NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Memorandum 33-662

Mariner Mars 1971 Inertial Reference Unit

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PREFACE

The work described in this report was performed by the Guidance and Control Division of the Jet Propulsion Laboratory.

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ABSTRACT

This report presents the design concept of an Inertial Reference Unit using the gyroscope in a rate mode in conjunction with a unique application of a drift-compensated and temperature-stabilized solid-state integrator for the inertial mode. A typical test program and results for an Inertial Reference Unit as applied to the Mariner Mars 1971 program to achieve highly reliable spacecraft operation are also described.

I. INTRODUCTION

Historically, the JPL unmanned spacecraft carrying the scientific instrumentation necessary in deep space exploration requires that the spacecraft attitude be maintained in a specific orientation. The attitude is controllable by on-board sensing or ground commands, and is controlled within finite limits within a coordinate reference system. This was done during the cruise mode of the Mariner Mars 1971 mission by means of Sun sensors to control the X-Y plane of the spacecraft relative to a Sun line (X-Y plane perpendicular to the Sun line), and a sensor to detect Canopus (selected for its known position relative to the Earth-Sun ecliptic plane) as the additional reference about the spacecraft roll or Z-axis.

During all other modes whereby the spacecraft is required to be positioned off of the Sun-star reference, such as the mode for making changes to the velocity of the spacecraft (ΔV) to correct to a desired trajectory, it is necessary to resort to another frame of reference other than the celestial references. The selected reference is referred to as an "inertial reference." The use of this reference system requires the use of gyroscopes. The design of an Inertial Reference Unit (IRU) for the 1971 mission was based upon the "inertial reference" need.

II. PHYSICAL DESCRIPTION

The Mariner Mars 1971 (MM'71) IRU was comprised of two subassemblies: an Inertial Sensors Subassembly (ISS) and an Inertial Electronics Subassembly (IES), and functioned as part of the MM'71 Attitude Control Subsystem (ACS) to provide a 3-axis inertial reference for the spacecraft. The ISS assembly contained three gyros, one accelerometer, and one printed wiring circuit board. The chassis was produced from a highly dimensional stable ZK 60 magnesium forging. The size of the assembly was 9.40 × 16.76 × 17.63 cm (3.7 × 6.6 × 6.94 in.). The IES assembly contained the power conversion circuit board, the integrator circuit board, and the loops and drivers circuit board that contained all other circuitry which will be described later. The chassis was produced from ZK 30 magnesium tool plate. The size of the IES assembly was 5.08 × 16.76 × 38.10 cm (2 × 6.6 × 15 in.).

Each subassembly ISS and IES weighed approximately 2.27 kg (5 lb) each. The weights grew slightly from the production prototype (2.13 and 1.95 kg (4.7 and 4.3 lb) for the ISS and IES, respectively) to 2.22 and 2.27 kg (4.9 and 5.0 lb) for the flight units. This was due to a protective coating added to all circuit boards and wire harness and terminal connections for the flight units which were elected to not be used on the production prototype model.

III. ELECTRICAL DESCRIPTION

The IRU required 400 Hz, 3 phase, 27.2 V for the gyro spin motors. This power was a quasi (stepped square wave) sine wave for the "Y" configuration of the spin motors. The electronics required 2.4 kHz, 50 V square wave as an input. This power was converted to 6.3 V for the gyro primary excitations (using a filter for conversion to sine wave), and the accelerometer digital clock circuitry. Additional conversion was required for precision ±20 Vdc, ±15 Vdc, and ±20 Vdc unregulated power for relays. The following was the typical power measured in the modes of IRU operation at 2.4 kHz:

Launch mode	13 W
Roll only mode	6.5 W
Roll only inertial	7.4 W
Pitch/yaw/roll inertial	14.7 W
Yaw/roll inertial commanded turn	16.6 W

A significant portion of this power was required for relay activation. There was a total of 18 relays associated with the above modes. For example, a full IRU and a full inertial mode while performing a yaw or roll turn required the activation of 14 relays. These relays were configured in pairs of series or parallel arrangements. The nominal wattage for a pair of relays was 0.91 W. When in a commanded turn mode of operation, the relays were dissipating 6.4 W out of a total of 16.6 W. An additional significant power drain was associated with the heaters on the operational amplifiers, Fairchild μA726, used as an active integrator in each gyro axis circuitry. This required the ±15 Vdc regulated power. The heaters were part of the μA726 chip configuration and stabilized the chip temperature at approximately 383.15 K (230°F). This allowed the integrator to operate over a wide external temperature range to minimize drift. There was a total of six μA726 amplifiers, and the wattage dissipation per chip was approximately 0.42 W. Total integrator power including circuit and relay power was 4.9 W.

IV. FUNCTIONAL DESCRIPTION

As seen by Figs. 1 and 2, the IRU circuits were designed to provide rate damping during all celestial sensor acquisitions, commanded turn and motor burns, position information during the inertial mode (motor burns and commanded turns), and pulse outputs to the Central Computer and Sequencer (CC&S) from the accelerometer digital circuitry during the velocity correction (ΔV) phases (motor burns).

The gyro rate loop (see Fig. 1) consisted of a pre-amplifier, demodulator, and current boost amplifier. Rate sensing was provided at the current boost amplifier output for the autopilot electronics during motor burns. Rate sensing during acquisition and inertial modes was provided at a point in the rate loop between the torquer and R's for the switching amplifier electronics cold gas jet valve system. The inertial mode required switching the integrator from a low-gain amplifier (1:120) to the integrate mode, by command from the attitude control electronics (ACE). Position information was then supplied to both the (A/P) and (S/A) switching amplifiers. By command from ACE, ±20 Vdc was voltage-divided and fed to the input to the integrator for all commanded turns (650 deg/h nominally). The accelerometer circuitry (see Fig. 2) provided analog rebalance during the launch phase and a digital loop which captured the accelerometer during motor burn phases and provided a calibrated 0.03 m/s per pulse to the CC&S for velocity correction information.

V. INERTIAL SENSORS DESCRIPTION

The gyros used for the MM'71 program were the same type as those used in the Mariner 1962, 1964, 1967, and 1969 programs. These were Kearfott C702565007 rate integrating gyros. The gyros were always used with a capture loop in a rate mode (see Fig. 1). The accelerometer was a Kearfott C702401-027 miniature single-axis flexure-suspended type.

The following information pertinent to the inertial sensors is listed:

Gyro type	Single-degree-of-freedom, rate-integrating
Gyro weight	0.318 kg (0.7 lb)
Operating temperature	319.26 K (115°F)
Gyro size	4.67 cm diam × 7.02 cm (1.837 in. diam × 2.765 in.)
Gyro gain	235
Gimbal moment of inertia	103 gram·cm^2 (OA)
Spin motor design	Hysteresis synchronous "Y" wound, 400 Hz, 27.2 V, quasi-sine wave
Maximum running power	3.2 W at 26 V rms sine wave running, 0.57 power factor
Angular momentum	227 000 gram·cm^2/s at 24 000 rpm
Pickoff scale factor	159 mV/deg
Command rate scale factor (K_T)	280 deg/(h·mA) for pitch and yaw, 135 deg/(h·mA) for roll

Accelerometer type	Miniature single-axis, flexure-suspended type
Size	2.90 × 3.24 × 5.14 cm (1.14 × 1.275 × 2.025 in.)
Weight	0.15 kg (0.33 lb)
Operating temperature	297.03 ± 13.89 K (75 ± 25°F)
Pendulum freedom	±2 mrad
Flexure spring rate	>2.0 g/rad
Threshold	$>10^{-6}$ g
Damping	$145\,000 \times 10^{-5}$ N·cm/(rad·s) (145 000 dyne·cm/(rad·s))
Pendulosity	$1\,100 \times 10^{-5}$ N·cm/g (1 100 dyne·cm/g)
Pendulum moment of inertia	14.0×10^{-5} N·cm/s^2 (14.0 dyne·cm/s^2)
Pickoff scale factor	10 Vac/rad
Torquer scale factor	1 mA/g
Bias	±300 μg maximum after environmental exposure
Power	Negligible, >0.1 W

VI. GYRO MECHANIZATION

The gyros (see Fig. 1) were always in a rate mode using analog feedback to their dc permanent magnet torquers. The pickoff output was the first-stage operational pre-amplifier with a gain of 6 in the pitch and yaw channels and a gain of 12 in the roll channel. These amplifiers were located in the ISS chassis. The pre-amplifier outputs were fed to a transistorized demodulator (using 6.3-V, 2.4-kHz square wave as the switching frequency) and an amplifier with a gain of 20. The output was fed into a current boost amplifier (voltage gain of unity). This was necessary since the operational amplifier stage of the demodulator could only deliver 25 mA maximum. The gyro loops were designed to have a capture capability of 2.5 deg/s in the pitch and yaw channels and 1.25 deg/s in the roll axis or approximately 33 mA of capture current. The output of the current boost amplifier was fed back to the gyro torquer with a nominal dc resistance of 266 ohms in series with a torquer low resistor selected to have a resistance equal to the measured gyro torquer scale factor, nominally, 280 deg/(h·mA) or 280 ohms. This was done to provide a scaling of 1 mV/(deg·h) at the input to the integrator and the rate amplifier.

VII. GYRO INTEGRATOR

The integrator circuitry was a unique conceptional design change from the previous Mariner missions. The previous design utilized large tantalum capacitors (1 020 μF each) to obtain a wide-angle storage capability necessary when using low damped gyros. The design of the integrator circuit consisted of (see Fig. 3) one Fairchild dual amplifier (single chip) μA726 used as a voltage follower. To obtain an overall open loop large gain (greater than 1×10^6) necessary for good integrator design, the output of the voltage follower was followed by an additional LM101 operational amplifier. The feedback was a low leakage, polycarbonate, 10-μF capacitor. To obtain a dc polarity inversion with respect to the input to the integrator, an additional unity-gain LM101 amplifier was added. The position gain of the integrator was 3 V/deg and had a maximum storage angle of ±6 deg. To obtain the commanded turn of nominally 650 deg/h, a precision ±20-Vdc voltage on command was applied at the voltage divider network into the inverting input to the integrator for the yaw and roll channels only. Also, as seen in Fig. 3, a resistive network was added at the noninverting input for drift compensation. This network with a capability of applying a plus or minus dc voltage consisted of a coarse and a fine resistor, selected at final IRU test, for biasing gyro and integrator drift to less than ±0.08 deg/h. Since this portion of the design compensated for the fixed drift, the temperature drift sensitivity of the integrator was compensated by the built-in heater circuitry of the μA726 as previously referred to.

VIII. ACCELEROMETER MECHANIZATION

The accelerometer pickoff output was (see Fig. 2) amplified through a two-stage operational amplifier located in the ISS. All operational amplifiers, both in the gyro and accelerometer circuitry, were the conventional LM101AH, produced by National Semiconductor. The overall gain of these stages was 560. The amplified output was ac-coupled to a demodulator and dc amplifier of the same design as the gyro loops. The overall gain of the accelerometer loop was 10 400.

The output of the demodulator was coupled to an analog circuit and a digital circuit. For the launch mode, the dc output was fed through a launch mode relay through a 1 000-ohm resistor (for ground tests) to the accelerometer torquer, forming a rebalance loop with a ±18-g capture capability. During the motor burns the rebalance mode of operation was performed by a digital circuit. The output of the demodulator-dc amplifier stage was compared to a preset voltage at the input to an operational amplifier. The loop was unidirectional and captured only in a plus g sensing. The output of the comparator gated the output stage of a four-stage digital counter consisting of Signetics SD124 logic circuits. The input from a 2.4-kHz reference was divided by 16 through the divider chain which produced a maximum output of 150 pulses/s. The nominal digital scale factor was 0.03 m/s per pulse or 326.543 33... pulses/s per g. Since the output pulse rate was limited to 150 pulses/s, the capture capability of the digital circuit was limited to approximately 0.45 g. The output of the digital clock was also gated to the CC&S for totalizing the required pulses (ΔV) during motor burns. The scale factor of 0.03 m/s per pulse was established by the use of a select resistor during final assembly under a known g orientation.

IX. TEST PROGRAM

There was a total of 15 test specifications including two procurement specifications (gyro and accelerometer) which assured that the IRU design was qualified to meet MM'71 mission requirements. The test and build cycle, in order of occurrence, was as follows:

(1) Upon receipt of inertial sensors at JPL from the manufacturer, receiving tests were performed to verify manufacturer's compliance to the procurement specifications, such as gyro fixed drift, "g" sensitive drift, motor performance, and accelerometer bias and scale factor. This data served as the initial

starting point for evaluating future test results for predicting degradation and performance during the mission.

(2) The sensors were then sent to the selected system contractor who in turn was required to verify the performance requirements previously obtained.

(3) Each printed circuit board was tested at the contractor in accordance with their individual test specifications for determining amplifier gains, dc voltage offsets, frequency response, saturation, power supply voltage regulation, relay operation, and integrator performance at ambient temperatures. Each board was then assembled into its respective chassis and harness and interconnect wiring assemblies.

(4) The IES assembly was then tested at ambient, high, and low temperatures (263.15 to 333.15 K (14 to 140°F)) for essentially the same parameters that were tested at the circuit board level.

(5) The ISS assembly (minus inertial components) was also tested at ambient, high, and low temperatures to demonstrate performance as part of a margin test program.

(6) The gyros and accelerometer were then assembled and aligned into the ISS chassis; all gyro and accelerometer performance tests were repeated. This again was another check in time against the original receiving data.

(7) The IES and ISS assemblies were then mated into a system configuration and tested for gyro noise, nulls, power turn-on, and steady-state currents. Commanded turn calibration curves were plotted for 302.59, 310.93, 319.26, and 327.59 K (85, 100, 115, and 130°F). Relay operational checks, integrator scale factor determination, rate scale factor determination, and final resistor selection in the accelerometer digital rebalance circuitry for 0.03 m/s per pulse scaling were performed. Also a selected final resistor was installed for biasing gyro and integrator drift to less than ±0.08 deg/h. Curves were generated for accelerometer digital scale factor at 291.48, 299.82, 308.15, and 316.48 K (65, 80, 95, and 110°F).

(8) Upon completion of the previous test, the IRU was then environmentally exposed to sine vibration, random vibration, and thermal vacuum.

(9) Upon completion of the environmental tests, the IRU was retested essentially to all subassembly tests performed during the build cycle to ascertain the integrity of the design after stringent imposed environments to assure mission success.

(10) The IRU was delivered to JPL, and JPL personnel performed acceptance testing to verify that the contractor met all of the design criteria and test requirements as specified.

(11) The IRU was then installed into the spacecraft and exposed to extensive system-level tests to verify its compatibility with the overall systems design.

(12) Upon completion of these tests, the IRU was removed from the spacecraft system tests and retested in the inertial sensors laboratory in preparation for the Eastern Test Range (ETR) (Canaveral, Florida) prelaunch spacecraft readiness tests. The retest of the IRU was to verify that the electronic circuitry, or critical inertial sensor performance parameters, had not changed significantly to affect required mission performance. These test results were compared with earlier tests to check trends in performance that may have indicated degradation (poor mission performance) or degradation resulting in eventual failure and thus failure of the mission.

(13) The IRU was then transported to the ETR test site where duplicate test equipment had been installed for spot checking the JPL laboratory pretest results and was then delivered to the spacecraft final assembly area for prelaunch readiness and prelaunch tests.

X. TEST HISTORY RESULTS, CRITICAL PARAMETERS

The adequacy of the IRU design for the MM'71 mission was demonstrated by the extremely successful performance of the IRU in meeting the required test specification requirements, and the demonstrated successful mission performance of the MM'71 mission. Minor redesign was necessary during the build of the IRU to correct: (1) high-frequency amplifier oscillations at about 500 kHz (the LM101 was susceptible to oscillations if not properly decoupled from the ±20-Vdc supplies), (2) erratic and out-of-specification drift due to rectification of high-frequency noise at the input to the integrator circuitry (corrected by filtering at the inverting input to the integrator), and (3) interaction of the accelerometer excitation oscillator frequency with the basic gyro reference frequency of 2.4 kHz. This required detuning the oscillator to prevent third-harmonic beat frequencies that were interacting with the gyro electronics.

The three parameters important to meet mission accuracy requirements were: combined gyro and integrator drift, commanded turn accuracy, and accelerometer scale factor accuracy. Table 1 lists the gyro fixed drift (D_F) as measured over a period of approximately one year and was derived from single-gyro eight-position heading tests.

The average long-term drift change compared against vendor data for 12 gyros was 0.085 deg/h. The most stable gyros were selected for the proof test model (PTM) and flight units, and the long-term drift value was ascertained to be 0.076 deg/h. Table 2 lists the combined gyro and integrator drift compared against the initial trim value for short term and long term.

The short-term average change from the initial trim was 0.026 deg/h, and the long-term value was 0.036 deg/h. It appears that the integrator and gyro drift is more repeatable than that portrayed by the single-gyro heading data. The data indicates that the dynamic test similar to the spacecraft operation is more accurate than the static single-gyro heading test technique. These values reflected absolute drift to be well within the mission requirements of 0.15 deg/h, 1 σ.

The commanded turn repeatability was also carefully monitored over a one-year period. Curves for the exact value are supplied prior to launch for turns at gyro temperatures of 302.59, 310.93, 319.26, and 327.59 K (85, 100, 115, and 130°F). The average values of the change from the initial turn command calibrations were 0.33 deg/h for a plus turn and 0.29 deg/h for a minus turn. A peak value noted was for the PTM unit. This value was 0.89 deg/h in the "roll" channel. The mission requirement levied against the curve data was 1.5 deg/h uncertainty, 3 σ.

The digital scale factor (DSF) for the accelerometer was also carefully tracked over the test period prior to launch. Calibration curves for 291.48, 299.82, 308.15, and 316.48 K (65, 80, 95, and 110°F) were also supplied for use during the mission. For a period of approximately 160 days, the accelerometer analog scale factor was noted to be changing (increasing) at a rate of from 2 to 3 parts per million per day for the PTM, Flight 1, and Flight 2 IRUs. The change in scale factor is attributable to aging of the torquer magnets in the accelerometer. A peak value of change to the DSF during a 160-day period was 0.035%. Mission requirements were 0.1% 3-σ uncertainty. Since the known rate of aging effects on the DSF was predictable with time, the value of 0.1% uncertainty was not considered unreasonable.

XI. TEST EQUIPMENT

The test equipment used in determining the performance of digital components and the IRU subsystem had to be sufficiently accurate to prevent instrumentation errors from seriously affecting performance measurements. No attempt will be made in this report to describe the error contribution values due to test stand alignments relative to north and vertical or electronic readout resolution errors. However, as an example, the accelerometer scale factor changed from day to day because permanent magnet aging was known to be on the order of 1 to 2 parts per million per day. The precision Tinsley potentiometer used in determining the accelerometer analog performance values was accurate in resolving to as little as 1 part per million.

Table 1. Gyro fixed drift (degrees/hour)

Model	Axis	Gyro serial number	Vendor	JPL	GE	GE	JPL	After environmental tests	Pre-ETR	ETR	Max Δ
Production prototype	Pitch	202	+0.064	-0.018	-0.074	-0.011	+0.06	---	---	---	0.138
	Yaw	115	-0.138	-0.112	-0.149	-0.145	-0.022	---	---	---	0.116
	Roll	102	+0.028	+0.104	-0.010	+0.005	+0.04	---	---	---	0.076
Proof test	Pitch	201	-0.033	-0.088	-0.095	-0.061	---	-0.0009	-0.072	-0.0675	0.062
	Yaw	204	+0.063	+0.013	+0.012	-0.039	---	-0.0953	-0.1118	-0.1168	0.18
	Roll	103	-0.110	-0.1124	-0.109	-0.097	---	-0.0372	-0.0668	-0.0645	0.073
Flight 1	Pitch	203	-0.054	+0.037	-0.056	-0.032	---	+0.0009	+0.0066	-0.0056	0.091
	Yaw	205	-0.120	-0.044	-0.07	-0.05	---	-0.0945	-0.1068	-0.0567	0.076
	Roll	122	+0.072	+0.057	+0.042	+0.032	---	+0.046	+0.1067	+0.0222	0.050
Flight 2	Pitch	208	-0.039	-0.03	-0.031	-0.004	---	-0.0265	-0.0106	-0.0393	0.035
	Yaw	207	+0.004	-0.028	-0.056	-0.004	---	-0.0227	-0.0094	-0.0241	0.06
	Roll	124	-0.06	-0.071	-0.06	-0.066	---	-0.1154	-0.1210	-0.1095	0.06

Table 2. Combined gyro and integrator drift

Model	Axis	Initial trim value, deg/h	3-day stability max Δ (short term)	Pre-ETR (long term), deg/h	ETR (long term), deg/h	Max Δ, deg/h
Proof test	Pitch	+0.0059	0.002	---	+0.04	0.034
	Yaw	+0.0133	0.05	---	+0.079	0.065
	Roll	+0.002	0.016	---	+0.009	0.016
Flight 1	Pitch	-0.006	0.04	+0.041	+0.046	0.04
	Yaw	-0.02	0.021	-0.025	+0.006	0.026
	Roll	+0.023	0.039	+0.009	-0.008	0.039
Flight 2	Pitch	+0.023	0.017	-0.006	+0.021	0.017
	Yaw	-0.043	0.027	-0.089	-0.119	0.076
	Roll	+0.0004	0.002	+0.007	-0.007	0.015

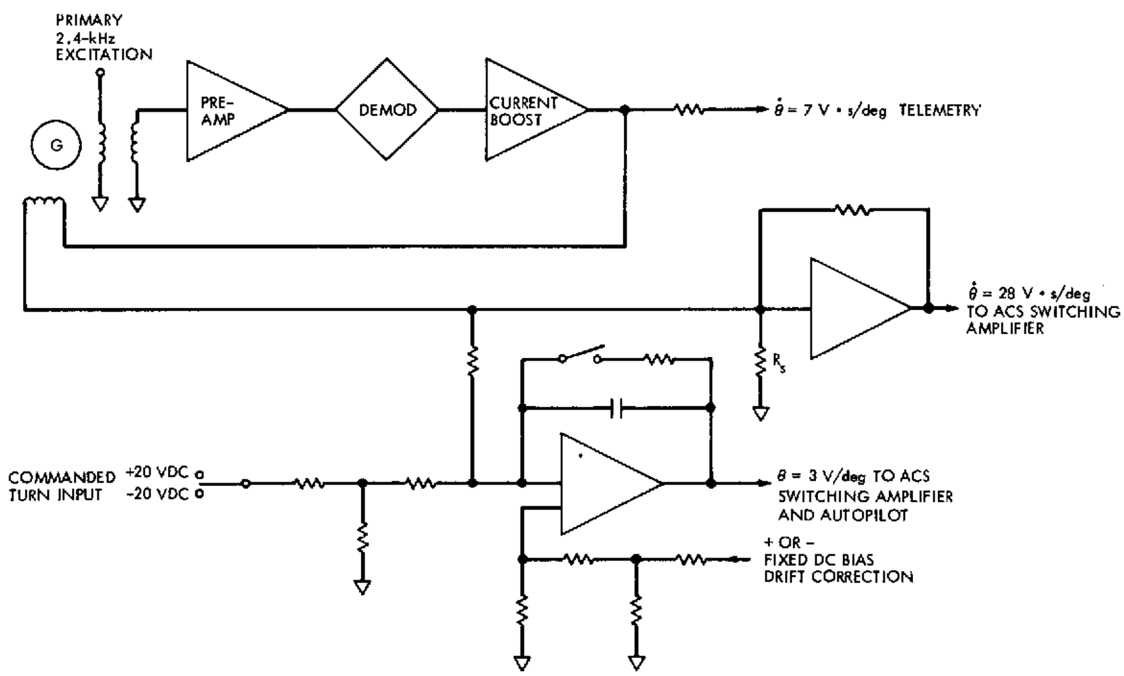

Fig. 1. Gyro circuitry

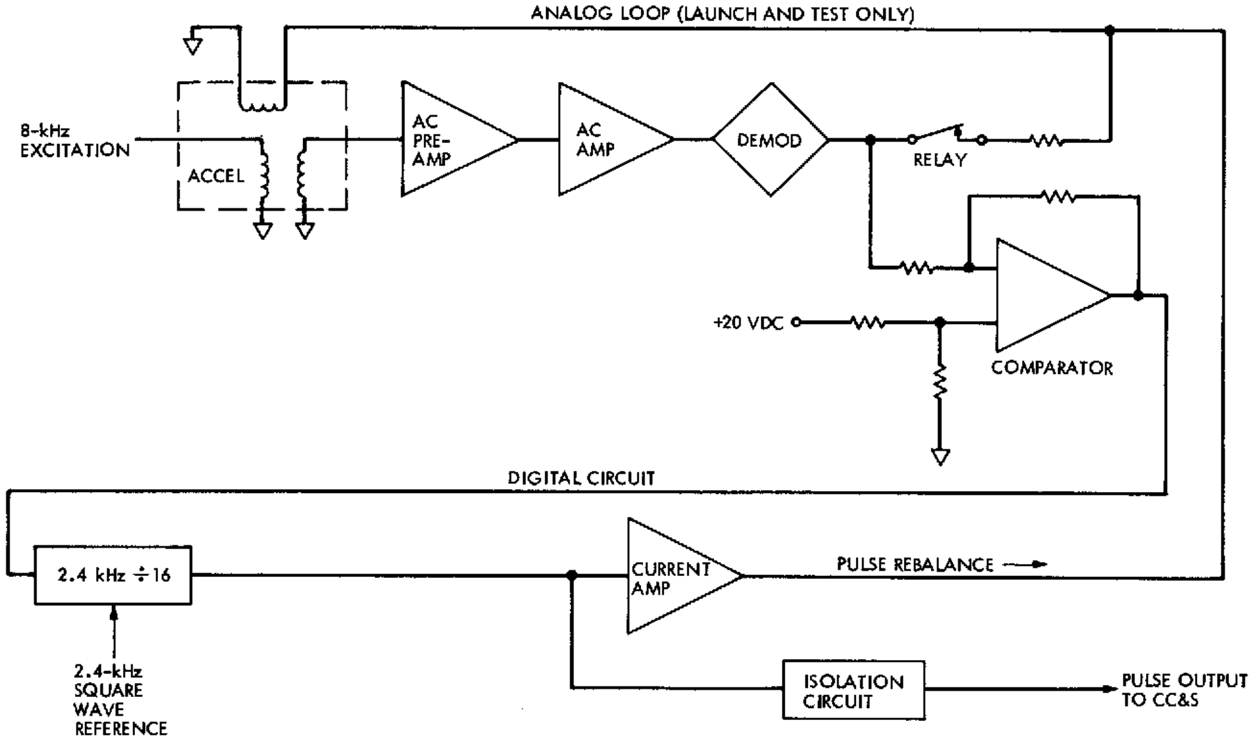

Fig. 2. Accelerometer circuitry

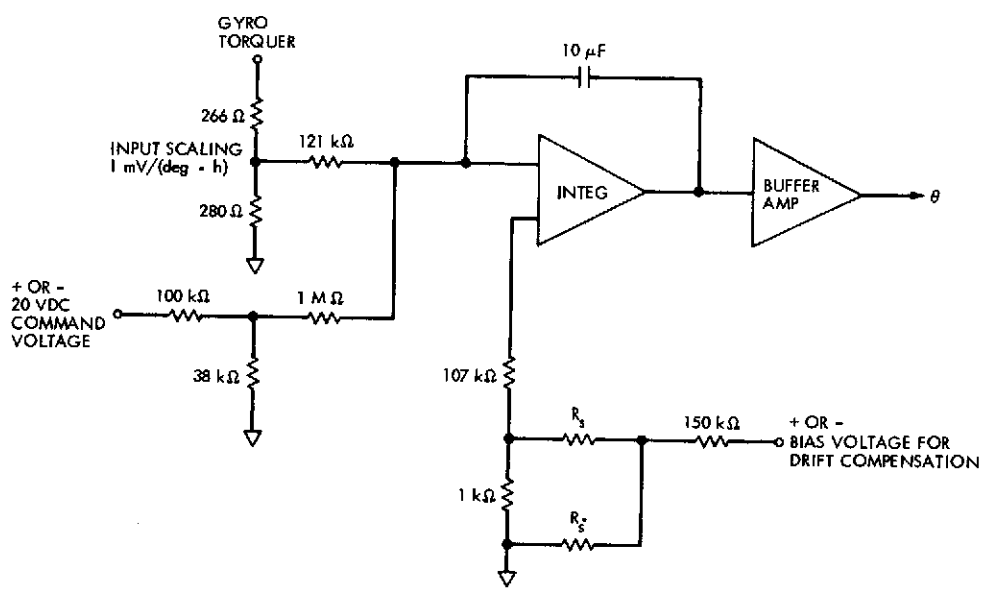

Fig. 3. Integrator circuitry

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