, then weight and volume could be saved. Because of the reduced mass, the natural frequency of the GIMU and navigation base would increase. For control considerations, (for rotational vibrations), this would be of some advantage.

2. Computer output to displays - the D/A converters are required. The IMU coarse align signal is not required. The attitude ball (FDAI) drive would be from additional D/A converters receiving information from the computer.

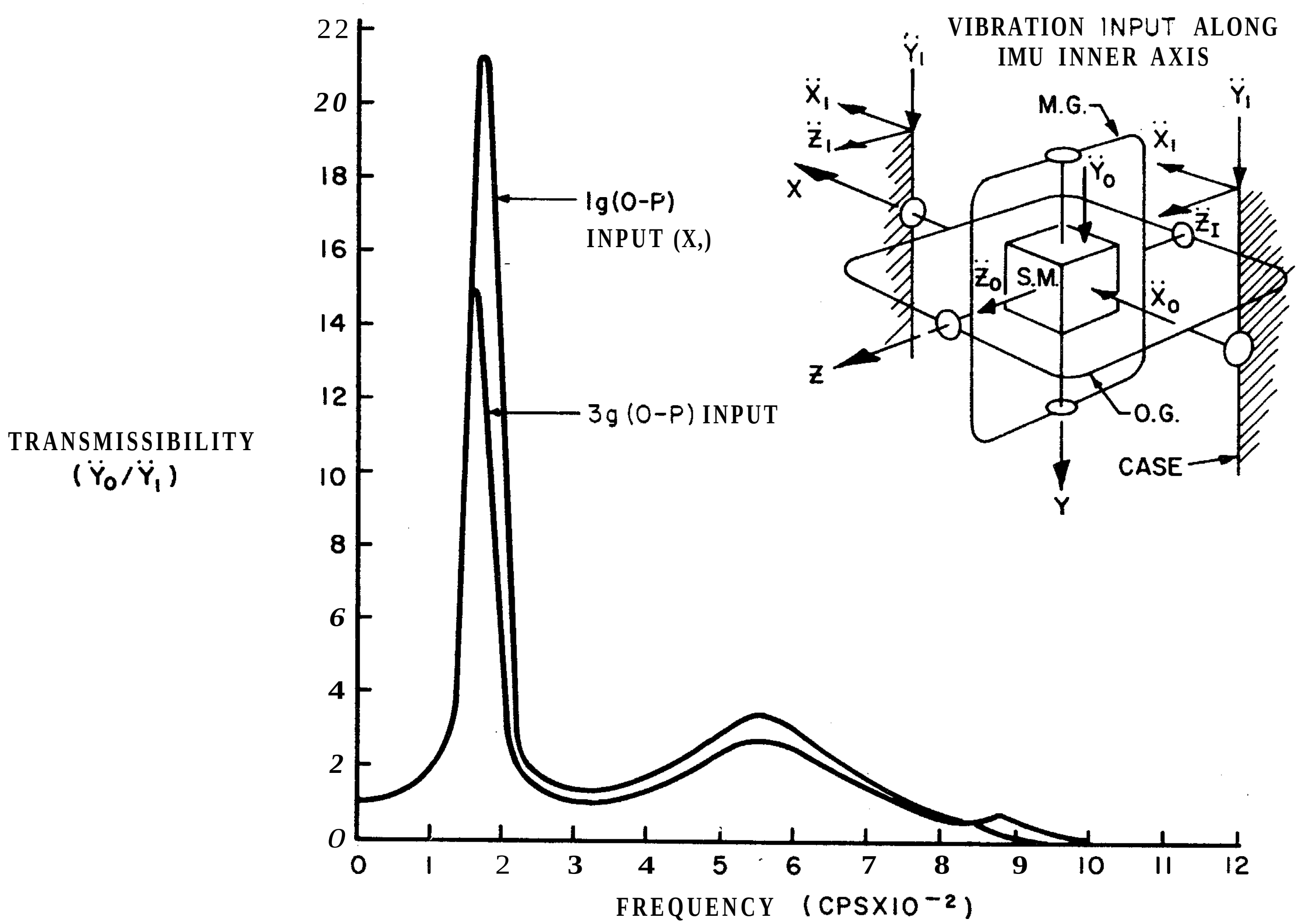


Fig. 15 Apollo IMU - VM (12. 5) vibration test results stable member response curves.

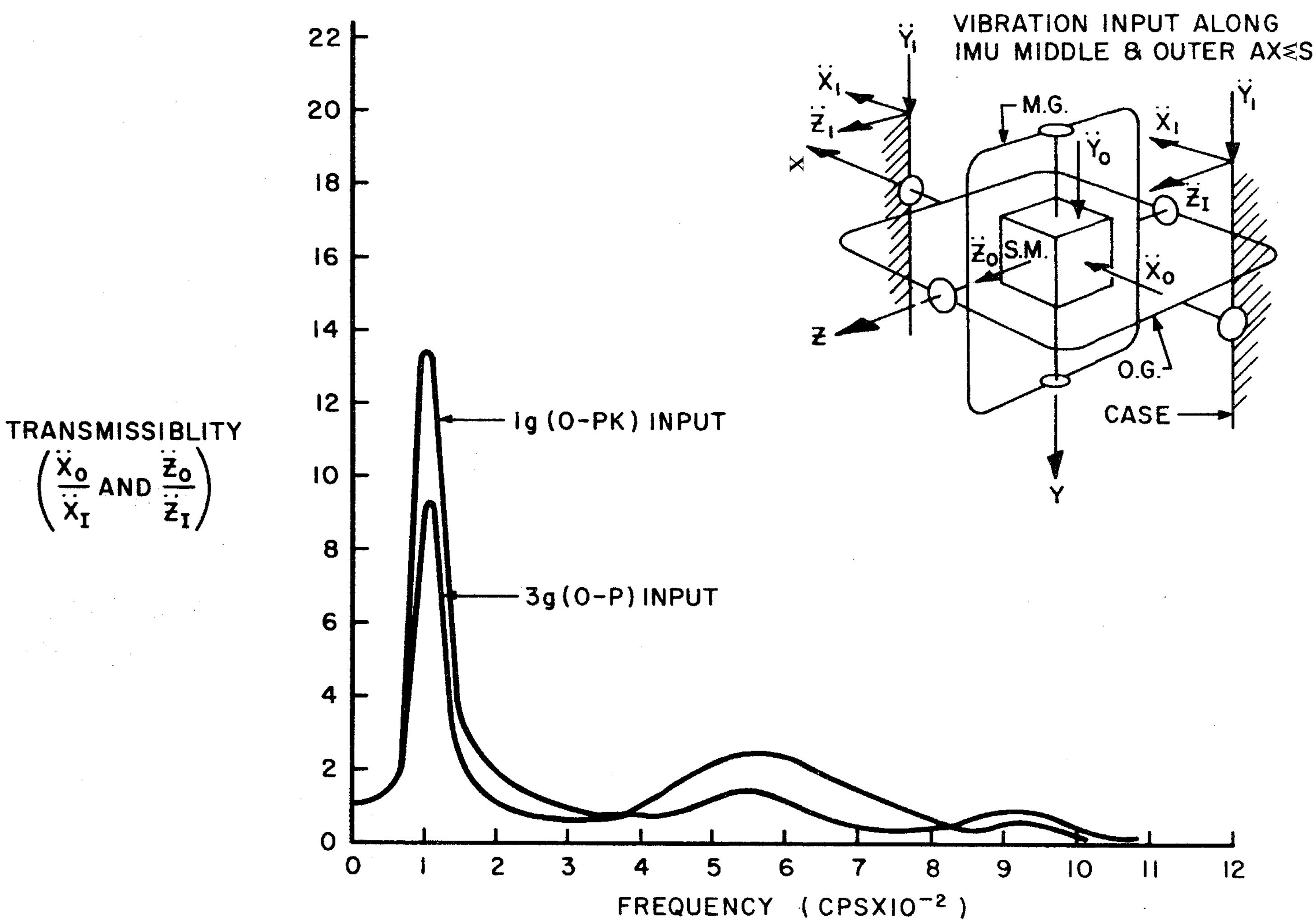


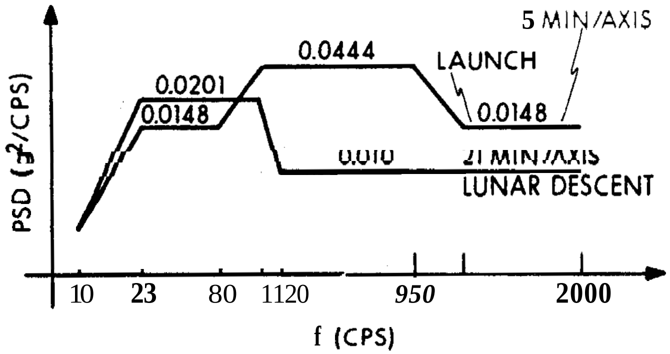
Fig. 16 Apollo IMU - VM (12. 5) vibration test results stable member response curves.

DESIGN EVALUATION TESTS

A. VIBRATION: TYPICAL

(EQUIPMENT MOUNTED TO PRIMARY STRUCTURE

RANDOM

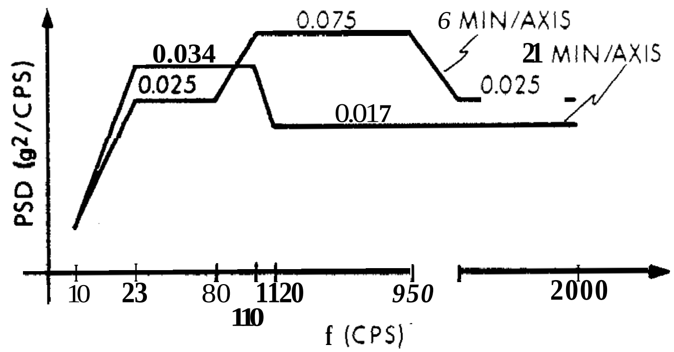


SINUSOIDAL

LAUNCH	5-18.5 CPS	0.154" DA	3 OCT/MIN
	18.5-100 CPS	2.69 g	
LUNAR DECSCENT	5-17 CPS	0.10" DA	$\frac{1}{2}$ OCT/MIN
	17-100 CPS	1.54 g	

A. VIBRATION: APM 1357 IMU 200-2

RANDOM



SINUSOIDAL

5-2000-5 CPS	1g (0-PK) (EA AXIS)	1 OCT/MIN
5-2000-5 CPS	3g (0-PK) (EA AXIS)	1 OCT/MIN

RESULTS : NO PROBLEMS ENCOUNTERED DURING TESTS

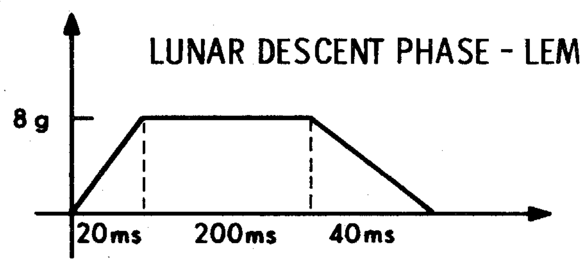
IMU FIRST MODE RESONANT FREQUENCIES AND TRANSMISSIBILITIES:

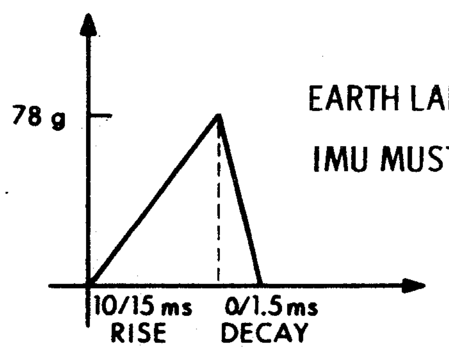
X-AXIS		Y-AXIS		Z-AXIS	
fN	Tr	fN	Tr	fN	Tr
125	8.8	180	10.6	155	10.0

Fig. 17 Apollo IMU mechanical integrity (block 2 system).

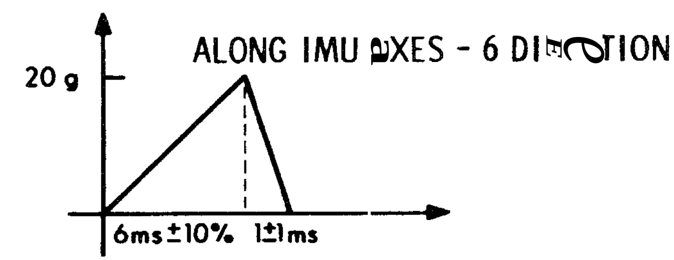
DESIGN EVALUATION TESTS

B. SHOCK: TYPICAL





B. SHOCK: [illegible] 1337 IMU 200-2



6 MS RISE WAS USED FOR TESTS BECAUSE IT WAS CONSIDERED MORE SEVERE

RESULTS: NO PROBLEMS ENCOUNTERED DURING TESTS

IMU STABLE MEMBER SHOCK RESPONSES WERE BETWEEN 38.6 g's AND 46.4 g's

Fig. 18 Apollo IMU mechanical integrity (block 2 system).

CHAPTER V

DESIGN OF AN APOLLO TYPE SYSTEM WITH A GIMU

Strapped-down sensors and the associated transformation computations are more than a replacement for the gimbals. Properly integrated hardware will result in a major reconfiguration of the system. If the decision to use body-fixed sensors is correct, benefits should be observable. This chapter considers a completely new (redesigned) Apollo type guidance, navigation, and control system utilizing a gimballess inertial measurement. The benefits stated in Chapter IV apply as well as those discussed below.

5.1 In-Flight Replacement/Reliability

The in-flight replacement of guidance and navigation equipment as a means of enhancing reliability appears as an unwarranted technique. The space mission considerations will probably always be weight limited as well as space limited. As more and longer space missions of a manned nature become planned and appear as realities, the size and complexity of equipment required to complete space experiments will grow. It is the role of guidance and the guidance system to accomplish these missions with the minimum weight and volume and with the greatest efficiency, Thus, it appears that careful design backed by sound reliability programs conducted on the ground will accomplish the mission. Inertial sensor reliability must be attained equal to that of the semiconductor technology. For the pulse torqued pendulum, this goal is near. The inspection and acceptance techniques are progressing to the point where only "good" inertial instruments are installed in systems. The concept of inertial sensors with no moving parts except fluid connected, as is the case of the pendulum, negates the mechanical wear problem. Indeed, the same philosophy is being carried to the gyroscope by having air bearing wheels. With proper start-up techniques for the system and in-flight operation, the goal of MTBF's for the sensors in the region of 10^6 hours can be realized.

The use of standard building elements for circuit applications as well as a restricted number of different parts allows an economical reliability program for part utilization. This sort of a program, while yielding the maximum reliability, often leaves one with an out-of-date system at flight time. However, this phenomenon is well known to all system designers. It appears as the method for obtaining high reliability designed into the system.

5.2 Primary Voltage Regulation

The power requirements for a GIMU system will vary as a function of primary bus voltage, acceleration, angular velocity, and temperature. The acceleration and angular velocity power requirements are quite straightforward in that restoring torque on the sensors is required only for those inputs. For an overall consideration, the GIMU is essentially a constant power device. As the primary voltage of the spacecraft changes, in order to provide for margin of operation in these conditions, power must be dissipated or thrown away in the GIMU electronics.

As has been the case in the past, each specific circuit has been designed to operate under these extremes of voltage variation. This results in a large number of zener diodes, capacitors and resistors for decoupling and voltage regulation. The primary voltage variations will probably be present. It is likely that the vibration levels will not be exactly known until after the flight. It appears to be desirable to localize the circuitry utilized for contending with the variable input voltage. In one module or circuit is the preregulator and filter for the primary bus. All other circuits operating off this preregulator would be designed to operate with very small voltage changes to minimize voltage regulation power dissipation. This concept will localize voltage problems to one circuit such that as the program develops, and more knowledge of the primary power is obtained, changes to equipment are minimized. Such an approach to reduce total energy consumption for spacecraft is required. This has the effect of allowing the GIMU design to proceed without waiting for exact knowledge of primary power, and as the characteristics of primary power become known, a relatively simple change is possible to take advantage of reducing power. This concept of preregulation will help to have the power dissipated in the circuit components constant, and will result in a more uniform temperature for circuit elements.

5.3 Thermal Design

The other remaining power variability is in the thermal control system for the inertial sensors. This can be examined for our purposes here in three parts. The first part has to do with the power variation as a function of acceleration and angular velocity. As their sensed inputs increase, the power dissipated in the inertial sensors increases, and the required thermal control power decreases. Indeed, the combination for a fixed environment temperature is a constant. Thus, the overall power requirements could be reduced by proper switching between torque generating power and temperature control power.

The second variable in temperature control is the power dissipated in gyro wheels. It has been shown that the gyro wheel power can be reduced by overexcitation of the wheel while running at synchronous speed. A graph of wheel power saving for the overexcited case is shown in Fig. 19.

The overexcitation phenomenon is one of strengthening the magnetic pole in the wheel hysteresis ring. The difficulty with this operation is that, as the wheel oscillates around its synchronous frequency of operation, it tends to weaken the magnetic pole in the hysteresis ring. As a consequence, the power required for operation will, in time, rise to the case that is not overexcited. This phenomenon can be cured by doing the following: reduce the hunting of the wheel by reducing the uncertainty torques required to operate the wheel since the air bearing wheel for gyroscopes has very low ,uncertainty torques, Shown below is a monitor of the power required for an air bearing wheel compared to a good ball bearing wheel for the comparison of the required power variations.

While reducing the power in gyro wheels may be accomplished by overexcitation, the total wheel power variability remains as one of the largest variables in required thermal control power. Temperature control concepts to date have consisted of controlling the external gyro case or end mount temperature. This required control will still be present under the environmental changes. However, to insure more uniform temperatures throughout the gyro and to reduce required dynamic range of operation of the temperature control system, a constant power wheel operation is desired. This will have three effects. First, by having a constant power device coupled with an ability to automatically re-overexcite the wheels, the total power will remain low. Secondly, by having a constant power dissipated in the wheel, it will have uniform and non-varying temperature gradients thereby reducing uncertainty torques in the instrument. Third, it will reduce the required power operation of the temperature control system. By designing an integrated system which accounts for all environmental changes, the total reliability will be increased.

The third variable in the temperature control system is the environmental temperature -- that of the cold plate or heat sink, the structure, and the ambient. Here again, the concept of a preregulator may become useful. While the total power dissipated in the inertial sensor package has become nearly a constant, the disturbances of the environment have not remained constant. It may be possible to preregulate the heat sink temperature as a function of structure, ambient coolant temperature, and flow rate which would minimize the overall power required.

5.4 Redundant Sensors

A desire to insure a high probability of the availability of a working attitude reference system suggests the use of redundant sensors. The implementation of redundancy may take two paths:

1. The addition of optical or radiation sensors working in parallel with the GIMU
2. The addition of redundant gyros to the GIMU assembly itself

The first approach has value in providing a backup attitude reference or in minimizing the activated time of the gyros. The second approach provides two variations. If all the gyros are operated in parallel, then protection for catastrophic failure is provided. In addition, with a suitable number of units, (seven), in-flight checking of gyro performance may be accomplished. This discussion presumes that simultaneous operation of the redundant gyros will result in all units wearing out at the same time. If only a minimum number of gyros are operated, the equivalence of automatic in-flight replacement is acquired without requiring access to the GIMU assembly.

5.5 Stabilized Line of Sight

The present Apollo navigation measurements taken with the scanning telescope and sextant require aiming and stabilization of the spacecraft with respect to an inertial or local vertical frame. See Figs. 20 and 21 as an illustration. However, measurements are always made with some finite spacecraft angular velocity. The availability of precise body rotation information would permit isolation from base motion for the optics. As an example, referring to Figs. 22 and 23 with a gyro (GIMU) input axis parallel to the optics principal axis, one degree of isolation is provided. The optics shaft angle encoder would then provide the sine and cosine information necessary to convert information from the other two body-fixed gyros into optics trunnion axis isolation drive signals.

5.6 Astronaut Displays

In the Apollo guidance, navigation, and control system, numerous displays are provided for the astronaut to permit him to either execute his functions in the mission, or to monitor the state and performance of the spacecraft. With the exception of the computer display keyboard, (DSKY), information is presented by devices driven by analog signals.

With the introduction of a gimballess inertial measurement unit, (GIMU), the guidance, navigation, and control system becomes all digital in its internal mechanization. Thus, it is desirable to have the associated displays either digital in nature, or at least digitally driven.

Analog displays usually result in one meter for each item of information being monitored. The instrument panel becomes both crowded and cluttered with the astronaut requiring intensive training to insure against "pilot error" when executing complex spacecraft operations. Most of the display information would be supplied by the spacecraft computer. To do this, real time display programs

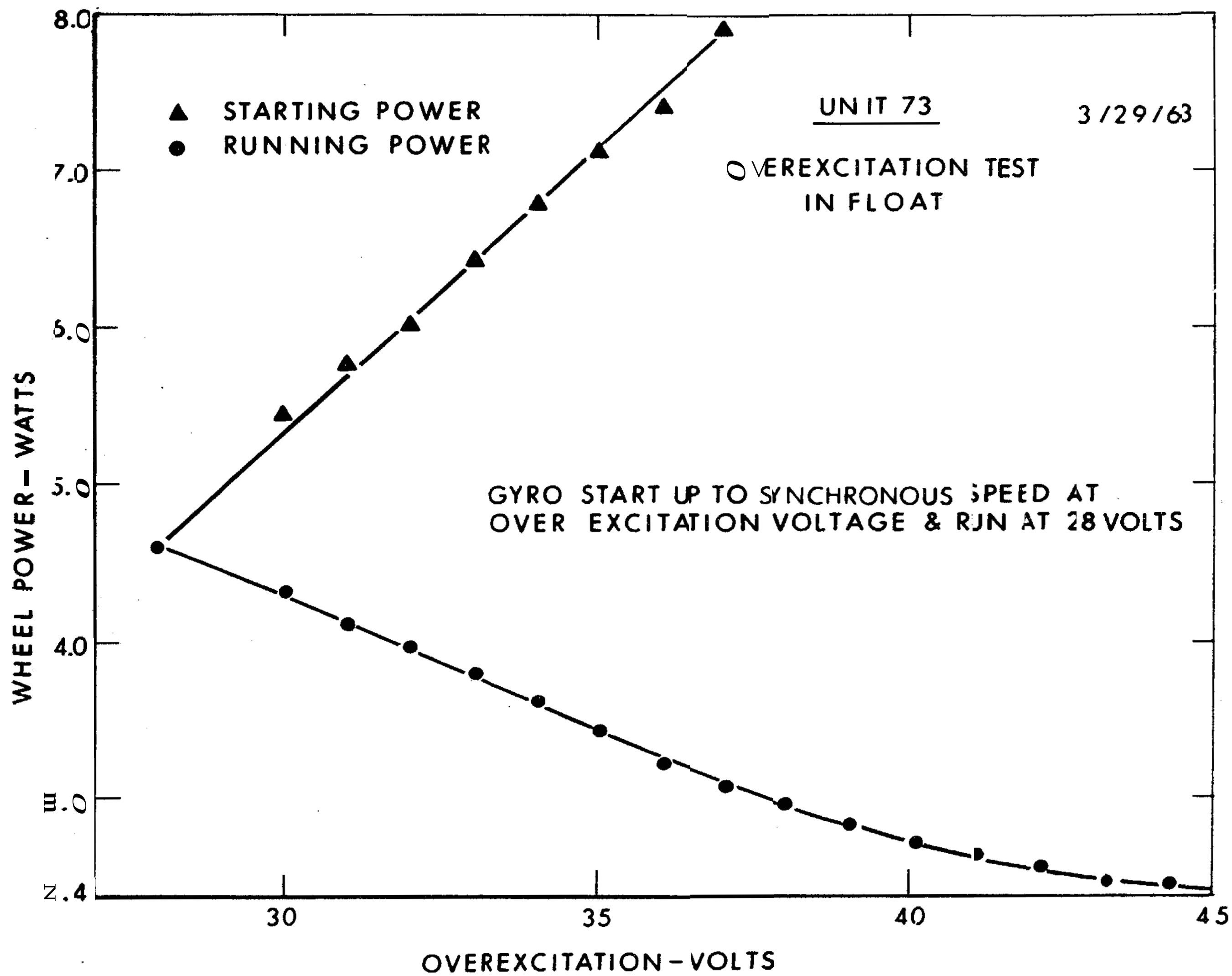


Fig. 19 Wheel power vs excitation voltage.

must be continuously executed -- one for each item of information whether it is immediately required or not. An improvement would be to have luminous electronic displays whose information would be selected by the computer as a function of the spacecraft operational mode. Such would place less reliance upon the astronaut's memory and focus attention upon importance events (information). In parallel, the computer would still execute a watch function with the "watchdog" displays illuminated by either a scanning or alarm program.

An all attitude system requires an all attitude display, something which the present electromechanical attitude ball (FDAI) cannot provide. Two apparent solutions are the following. First, design a mechanical ball driven along true body axes by gyro information. This device would be hard to equip with angle feedback information. In addition, it would probably belong to the class of low reliability components that would be undesirable €or long duration missions. Secondly, design an electronic display such as a colored cathode ray oscillograph which would portray in three dimensions spacecraft attitude with respect to a graphical planet, space, or other reference, such being a function of the .mission mode.

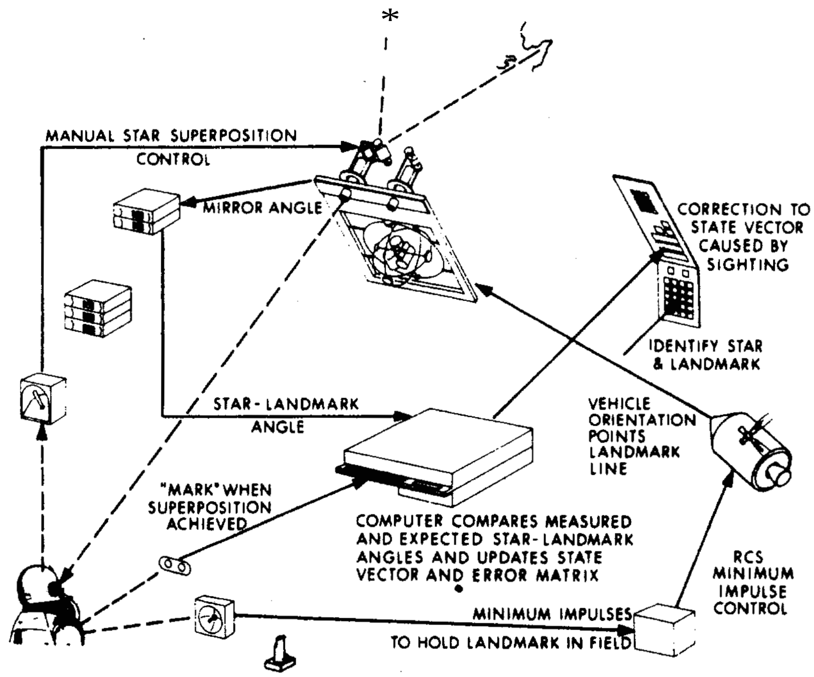


Fig. 20 Midcourse navigation -- manual star -- landmark measurement,

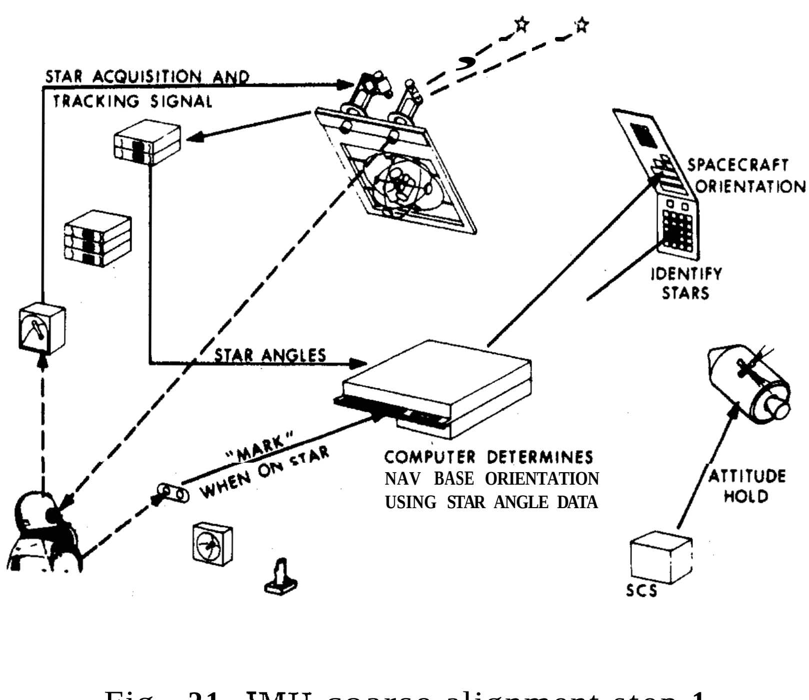


Fig. 21 IMU coarse alignment step 1.

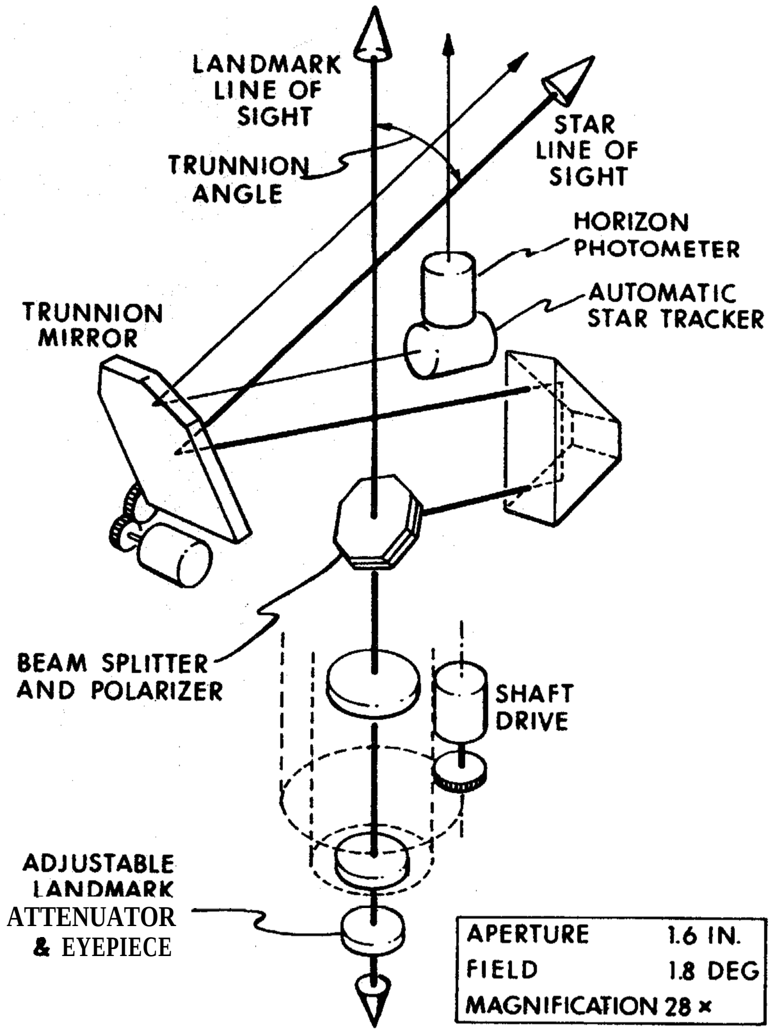


Fig. 22 Sextant schematic.

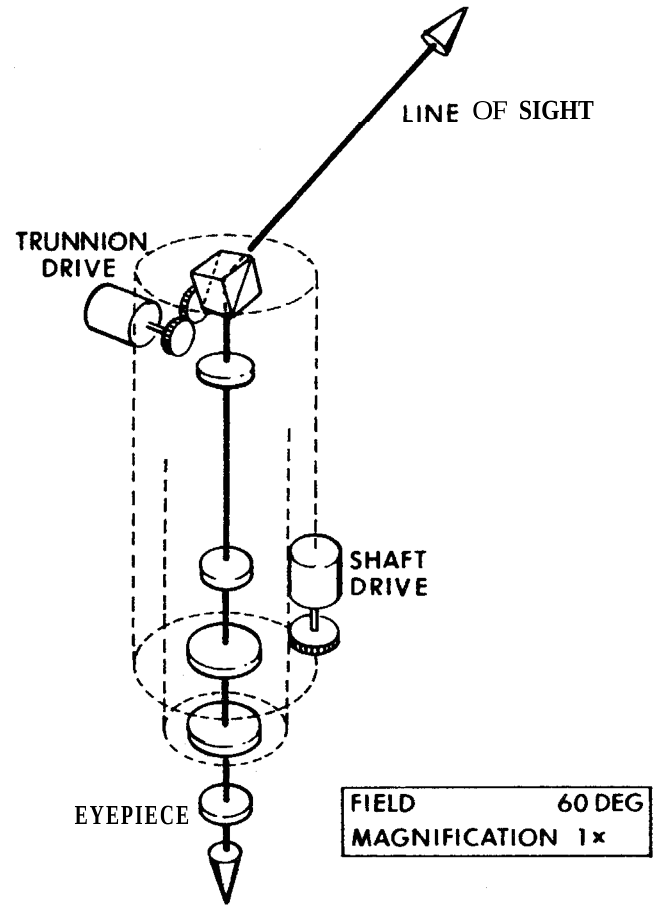


Fig. 23 Scanning telescope schematic.

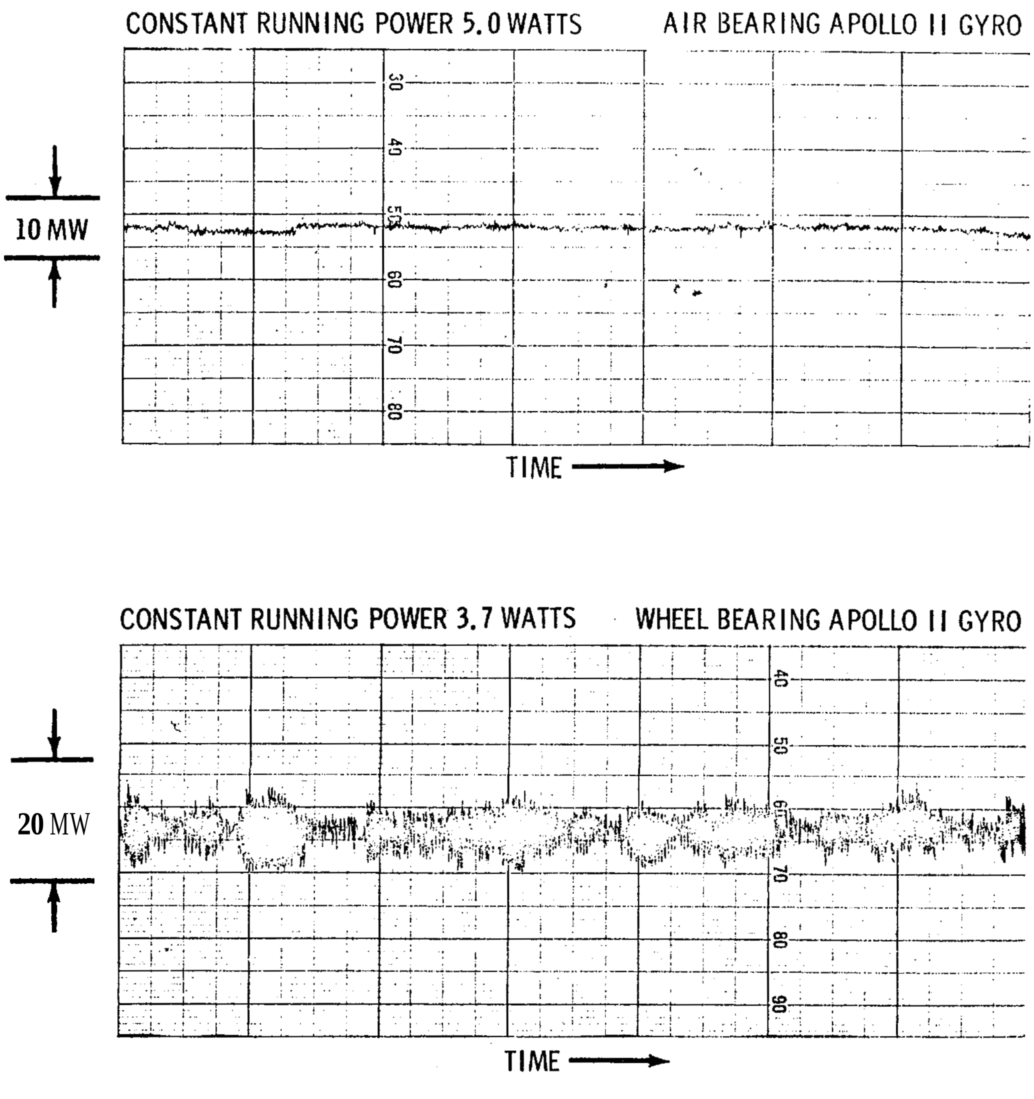


Fig. 24 Apollo gyro power deviations.

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2. Bumstead, Robert M., and Vander Velde, Wallace E., Navigation and Guidance Systems Employing a Gimballess IMU, MIT Instrumentation Laboratory Report R-412 Rev A, October 1963.