DELFT UNIVERSITY OF TECHNOLOGY

DEPARTMENT OF AEROSPACE ENGINEERING

Report LR-308

DESIGN AND EVALUATION OF AN INSTRUMENTATION SYSTEM FOR MEASUREMENTS IN NON-STEADY SYMMETRICAL FLIGHT CONDITIONS WITH THE HAWKER HUNTER MK VII

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SUMMARY

A high accuracy flight test instrumentation system, designed at the Department of Aerospace Engineering of the Delft University of Technology is described. The system was used during the cooperative program with the National Aerospace Laboratory, Amsterdam, to performe non-steady symmetric flight tests with the NLR laboratory aircraft a Hawker "Hunter" MK VII. The report deals with the design and evaluation of transducers, signal conditioning and the data collection system.
The goal of this program was to demonstrate the technique of measuring performance characteristics as well as stability and control characteristics simultaneously in dynamic flight conditions.

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The autor likes to thank mr. J.C. van der Vaart for his careful reading of the manuscript.

CONTENTS

1.1. List of symbols

A	static gain
A_x, A_y, A_z	specific force along the X_d X_d resp. Z_d axes of the accelerometer box
f_1	signal frequency (Hz)
f_o	filter cut-off frequency (Hz)
f_s	sample frequency (sample per sec)
i_h	stabilizer trim angle
M	output-input amplitude ratio
N	frequency ratio f_1/f_o or ω_1/ω_o
n	order of filter characteristic
n	engine speed
P_{ref}	absolute pressure in reference reservoir
Δp	differential pressure
q	rate of pitch
r	rate of yaw
T	total temperature
X_B, Y_B, Z_B	aircraft's body reference axes
X_d, Y_d, Z_d	accelerometer box reference axes
α	angle of attack
δ_e	elevator deflection angle
ζ	damping ratio
ζ_r	required damping ratio
$\Delta\tau$	time shift
$\Delta\tau_o$	time shift at zero frequency
φ	angle of bank, phase angle
φ_D	angle between X_B and X_d
ϕ	angle of yaw (change in heading)
ω_1	signal corner frequency (rad/sec)
ω_o	filter cut-off frequency (rad/sec)

1.2 List of abbreviations

ac	alternating current
BCD	binary coded decimal
CMRR	common mode rejection ratio
dc	direct current
DUT	Delft University of Technology
DVM	digital voltmeter
EGT	exhaust gas temperature
kc	kilo cycles per second
NLR	National Aerospace Laboratory
NTC	negative temperature coefficient
ppm	parts per million
PSO	parallel to series converter
RPM	revolutions per minute

1.3. List of figures

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*) These figures are folded at the back of this report.

2. INTRODUCTION

In 1967 and 1968 flight test programs were carried out with the DHC-2 Beaver laboratory aircraft of the Delft University of Technology, Department of Aerospace Engineering.
Relative short (< 60 sec) symmetric non steady manoeuvres demonstrated the flight test technique developed at this Department. To investigate the feasibility of this technique for the class of high subsonic jet propelled aircraft, a cooperative program was started with the National Aerospace Laboratory (Amsterdam) enabling flight tests to be performed with the Hawker Hunter MK VII, see fig. 1 and fig. 2. The basic principles of the non steady flight test technique are described in ref. 1 and ref. 2, whereas the results of the flight test program are discussed in ref. 3.
The applied data collection system described in this report has a sample rate of 400 measurements per sec and an overall accuracy of 0.02%. The transducers used have different accuracies depending on the type and application, from 0.01 to 5%. To monitor the accuracy during the flight test period, rather extensive calibrations were carried out before, between and after the actual measurement flights. The used calibration technique applied is described in ref. 4. Some calibration results will be presented in chapter 3.9 of this report.
The inflight measurements are recorded on magnetic tape in digital form, this tape is indicated as "tape 0". With special groundequipment and a digital computer the "tape 0" is converted into IBM compatible tape, indicated as "tape 1". This tape contains transducer output voltages and administrative data recorded in sequential form. "Tape 1" is subsequent by converted into "tape 2" which contains the physical quantities, computed by the application of calibrations. The program used is described in ref. 5 (in Dutch). The instrumentation system is designed and developed at the Department of Aerospace Engineering in Delft, whereas the installation of the system including transducers is carried out at Schiphol Airport in close cooperation with the National Aerospace Laboratory, Amsterdam. For ease of installation and to reduce the installation time as much as possible, it was decided to install the data collection system in an underwing container (pod) of the "Hunter". Transducers used in the system were located at different places, in order to maintain accessibility. A small operators panel mounted in the cockpit provided the necessary controls of the instrumentation system.
The architecture of the instrumentation system as shown in fig. 5 is rather conventional. Much care however is devoted to the design and manufacture of all the separate components for maximum accuracy, precision and reliability. In fig. 6 a diagram is presented of the accompanying ground equipment. At the time of design few components were available in modular form, so several parts of the system had to be synthesized from discrete high quality components. Some components which were available had to be adapted to meet the rather severe environmental requirements. As follows from fig. 5, conceptually the operation of the system is straight forward. The analog d.c. voltage output signals of transducers or signal conditioners are passed through low-pass filters. Filtered signals are fed to a multiplexer, the output of which is converted to a digital signal. The parallel digital output of the analog to digital converter is serialized and recorded on magnetic tape simultaneously with some administrative data. On the ground this tape is replayed and the information is passed to a digital computer (EAI 640) in parallel format. The computer finally produced a nine-track IBM compatible tape as well as quick look time histories.

The report is organized as follows.
In chapter 3 formal requirements for flight test instrumentation systems are discussed, a description of the transducers used in conjunction with the instrumentation system is given in chapter 4. A fairly detailed and comprehensive discussion from the electronic engineering point of view is presented in chapter 5. The work which remains to be done after the fully calibrated instrumentation system, including the transducers, has left the laboratory and before actually taking off for the first flight test measurements is the subject of chapter 6.

3. REQUIREMENTS OF THE INSTRUMENTATION SYSTEM

3.1. Introduction

The requirements of the instrumentation system were accumulated during several meetings of a project group in which experts from NLR as well as DUT took part. The group was formed by pilots, flight test engineers instrumentation technicians aircraft technicians and software programmers. During the specification phase of the project the following aspects concerning the onboard system received attention.

3.2. Number of channels to be recorded

The number of channels to be recorded was established at 19. During the preparation phase of the program it turned out however, that two additional transducers would be essential for accurate post flight engine net thrust calculation. Because some of the essential hard ware was already completed at that time, it was decided that these transducers would each share one measuring channel with an already planned for but not very essential transducer. Consequently a rational choice of transducers to the actually recorded had to be made before each measurement flight.

3.3. Sample rate

The sample rate of each measured variable was fixed at 20 measurements per sec. For reason of system simplicity no attempts have been made to design a random address facility for the multiplexer applied, so all measuring channels had equal sample rate.
The sequence of the measured variables was also fixed, therefore no individual channel numbers had to be added during the recording process.

3.4. Measurement accuracy and resolution

Accuracy is considered to be of prime importance for the non-steady flight test technique. This holds in particular for the inertial measurements (A_x,A_y,A_z and q).
The accuracy of the data collection system was specified to be 0.02% for each measuring channel. Hence the combined stability of scale factor and zero of the applied analog to digital converter, with a range from 0 to 10V, had to be within ± 2 mV over the entire design temperature range, from -25 to +30°C. The transducers applied in the system had different accuracies, depending on type and measurement range.
The resolution of the data collection system was determined by the analog to digital converter. The applied converter has a 4 digit BCD output, the least significant bit equals 1 mV or 0.01%.

3.5 Dynamic performance

The frequency response of the measuring channels is important with respect to aircraft motions to about 0.5 Hz introduced during non-steady flight manoeuvres. Due to the limited sample rate of the data collection system aliasing errors have to be considered. In each measurig channel, linear presampling filters attenuate the transducer output signal contents at frequencies above the filter "cut-off" frequency. The sample rate should be high enough to avoid significant aliasing errors. The flight test manoeuvres at the other hand should be designed such as to avoid significant errors of omission. The low pass frequency characteristics of the filters are selected such as to minimize difficult to recover distortions.

3.6. Environmental conditions

The environmental specifications of the instrumentation system were based on the environmental conditions expected and measured during exploratory flight tests. The data collection system including the tape recorder had to operate properly in an ambient temperature range from -25°C up to 30°C. Vibration was considered to cause no mechanical problems. Shock mounting of the electronic boxes was expected to effectivily eliminate any effect of airframe vibrations. Accelerometers and rate gyro's were strapped down dorsally mounted to the aircraft. Exploratory flight test showed the vibration level here low enough as to eliminate the need for any mechanical damper, this in contrast to earlier flight test programs with a piston engined aircraft, ref. 2.

3.7. Size and weight of the systems

The individual components of the systems were designed to be placed in a drawer which would easily fit into the underwing pod. The total weight of all components was approximately 60 kgf.
The pressure transducers were placed in one of the former ammunition boxes in the fuselage of the aircraft, for easy access and maintenance. The total weight of the "pressure transducer box" (chapter 4.2) was 22 kgf.

3.8. Electrical power supply

The instrumentation system was powered by 28 V dc and 115 V ac. The required dc power of 200 W was taken from the aircraft's dc bussystem, the ac power about 200 VA being supplied by a three phase rotary inverter.

3.9. Operation during flight

A first idea on the operation of the instrumentation system during flight was to have only a very basic control panel by the pilot. Later, it was thought that this would increase pilot workload to an undesirable level. Consequently an extra crew member would have to operate the instrumentation system. A second version of the operators panel was designed with more control and check facilities, such as a digital measuring channel voltage output display and a manual channel selector.

3.10. Calibration procedure

No separate components but rather the entire measuring channels were calibrated: transducer with associated signal conditioning including filters, multiplexer and analog to digital converter. For obvious reasons, the T and EGT transducers and measuring channels were calibrated separately as exceptions to this general procedure.
Calibrations were performed at least twice prior to the flight tests, one more during the flight test period and finally two after completion of the flight test program. As a consequence of the high ratio of transducer bandwidth to signal frequency contents only static calibrations were required. The calibration procedures are described in ref. 6, in Dutch. All calibrations were carried through in the laboratory except for the elevator deflection - and stabilizer deflection transducer, the angle of attack - and the EGT transducer. These latter calibrations were performed with the transducers installed in the aircraft. After installation of the pressure measurement system and the pneumatic tubing, dynamic characteristics of the complete pneumatic system were evaluated. It turned out that each pneumatic system consisting of pneumatic tubing, valves, connections and probes could adequately be described by a first order linear

filter. All the associated time constants were subsequently measured as further discussed in chapter 6.4. Results of calibrations are listed in table 8 which shows desired and achieved accuracies in % RMS. The reproducebility of separate calibrations is also shown in this table. As discussed in chapter 4.2, differential pressure transducer zero shifts are measured and corrected for during post flight data analyses. The effect of these corrections on the precision to be expected in flight follows by comparing the figures in the last two columns. Typical calibration results are presented in fig. 3 and 4. In these plots the data of several calibrations are combined. The differences between separate calibrations show up as deviations from a to all calibration data in the least squares sense, fitted curve. Evidently, the sensitivity of the measuring channel as well as the zero output as estimated from one calibration are in fact random variables depending on time.

3.11. Mechanical work

The mechanical work was devided into two parts. One part consisting of transducer installation, routing of electric cabeling and installation of pneumatic tubes was carried out at Schiphol Airport. The other part consisting of the construction of the drawer for the wing pod and of transducer systems was performed at DUT. The elevator- and stabilizer deflection transducers were manufactured by NLR.

3.12. Installation in the aircraft

The installation of the entire instrumentation system was eased by the fact that the data collection system was located in the under wing pod. Only one multipole connector served as an electrical connection to the aircraft. Access to the tape recorder was provided by an inspection hatch on the pod. The pressure transducer box was easily installed through the ammunition bay door. Installation of the inertial package as well as of the control surface deflection transducers however required the removal of aircraft spline fairings.

4. TRANSDUCERS

The transducers applied in the flight test instrumentation system fall into 4 categories.

1. Inertial transducers.
2. Pressure transducers.
3. Position/deflection transducers.
4. Temperature- and engine speed transducers.

4.1. Inertial transducers

4.1.1. Accelerometers

The specific forces along the aircraft body fixed X_B, Y_B and Z_B axes are measured with three force-balanc type accelerometers of Systron-Donner.Table 2, shows the types and ranges of the transducers applied. Th 3 transducers are mounted orthogonally on a stainless steel frame. The mounting flanges of this frame ar finished to an angle accuracy of 0.1'. Repeatibility of alignment of transducers after replacement i necessary, is assured by reference end stops. The frame with the 3 accelerometers is positioned in stainless steel box on 4 precision heat-isolators, in order to achieve a high heat resistance between th frame and the metal box. The outersurface of the box is also finished to an accuracy of 0.1' fo accelerometer calibration purposes, ref. 4. The temperature sensitivity of the scale factor of th transducers used is about 0.02% per °C. Laboratory tests showed that scale factor changes due t temperature variations were mainly caused by the resistance change of the measuring resistor in d feedback loop of the accelerometer. The electric connections however allow for replacement of thi resistor with a low tempco resistor with a high long term stability. Improvement of zero stability i essential and achieved by stabilization of the inside temperature of the box at 45°C. To this end proportional temperature control circuit is constructed inside the stainless steel box. A small blowe circulates the air to achieve the homogeneous temperature distribution inside the box.
Isolation from the outside world turned out to be essential for a small temperature gradient and could b accomplished by urethane foam isolation on the inside walls.
A scheme of the simple temperature regulator is given in fig. 8.
Accelerometers of the type as applied here suffer from cross-axis sensitivity. The cause for this is tha their operation depends on the deflection of a small pendulum. To keep the influence of this sensivity a small as possible the transducers are positioned on the frame in the following manner. The transverse ax of the X- and Z-accelerometers are placed along the Y-axis and of the Y-accelerometer along the X-axis This mounting scheme followed from the fact that only nominally symmetric manoeuvres were carried ou resulting in relatively small accelerations along the Y-axis.
The necessary output ranges are achieved by selection of the proper values of the measuring resistor leaving the total resistance at the same value as the internal resistance.
The connection scheme of the accelerometers is given in fig. 9.
Floating power supplies are used and signal returnes are connected to the "high quality ground" of t instrumentation system.

4.1.2. Pitch rate gyro

For the measurement of the pitch rate (q) a wide angle miniature integrating gyro, Honeywell, type GG87B is employed. The required electronic circuits for rate measurements applying the torque balance princip will be described next. For a reference schematic diagram of the gyro and electronics see fig. 10.

4.1.2.1. Principle of operation

The gyro is operated in a closed loop configuration, so the precession torque developed as a result of the input rate is balanced by a torque produced by the gimbal torquer. The torque produced is proportional to the torquer coil current which is also passed through a precision resistor (Rm), the voltage developed across this resistor being proportional to the input rate. Gimbal deflection is detected by an angle position transducer or micro-syn, the output of which is connected to an amplifier demodulator. The demodulator output signal is filtered and connected to a power amplifier stage providing the gimbal torquer current. The maximum torquing rate of this type of gyro is 23° per sec, the over all drift rate is less than 1° per hr.
Typical characteristics of the gyro are listed in table 6.

4.1.2.2. Gyro electronics

A functional diagram of the gyro electronics is given in fig. 11.
The frequency of the spin motor supply voltage should be stable in order to realize the inherently high acccuracy of the gyro. This frequency is derived from the X-tal controlled time base generator of the data collection system. The 400 Hz square wave signal is converted into a 7 V sine wave signal, the so called ac reference voltage. This reference voltage is the input to the spinmotor power supply, to a current source and to the demodulator. In the current source circuit the ac reference is converted into a current of 50 mA ac for the supply of the primary winding of the micro-syn. The output from the secondary winding (12.5 V per rad) is amplified and corrected for phase-shift in the ac amplifier stage to 7 V at the maximum input rate of 23° per sec. Phase correction is required, because of the 70° phase difference between the ac reference voltage and the micro-syn output voltage. The amplified signal is demodulated by a multiplier circuit. In this circuit the signal is multiplied by the reference voltage and the product is divided by 10, to keep the amplitude within the circuits operational range. The output is a double frequency ac signal (800 Hz) with a maximum dc component of ± 5 V. A notch filter, tuned to 800 Hz removes the ac component from the composite signal. The resulting dc signal is passed to the power amplifier, providing ± 100 mA dc at maximum input rate. The output of the amplifier is passed to the torquer coil of the gyro connected in series with the precision measuring resistance Rm.
Spin motor power is supplied by a power stage also connected to the ac reference. An output transformer is applied to prevent any dc flowing trough the spin motor coils, thus preventing additional drift of the gyro. Phase splitting capacitors provide the amount of phase-shift required by the splitfield spin motor.
The very low drift rate and the proper amount of viscous shear damping are achieved by temperature regulation of the gyro housing. To this end the gyro is provided with two heater windings and a temperature sensor. At the nominal operating temperature of 82°C the resistance of this sensor equals 780 ohm. The sensor is connected as one arm of a full bridge circuit. The bridge output is amplified to a signal which controls the power stage providing the heater current required. Gain, bandwidth and damping are adjusted to achieve stable proportional temperature regulation. With the two available heater windings connected in parallel, a maximum power of 50 W can be dissipated, requiring a current of 1A.
The gyro electronic circuits are powered via two mains transformers (115 V ac), with rectifying and regulating circuits. The box containing the electronic boards and power supplies measures 10 x 12 x 30 cm^3 with a weight of 3 kgf.
The gyro itself is mounted in a frame fastened to one side of the accelerometer box. The strainfree norál frame is provided with adjustment screws to permit exact alignment of the output-input- and spin reference axis to the reference axes of the accelerometer box. The alignment procedure was carried out on a precision turntable (Genisco) with the gyro operating in open loop configuration.
Complete schemes of the electronic circuits are given in fig. 12a and 12b.

4.1.3. Yaw rate gyro

For the measurement of the yaw rate (r) an open loop type gyro is used with a measuring range of ± 7° per sec. This gyro type I14 SFIM is provided with a wirewound potentiometer as transduction element. No electronics are required to operate this gyro as the spin motor runs on 28 V from the aircraft bus. Wheel speed is stabilized via a centrifugal force operated interrupter switch. The potentiometer was supplied by the 10 V reference voltage of the instrumentation system (chapter 5). In particular with respect to zero-shifts, the precision of this gyro is significantly inferior to the pitch rate gyro. In nominally symmetrical flight test manoeuvres this transducer has an only very minor influence on the accuracy of the flight test results and its limited precision is therefore acceptable in the present case.

4.1.4. Heading gyro

Heading change is measured with a Sperry S3A directional gyro. This gyro is normally used as part of a compass system with flux valve. Because of the relatively short flight test manoeuvre (ref. 7) the flux valve was thought to be superfluous. The gyro is provided with a synchro connected to a synchro to dc converter with a range of 10 V per 180 °. To prevent step changes occurring in the output signal when the synchro passes zero or 180°, the gyro was slaved to 90° or midscale before each flight test manoeuvre. This is accomplished as follows, see fig. 13. When the data recording system is in the non-recording mode, the slaving torque motor of the gyro is connected via the contacts of two relays to the mains power transformer. The phase of the voltage as compared to the reference of 115 V 400 Hz depends on the position of the first relay. The position of this relay is controlled by the output of a comparator. Inputs to this comparator are a dc reference voltage and the output of the synchro to dc converter. Components are chosen such as to force the comparator to switch from one state into the other when the analog input voltage reaches 5 V, which is equivalent to a 90° synchroposition. Built in hysteresis provides for a dead-band of ± 0.5°. The second relay is energized in the recording-mode of the recording system, which in turn switches of the slaving torque motor. So whenever the recording process is initiated during flight, the output of the gyro is in or near by the center position of its measuring range. It will now be obvious that no magnetic heading information is obtained, but only the change in heading during the flight test manoeuvre.
Components of the circuit are on one printed circuit which is positioned in a slot of the analog signal conditioning box to be described in chapter 5.

4.1.5. Vertical gyro

Bank angle (φ) is measured with a Sperry HGU-B vertical gyro which is provided with output potentiometers for both pitch- and bank angle measurements. Only the bank-output potentiometer is connected to the 10 V reference voltage. The accuracy achieved is limited to 0,5% of the measuring range of ± 20°, due to the limited resolution of the potentiometer and the vertical erection system which is not disabled during the test flight manoeuvres.

4.2. Pressure transducers

In the flight test instrumentation system eight pressure transducers are incorporated. Except for one absolute pressure transducer the seven remaining transducers are mounted in the so called "Pressure transducer box", installed in the ammunition bay. The types, ranges and accuracies of the pressure transducers are listed in table 3.

4.2.1. Absolute pressure transducer

The absolute pressure (P_{ref}) is measured with a force-balance pressure transducer (KTG-1902 Kelvin Hughes). The basic elements of this transducer are shown in fig. 14. In this type of transducer the force output rather than the deflection of the capsule is measured, only its piston characteristics are therefore of importance, and these are improved by the constraint imposed on the capsule.
A change of pressure in the capsule causes a movement of a beam supported on cross-spring pivots. The movement is detected by a differential transformer sensing element (E-core). The output is fed to an ac amplifier feeding the control phase of the servomotor. The lead screw is driven by the motor through suitable gearing. A nut running on the screw is attached to one end of the bifilar precision spring. Rotation of the motor adjusts the force exerted by the spring on the beam until the deflection of the beam and capsule is zero again. The angular rotation of the lead screw is thus a precise measure of the capsule force and hence of the pressure in the capsule. In practice two capsules are used, one on each side of the pivots, one is evacuated and sealed while the other is connected to the pressure to be measured. The beam is statically balanced, whilst a bimetal compensator is fitted to minimize temperature effects. The output is in the form of a shaft rotation out of the gearbox, giving 150 revolutions for full scale output. A precision slide wire potentiometer is connected to the output shaft. In the instrumentation system this potentiometer is connected to the 10 V reference voltage. The transducer is mounted by shock absorbers in a aluminium box, having an on/off temperature regulator, set at 43°C. Installation in the box is such that the pivot axis of the beam is parallel to the Z_B-axis of the aircraft. The reason or this alignment being that the transducer suffers from angular acceleration sensitivity along the pivot axis due to inertia of the rotating beam.

4.2.2. Differential pressure transducers

The differential pressure transducers are components of an integrated measurement system housed in the so called "Pressure transducer box". As discussed in ref. 3, the flight test technique depends on very accurate flight path reconstruction for which accurate change of altitude measurements are essential. This requires the measurement of change of static pressure with respect to a reference pressure which is captured at the beginning, and held constant during the flight test manoeuvre. This led to the development of a subtle pneumatic "Sample and hold" unit installed in the "Pressure transducer box". Also in the box are installed the transducer signal conditioners, the zero/measurement valves and the electronic temperature regulating circuits.
Pressure transducers are sensitive not only to pressure input but also to temperature, acceleration, power supply variation and even elapsed time, due to aging of sensing elements and drift of electronic components, resulting in zero-shifts and scale factor changes. Other properties of pressure transducers are non-linearity, hysteresis and some times different sensitivities for positive and negative pressure inputs. The effects mentioned are taken into account during design, development and calibration of the pressure measurement system.
In the "Pressure transducer box" the temperature is kept at 43 ± 1° via an electronic proportional heater control system with NTC temperature sensors. One single heater filament installed in a tube provides hot air which is circulated through the box by a small electronic lower, placed at the airintake of the tube. Small inside temperature differences during the warm-up periods are achieved by the application of several temperature sensors located at different places in the box. To enhance the temperature stability, the box is effectivily thermally isolated. All g-sensitive pressure transducer axes are set parallel to the aircraft's Y_B axis which is optimal in the case of symmetrical flight test manoeuvres. The g-sensitive axis of a pressure transducer is usually normal to the mebrane or capsule, any way in the direction of the deflection of the sensing element. Because maximal g-sensitivity is in most cases smaller than 4% per g

and the maximum expected vaule of the specific force along the Y_B-axis is approximately by 0.5 m/sec^2, this results in an only small contribution to the overall measurement error.

The zero-shifts due to temperature changes are reduced, by keeping the transducers at a constant temperature, however not completely eliminated. Therefore each transducer is provided with a pneumatic valve operated by a small electric motor. The two posititions of the valves are: "Measurement" and "Zero", fig. 17.

In the position "Measurement" one side of the differential pressure transducers is connected to the measurement probes fitted on the aircraft, the other side via a labyrinth permanently to the pressure transducer box, with absolute pressure P_{ref}. One differential pressure transducer Δp_6 is an exception to this general rule, fig. 17. Because the pressure transducer box is vented, P_{ref} depends on altitude. In the "zero" position of the valves both sides of each transducer are short circuited. This position is used for the measurement of zero-shifts before and ater each flight test manoeuvre. These zero-shifts are subsequently corrected for during data analysis. The static reference pressure is stabilized against fan induced pressure variations via the labyrinth mentioned above.

For the measurement of change of altitude mentioned above the differential pressure transducer indicated as Δp_1 in fig. 17 is of major importance.

With Δp_1 the relative change of P_{ref} is measured more accurate then could be deduced from absolute pressure measurements. The measurement problem becomes more complicated however and requires a pneumatic "Sample and Hold unit".

An important item of this system is a thermosflask connected through a two-way valve and a second flask to the reference pressure. At manoeuvre initiation the flask is shut of from the reference pressure, which is at that moment captured in the flask. The low range differential pressure transducer ΔP_1, between the flask and the reference pressure, measures consequently the change of the reference pressure P_{ref} with respect to the captured pressure. Change in static pressure equals the sum of ΔP_1 and ΔP_4 outputs.

The accuracy achieved in this way not only depends on transducer accuracy but also on the stability of the pressure captured in the flask. The temperature in the flask has to be constant within ± 0.03°C during each flight test manoeuvre (approximately 4 minutes). To achieve this temperature stability the heat capacity inside the flask is increased by insertion of 150 grf of stainless steelwool. The large contact surface guarantees a fast heat exchange. The isolation of the flask is extremely important, also with respect to the connection to the valve and the pressure transducer. Actually poly-urethane foam is used to provide sufficient isolation. A small electric heater and a temperature sensor are mounted in the flask in order to achieve a reasonable warm up time of about 45 min. starting from 20°C ambient temperature. The heater element in the flask is controlled by the temperature difference between the box and the flask, which tracks the box temperature without overshooting.

The second flask is used as a temperature equalizer and prevents cold air entering through the valve in the measurement flask during non-measurement descents. This flask is also filled with steelwool and provided with a tracking heatercontrol.

The valves used in the system are manufactured by Circle Seal, type 9300 3-way selector valves. This are valves with a cylindrical plug provided with a special arrangement of Buna N O-rings to achieve a "bubble tight" seal. Puppet valves available in much smaller size, do inject air at the moment of closing henceforth creating an offset pressure when used for zero shift measurements.

Dimensions of the pressure transducer box, are 20 x 40 x 60 cm^3, weight 22 kgf, and was extensively tested under different environmental conditions.

A step of change of outside temperature of -55°C resulted in an inside temperature change of 3°C after one hour. This temperature shock measurement carried out in the available test chamber resulted in a maximum rate of change of a 4 Pa per min. of the pressure in the closed thermosflask, a figure which equals an altitude variation of approximately 0.3 m/sec at sea level occurring one hour after the thermal shock was applied. A short flight test program with the laboratory aircraft DHC-2 "Beaver" was carried through to perform combined temperature/low pressure measurements. During a test flight lasting for about 3 hours, an altitude of 20.000 ft was reached with a cabin temperature of -4°C. The maximum temperature change inside

:he box was 1.5°C.
complete scheme of the pressure transducer box is given in fig. 18, the heater regulators are drawn in 'ig. 19. In fig. 20 a picture is presented of the box as installed in the aircraft.

.3. Position/deflection transducers

Three transducers were required to measure elevator deflection, stabilizer deflection, or trim angle and :he angle of attack. The transducer types and measurement ranges are given tabel 4 (ref. 8).

.3.1. Elevator deflection transducer

The elevator deflection angle is measured with a transducer designed and manufactured in close cooperation ith NLR. A precision cermet potentiometer which runs in ball bearings is driven by a set of gears with a atio of 6 to 1. This results in an input freedom of about 55°. The elevator deflection, relative to the orizontal stabilizer, ranges from 9° down to 22° up. Lever lengths are chosen with a ratio of 3 to 5. In this way the rotation angle of the potentiometer equals about 310° for the full deflection range of the levator. The cermet potentiometer, Beckman Helipot Division type 6603/2, is chosen for its high eliability, reasonable linearity (0.5%) and low temperature coefficient 100 ppm. The gear box is provided ith an antibacklash, spring loaded split gear segment driving the pinion on the potentiometer shaft. orques and acceleration induced forces were considered for the calculation of the necessary spring ension. The connecting rod between the two levers is made adjustable, providing for the necessary offset, ue to the unequal up- and down travel of the elevator. The levers have steel-bronze-teflon precision ball oints to allow for smal misalignment of the transducer reference axis and the elevator lever axis. The ransducer itself is attached to the inner surface of the horizontal stabilizer root rib, with an adjust-ble mounting flange. The elevator lever is bolted to the elevator control horn. A drawing of the location f the transducer is given in fig. 21. A certain measure of nonlinearity is necessarily introduced by the pplication of levers with unequal lengths and by the interconnection mechanism of the two elevator alves. The applied calibration technique, however can deal with nonlinearity. Unfortunately nothing could e done to reduce the play in the 8 bearings of the cardanlike actuator mechanism of the elevator. The ffect of play reaulted in a "slack" of approximately 0.08° measured at one of the elevator halves, which s indeed twice the accuracy of ± 0.04° achieved durig the flight test period. See ref. 4.

.3.2. Stabilizer trim angle transducer

he transducer for the measurement of the trim angle is designed very much like the elevator transducer. he potentionmeter and the gears are the same except for the number of teeth of the segments. The up- and own travel of the stabilizer are respectively 2.5° and 3°. In order to achieve reasonable potentiometer otation the lengths of levers are chosen quite different: 24 mm for the transducer lever and 210 mm for he stabilizer lever.
his introduced a rather non-linear calibration curve, which is not a great disadvantage, as the trim etting for each flight-test manoeuvre was fixed.
he measurement principle and the transducer are shown in fig. 22. The accuracy achieved during the flight est period is 0.01° or 0.2%.

.3.3. Angle of attack transducer

he angle of attack is measured with a wind vane transducer, fig. 23, mounted on a nose boom.
hen the transducer was installed, no use could be made of the nose pitot tube.
uring flights with the angle of attack transducer installed, the total pressure was measured via a wing ounted pitot tube.

The transducer, designed by NLR is provided with a precision synchro, Kearfott 7E-911-1 with an accuracy of 7'. Conical gears are applied to achieve right angle translation of the vane axis. The gear ratio is 2 to 1, so the synchro rotation is twice the vane axle rotation. The gear on the vane shaft is preloaded with a leafspring to minimize play at different temperatures.
The output of the synchro, 11.8 V, 400 Hz is converted to dc voltage with a synchro converter having a range of 180° for full scale output. In this way full scale output is achieved with a vane deflection of ± 45°. The calibrated range is ± 30°, relative to the aircraft X_B-axis. The converter used is the same as used for the heading change measurement and was placed in the analog signal conditioning box, see chapter 5.7.

4.4. Engine speed measurement

Engine speed measurement is performed with the tacho generator already installed in the aircraft as a transducer. The frequency of the three-phase output voltage is used as input to a frequency to voltage converter, see fig. 24.
At the converter the three voltages from the generater are connected to three operational amplifiers, which act as full wave rectifiers. The triggers connected to the outputs convert the output signals into two positive pulses for each period. These pulses are differentiated and the negative going edges are summed at the clear input of flip-flop A. The clock input is connected to a divide-by-hundred circuit, consisting of two decade counters. The counter input is the 40 kc master clock of the instrumentation system. The second flip-flop synchronizes the output with the master clock, thus removing the uncertainty of one clock-pulse of 25 µs. In this way a stable pulse width of 2.5 ms is achieved. The output is connected to an analog switch which is supplied by the systems 10 V reference voltage.
The output of the switch is a precizely controlled positive pulse with a stable amplitude. Each tacho generator revolution produces six of those pulses. This yields a maximum 4000 RPM for the generator, permitting a maximum of 8150 RPM for the jet-engine due to the gear ratio applied in the generator drive. The pulse train is low-pass filtered to a dc voltage proportional to the frequency of the generator output voltage.
The accuracy of this circuit depending on the stability of the clock frequency, the reference voltage stability and on characteristics of the analog switch, is better than 1 mV over a temperature range from -30 to + 50°.
The circuit components are mounted on a single printed circuit board, placed in the analog signal conditioning box.

4.5. Total temperature measurement

Outside temperature is measured with a total temperature sensor, type 102-W2W from Rosemount Engineering Company Ltd. This model is a non de-iced transducer provided with two 50 ohm sealed platinum resistance sensing elements. The transducer shown in fig. 25 was mounted on top of the underwing instrumentation container. Only very short electrical connection leads to the signal conditioner were required which in turn simplified the electronics. Ref. 9 gives detailed information on in flight temperature measurements.
The signal conditioning, see fig. 28 is straightforward. However, special attention is payed to the temperature sensitivity of the circuit, for relative high voltage gain is required to achieve the standard dc signal level.
A stable bridge voltage is derived from the -5 V reference voltage with low temp.co. resistors and an operational amplifier.
Three more resistors complete the four arm bridge in which the sensor elements form the active arm. The bridge is slightly offset to create an unipolar output. A differential amplifier with a high cmrr connected to the bridge provided an output of 100 mV per °C. The initial offset is adjusted to zero with the balance potentiometer. The overall temperature coefficient of the entire circuit is about 10 ppm per

°C over the temperature range from -30 to +50°C, which is about 0.1°C related to the output range of ± 50°C.

It must be emphasized that this figure only concerns the electronic circuit. The accuracy of the temperature measurement depend heavily on the transducer accuracy which is stated by Rosemount to be ± (0,25 ± 0.005t)°C. This figure results in ± 0,5°C at the extreme ends of the measuring range, see fig. 26. The time constant for this hermetically sealed sensor system depends on airspeed and Machnumber and may vary between 3 en 5 sec, see fig. 27.

4.6. Exhaust gas temperature measurement

The standard equipment of the "Hunter" was used to perform EGT measurement. A functional diagram of the installation is presented in fig. 29. Eight chromel-alumel thermocouples are installed in the jetpipe of the engine. These thermocouples are connected in parallel to provide an input signal for a magnetic amplifier and a cockpit indicator. A third circuit connected to the thermocouples is used for flight test puposes. This circuit consists of a servo indicator, Howell BH187B80, which accepts thermocouple input. The servo system is provided with a second potentiometer which is connected to the 10 V reference voltage. A disadvantage of this system is the hysteresis of the mechanical part of the servosystem which is equivalent to 3°C, and the null uncertainty of about 8°C. The accuracy achieved with this system over the flight test period is ± 0,6% or ± 4°C with a measuring range from 0 to 700°C.

5. ELECTRONIC PARTS OF THE INSTRUMENTATION SYSTEM

In this chapter the different electronic systems will be discussed separately in the following order:

1. Analog filters
2. Multiplexer
3. Analog to digital converter
4. Parallel to series converter
5. Digital recorder
6. Analog signal conditioning box.

5.1. Analog filters

5.1.1. Introduction

Filtering transducer output signales is usually necessary before recording the data with a sampling digital recording system. Filters are used to attenuate the amplitude if the high frequency contents of the signals. The higher frequencies in the signals are considered as noise due to structural vibration, transducer noise and electromagnetic interference.
In the present flight test program, pilot induced motions had frequencies from zero to about 0.5 Hz. Attenuation of high frequency signal components is required to reduce aliasing errors caused by the finite sampling rate of the data collection system. This sample rate equalled 20 measurements per second for each variable.
Two conflicting requirements arise in designing pre-sampling filters. On the one hand, the filter cut-off frequency should be high as compared to the frequency range of the signal of interest, in order to obtain a flat frequency response and a constant group delay over the specified frequency range. On the other hand the filter cut-off frequency should be low as compared to the sampling rate in order to keep the amplitude of fold back frequencies withing the required accuracy limits.

5.1.2. Filter specification

The required bandwidth was 0.5 Hz minimum with an amplitude ratio of input- and output signal between 998 and 9.002 in the specified frequency range. The time delay of the filters should be within ± 2 msec. The static gain was not too important for this value is taken into account during the static calibration proces. The gain stability required however equalled ± 0.02%. The aliasing error should be less then 0.1% with a sample rate of 20 measurements per second. The expected ambient temperature variations ranged from -25°C to +50°C.

5.1.3. Filter design

Filter design is commenced with an attempt to establish the required order of the filters. This calculation carried through is based on the aliasing specification of 0.1% with the given sample rate of 20 s/sec. Further more it is obvious that the filter cut-off frequency is much smaller then the sample frequency, ref. 6.
For a n-th order low pass filter the following equation is valid when $\omega_1 >> \omega_o$

$$\log M \cong - n \log \frac{f_1}{f_o} \qquad (1)$$

The first image depends on the difference between the sample frequency f_s and the signal frequency f_1, see also fig. 30.

Substitution of $f_s - f_o = f_1$ in (1) yields:

$$\log M \cong - n \log \frac{f_s - f_o}{f_o} \qquad (2)$$

Now for different values of n and fixed values of M and f_s, the highest allowable filter cut-off frequency f_o can be calculated.
M = 0.001 and f_s = 20 substituded in (2) results in

$$f_o \cong \frac{20}{10^{3/n} + 1}$$

The values of f_o for different values of n are given in the following table:

n	f_o (Hz)
1	0.02
2	0.6
3	1.8
4	3.0
5	4.0

From the results above it is clear that the first order or second order filter will not satisfy the amplitude requirements with signal frequencies up to above 0.5 Hz at the same time satisfying the time delay requirements. Practical considerations (simplicity in design and component selection) lead to the choice of a 4-th order filter, to be realized with identical 2-nd order filters connected in series. The cut-off frequency was established at 19.0 rad/sec or practically 3 Hz.

The next problem to be solved was the determination of the damping coefficient of the filters. The following is a description of the method used.

The output-input ratio of a 2-nd order system is described by the equation:

$$M = \{(1 - N^2)^2 + (2\zeta N)^2\}^{-\frac{1}{2}} \qquad (3)$$

The phase angle between output and input is given by:

$$\varphi = - \text{arctg } 2\zeta N \, (1 - N^2)^{-1} \qquad (4)$$

Now in order to achieve a bandwidth as high as possible, with the amplitude ratio M between the gain limits, it is necessary to apply a damping smaller then critical. So the gain will increase to the upper limit of M at a certain signal frequency. From this frequency the gain will decrease, passes M = 1 and reaches the lower limit of M at the maximum allowable signal frequency. An illustration is given in fig. 31.

The required value of ζ is computed as follows. From setting the derivative of (3) to zero and solving for N is the result:

$$N = (1 - 2\zeta^2)^{\frac{1}{2}} \qquad (5)$$

With this value for N the maximum of the frequency characteristic is obtained henceforth the maximum value for M. When now the value of N from (5) is substituted in (3) and ζ is solved is the result:

$$\zeta = \{\tfrac{1}{2} - (M^2 - 1)^{\frac{1}{2}} \cdot (2M)^{-1}\}^{\frac{1}{2}} \qquad (6)$$

With the maximum value for M being equal to 1.001 the calculated required value ζ_r equals:

$$\zeta_r = 0.691$$

This value of ζ_r and the minimum value for M substituded in (3) and solving for N yields:

$$N = [1 - 2\zeta_r^2 + \{4\zeta_r^2 \, (\zeta_r^2 - 1) + M^{-2}\}^{\frac{1}{2}}]^{\frac{1}{2}} \qquad (7)$$

With the real values $\zeta_r = 0.691$ and M = 0.999

$$N = 0.329$$

Henceforth the maximum allowable signal frequency in accordance to:

$$f_1 = N.f_o$$

$$f_{max} \cong 1 \text{ Hz}$$

Having established the cut-off frequency and the damping ratio, the time delay of the filter as function of the impressed frequency can be calculated, referring to equation (4):

$$\varphi = - \text{arctg } 2\zeta N \,(1 - N^2)^{-1}, \qquad (4)$$

the time shift equals:

$$\Delta\tau = - \frac{\varphi}{\omega_1} \qquad (8)$$

Combining (4) and (8) yields:

$$\Delta\tau = \frac{\text{arctg } 2\zeta N \,(1 - N^2)^{-1}}{\omega_1}$$

The negative sign denotes that the output is lagging with respect to the input.
For different values of N the time shift is calculated with the following results:

$\omega_1(r.sec^{-1})$	f_1 (Hz)	N	φ(rad)	$\Delta\tau$ (sec)
0	0	0	-	0.0727
1	0.1592	0.0526	0.0728	0.0728
2	0.3183	0.1053	0.1460	0.0730
3	0.4775	0.1579	0.2202	0.0734
4	0.6366	0.2105	0.2905	0.0739
5	0.7958	0.2632	0.3726	0.0745
6	0.9549	0.3158	0.4515	0.0752
7	1.1141	0.3684	0.5324	0.0761

From the results it can be seen that the time shift increases with about 2 msec for frequencies to 1 Hz. A plot of the time shift characteristic is presented in fig. 32. As a result, for the frequency range relevant to flight testing, the time shift was established to 73.5 ± 1 msec for one second order filter.

5.1.4. Realization of the filters

The 2-nd order low-pass filters are realized by using operational amplifiers and stable R-C networks. The basic component configuration is given in fig. 33 which shows a multiple feedback circuit. This kind of filter has the advantage of mathematical simplicity and is based on the behaviour at low frequencies compared to cut-off frequency, with good phase and amplitude response.

The following equations describe the low-pass circuit:

$$\omega_o = (R_2 R_3 C_1 C_2)^{-\frac{1}{2}} \qquad (9)$$

$$\zeta = \left\{ \frac{(R_2 + R_3 + \frac{R_2 R_3}{R_1})^2}{4 R_2 R_3} \cdot \frac{C_2}{C_1} \right\}^{\frac{1}{2}} \qquad (10)$$

$$A = -\frac{R_2}{R_1} \tag{11}$$

The static gain (zero frequency) required is nominally unity, so the circuit produces a signal in version which adds an extra constant phase shift of 180° to the input-output phase relation.

Substitution of $R_2 = R_1$ and solving R_1 and R_3 from (9) and (10) yields:

$$R_1 = \{\zeta + (\zeta^2 - 2\frac{C_2}{C_1})^{\frac{1}{2}}\}.(\omega_o C_2)^{-1} \tag{12}$$

$$R_3 = \{\zeta - (\zeta^2 - 2\frac{C_2}{C_1})^{\frac{1}{2}}\}.(2\omega_o C_2)^{-1} \tag{13}$$

The calculation of the component values is initiated with the selection of the capacitors, because there are fewer standards of high quality capacitors than there are resistors. Also resistors are more easily used for final adjustment of the complete circuits.
Some trial and error work is required to select the proper capacitors in relation to acceptable resistor values. This is done with respect to circuit input impedance and amplifier characteristics, e.g. voltage and current drift due to temperature changes. The capacitor values determined are:

$C_1 = 0.47\ \mu F$ and $C_2 = 0.1\ \mu F$

The required resistor values are calculated with $\zeta = 0.691$ and $\omega_o = 19$ rad sec^{-1} substituted in eq. (12) and (13).
The nominal component values calculated this way are listed in the following table:

Component	Value	Unit
R_1	484	KΩ
R_2	484	KΩ
R_3	122	KΩ
C_1	0.47	μF
C_2	0.1	μF

It is necessary to look at the sensitivity of resistor values in respect to capacitor tolerance, the selected capacitors only having a tolerance of 1% and the requirement of all filters being equal. With a simple program the extreme values of resistors were calculated, thereby taking into account the capacitor tolerance. The calculated values are:

Component	Minimum	Maximum	Unit
R_1	470	500	KΩ
R_2	470	500	KΩ
R_3	118	127	KΩ
C_1	0.465	0.475	μF
C_2	0.099	0.101	μF

The actual resistors are assemblies of a fixed part and an adjustable part with the following values:

Component	Fixed part	Adj. part	Unit
R_1	470	25	KΩ
R_2	470	25	KΩ
R_3	120	10	KΩ

The resistor R_4 from fig. 33 is not yet mentioned in the description but for the measurement of bipolar input signals this resistor is needed to achieve an unipolar output. This resistor when connected to the -5 V reference voltage provided a bias to the output of the second stage of the filter circuit. The value of R_4 calculated to provide +5 V output is 728 KΩ.

5.1.5. Filter construction

Four second order filters are assembled on one printed circuit board of 30 x 6 cm^2, which is enclosed in a dural frame work. A total of 10 units is placed in the filter box. Four switches and four screwdriver potentiometers on the front panel of each unit provide for initial offset adjustment of the operational amplifiers. The signal- and the power supply connections are made through one connector on the backpanel of each filter unit, see fig. 34.

5.1.6. Filter components

Chopper stabilized amplifiers type 210 from Analog Devices are used because of the stability requirements. These amplifiers have a voltage offset drift of 1 µV per °C and a current drift of 3 pA per °C over a temperature range from -30 to +50°C. One disadvantage of this type of amplifier is the unbalanced power input resulting in current flowing through power ground leads. This effect however was reduced by the application of suitable bleeding resistors.
Polystyrene capacitors are preferable in precision filter circuits for low frequencies because of their low temperature sensitivity, usually below 25 ppm per °C. Their physical dimensions hower are much larger compared to polycarbonate capacitors. Metallized polycarbonate capacitors, type MKB-3, are chosen for the filter circuits, they exhibit a reasonable long term stability 3.10^{-4} per y and a temperature sensitivity of 50 ppm, per °C. The tolerance available at the time of construction was ± 1%.
Stability and low temperature sensitivity are the requirements of the resistors, the resistance values being of less importance. The resistors used are from Ultronix, type 205 RP, having a temperature sensitivity of 10 ppm per °C, tracking 2 ppm, a long term stability of 1.10^{-4} per y and a tolerance of ± 1%. The resitors are small bifilarly wound with radial leads for printed circuit mounting. The variable resistors are from Beckman Helipot. The small diameter potentiometers have cermet resistance elements and the necessary infinite resolution. Temperature sensitivity depends on the range used but is in the order of 100 ppm per °C.
A complete scheme of one filter unit with four 2-nd order filters is presented in fig. 36.

5.1.7. Adjustment of the filters

The final adjustment procedure is carried out with the analog computer EAI 680 of the Department of Aerospace Engineering. An analog model of the filter was made with the calculated values of gain, damping and cut-off frequency. Precision components with a nominal accuracy of 0.01% are used in this computer.
Fig. 35 shows the block diagram of the adjustment sett-up. Model and filter were excited with the same

input, a low frequency sine wave or a square wave signal. During the adjustment with the three filter potentiometers, the output of the summing amplifier was monitored. When equality of model and filter was achieved, this signal remained zero for every input. A peak to peak output of 2 mV was considered to be acceptable for full scale input signals.

5.1.8. Filters with voltage gain

Because of the ± 2 V full scale output of the pressure transducer system, six filters are required with a static gain of 5. This is realized in the first stage of the filter unit. Component values of these circuits, determined with the method described for the standard filters, are listed in the following table.

Component	Fixed part	Adj. part	Unit
R_1	286	20	KΩ
R_2	1430	100	KΩ
R_3	116	10	KΩ
C_1	0.47	-	µF
C_2	0.033	-	µF

The adjustment procedure is carried out in the way described, only the potentiometer setting in the analog computer being changed. The input signal ofcourse was reduced to an acceptable level.

5.2. Filter box

The filter box is a ruggedized container in which 19 filters are placed. Also mounted in this box are the systems reference voltage
unit, the voltage reference element, two high input impedance buffer amplifiers and the balance relais unit.

5.2.1. Systems reference voltage unit

Transducers with potentiometric sensing elements were supplied with a stable dc voltage: the systems reference voltage. Furthermore this voltage was used for the synchro to dc converters and the engine speed converter. Although the exact nominal value of this reference voltage is 9985 mV, it is referred to as the 10 V reference in this description. A second reference voltage of -5 V was used to provide the required offset for some filter outputs as mentioned in 5.1.4. The bridge supply voltage of the total temperature measurement circuit was also derived from the -5 V reference.
Two chopper stabilized amplifiers, see fig. 37, are used, with suitable resistor networks. Either the internal reference element or the reference element of the DVM can be used as the input for the reference unit. In both positions of the switch DVM/INT, the output voltage remains the same within ± 1 mV. This feature is included for control purposes during pre- and post flight checks.
Normal operation is with the switch in DVM position.
A second switch 0/-5, is used in order to create a zero input for the filter amplifiers provided with bias inputs, during the zero adjustment of the amplifiers.
The output of the second amplifier in the reference circuit is buffered with a single transistor to increase the output current capability. Output sensing takes place at the systems reference point (terminal strip) to avoid voltage drop in long connecting leads.

The output change is less than 1 mV over the ambient temperature change from -25 to +50°C.
The circuit is contained in a frame with dimensions equal to those of the filter units, both switches are readily accessable on the fron panel.

5.2.2. Voltage reference

A silicon reference element BZY25 is used for a stable reference voltage of 8.4 V. Although having a temperature coefficient of only 10 ppM per °C, the element is placed in a small temperature controlled box with a regulated temperature of 60°C. Fig. 39 is a scheme of the voltage reference element connection.

5.2.3. Buffer amplifiers

Two high input impedance buffer amplifiers type AD180 provide for the required unloading of the 100 KΩ potentiometers of the stabilizer- and elevator deflection transducers. Also filter characteristics are not changed, because of the low output impedance of the buffer amplifiers. The electronic circuit, fig. 38, is mounted on a filterboard instead of the second filter, so the 10th unit in the filter box containes one 4th order filter and the two buffer amplifiers.

5.2.4. Relay unit

With the relay unit applied, a means is created to record the filter off-sets during flight for correction purposes if necessary. The filter inputs are switched with 5 simultaneously operated relais to a 10 mV reference voltage derived from the 10 V reference. A so called "live zero" is applied. The relais are energized through the mode control switch installed on the operators panel in the cockpit.

A complete scheme of the filterbox is presented in fig. 39.

5.3. Analog multiplexer

The analog multiplexer used in the instrumentation system has 19 analog channels, which are scanned in sequential order. The sample rate is 20 times per second for each individual channel and no subcommutation is applied. For reasons to be described later the multiplexer consists out of two subsystems. A first system containing 19 electronic analog switches is used during sequential operation. A second system is used during the calibration of transducers and for the execution of pre- and post flight checks. It consists of 20 reed relais. Fig. 40 gives an impression of the multiplexer principle.
The electronic switches are single throw double pole ones, Epsylon DPAS2. Although these are differential switches, they are used single ended in the multiplexer. An excellent property of these switches is the high ratio between the "off" and "on" resistance, about 10^9. The switch control circuit is isolated from the analog signal path. The coupling of the gate control applied, however, introduces a disadvantage. The maximum "on time" is limited to only 2 msec whereas the required "off time" between on periods equals about 20 msec.
As a consequence the modular switches can not be used for static operation.
The series "on" resistance of each switch is determined. In the reed switch circuit small series resistors are included in the signal path. This in order to achieve equal signal path impedances undependent of the fact wether the signal is fed through the electronic switches or through the reed switches. The reed scanner is operated by a multi-position switch mounted on the operator's panel. In this way even during flight it was possible to monitor one particular analog channel. From the scheme in fig. 42 it can be seen that the first position of the multi-position switch activates the output of the electronic switches. The last position is used for energizing the reed switch connecting the 10 V reference to the multiplexer output. The reed switch contacts are connected in series to avoid possible short circuiting of two

channels during switching.

The gate control pulses of 2 msec duration each, separated by 0.5 msec are generated in the parallel to series converter, which will be described in chapter 5.5. The TTL outputs of this converter are buffered with non-inverting transistor circuits, to provide sufficient gate input current.

Reed switches, analog switches and buffer stages are mounted on two identical printed circuit boards, placed in a dural box. The rear panel is provided with 5 sockets, for digital- and analog inputs, for manual channel selection, analog output and for the power supply connection.

The required power supply voltages are made in the instrumentation systems digital power supply unit. Some general specifications of the multiplexer are given below.

Multiplexer specifications	
Acquisition time	5 μs
"On" impedance	~ 50 ohm
"Off" impedance	~ 10^5 Mohm
Offset voltage	100 μV max
Temperature range	-40 to +50°C
Signal current	1 mA max
Gate current	4 mA max
Supply voltages	
DPAS2	+6 V and +12 V
Buffer stage	+5 V, -15 V, +12 V and +6 V
Reed switch	+12 V

5.4. Analog to digital converter

The analog to digital converter (adc) used in the instrumentation system is a modified digital voltmeter (DVM) type 6000 from EAI. Experience gained with an earlier flight test instrumentation system, used for non-steady measurements with the De Havilland DHC-2 "Beaver" aircraft, lead to the selection of this DVM.

Significant specifications of this DVM	
accuracy	0.01% of reading ±1 LSB
resolution	0.01% of full scale
stability	0.01% + 1 LSB/year
input impedance	~ 500 Mohm
conversion time	1 msec unipolar input
	35 msec bipolar input
input ranges	± 1V ± 10V ± 100V d.c.
display	5 digits
temperature range	+10 to 40°C
power requirement	115V 60 Hz ~ 100 VA
weight	~ 14 kgf

The logic system of the DVM to make a conversion requires 16 trials for each new reading. There are four binary coded (8421) resistors per decade and a logic system determines wether each of these resistors should be dropped or retained for a given reading. Therefore four trials are needed per decade and 16 trials per reading. These trials proceed at a clock rate of 20 kc to give 16 trials plus command signals for one complete measurement within 1 msec.

A peculiarity of the DVM is the long time required (35 ms) to deal with a change in polarity of the input signal. This is caused by the application of a sign-relay in the DVM. This phenomenon is the reason why all input signals have positive polarity referred to ground reference.

Inputs to the DVM in the instrumentation system are:

- The time multiplexed analog signal.
- The sample/hold command.
- The conversion start command.

Outputs are:

- The conversion complete signal.
- The negative sign signal.
- The binary coded decimal output.
- The decimal output.
- The clock reference 10 kc, 20 kc and 40 kc.

A simplified timing diagram of the DVM is given in fig. 43.

A few details had to be modified before this DVM was acceptable for flight test purpose. The modifications carried out concerned the following:

- The reference clock.
- The input amplifiers.
- The reference voltage.
- The power supply.
- The display.

5.4.1. Reference clock

Originally the reference clock of the DVM consisted of a unijunction oscillator operating at a frequency of approximately 22 kc, its stability being insufficient to serve as a time reference. This oscillator is replaced by an X-taloscillator with an output frequency of 40 kc. Additional circuitery provids for level,shifting and frequency dividing to 20 and 10 kc, see fig. 44. The 20 kc is used in the DVM, resulting in a stable conversion time of 1,1 m/sec. The 40 and 10 kc are fed out to the digital output connector for use in other parts of the system. Stability and accuracy of the time reference clock is enhanced to 0.01 °/°° over the ambient temperature range of -25 to +50°C. The necessary circuits are mounted on a printed circuit board placed in space slot A7 of the DVM.

5.4.2. Input amplifiers

The two applied amplifiers of the DVM were stabilized with a electro-mechanical chopper, running on the line frequency. Since the aircraft power supply with a 400 Hz line frequency had to be used, the complete amplifier board was removed and replaced by a pin compatible dual amplifier board. This board contains two operational amplifiers type 1003/01, Teledyne Nexus. This are FET input amplifiers with an exceptional low voltage drift of only 1 µV per °C. They are specified to operate on supply voltages of ±28 V. The available ±20 V in the DVM is sufficient to achieve the required ±12.5 V output voltage of the amplifiers. This 12.5 V output with an input of 10 V is required for the realization of a stable high input impedance, by the application of positive current feedback.

Since the conversion time of the DVM is 1.1 msec and the signal variations due to aircraft motion may be up to 15 mV per msec, the necessity of a sample and hold circuit is obvious. This circuit, the principle given in fig. 45, is built around the input amplifier. The control signal for the sample/hold circuit is generated in the parallel to series converter (chapter 5.5.4.). The hold time is 1.4 msec consequently the sample time equals 1.1 msec. In order to achieve a low decay rate and a small voltage built up due to dielectric absorption, high quality storage capacitors, together with fast electronic switching components are required for precision sample and hold circuits. In the actual circuit a polystyrene capacitor is applied. In the switching circuit use is made of the low "on" resistance of PNP transistors and the high "off" resistance of Field effect transistors. The decay rate (voltage change in "hold") of this circuit is about 1 mV per sec, the offset from sample to hold is 0.2 mV and the effect of dielectric absorption is 0.01% of the applied voltage after 1 sec. The complete scheme of the sample and hold circuit designed is given in fig. 46.

5.4.3. Reference voltage

The reference voltage supply consists of a stable voltagesource and the reference amplifier, stabilized with a mechanical chopper. The zener diode network is placed in a temperature controlled oven. Increasing the thermal isolation was necessary because the maximum available power dissipated by the heating element was insufficient for low ambient temperature operation.
The small mechanical chopper which operated well even under vibrational conditions is maintained in the circuit. The necessary 50 Hz driving frequency is derived from the 400 Hz mains supply voltage. To this end an electronic circuit is placed in space slot A8 of the DVM. This circuit, see fig. 47, contains a pulse shaper connected to the mains transformer secondary, three flip-flops for frequency dividing and an output power stage tuned to 50 Hz. The necessary power supply components are also located on the printed circuit board.

5.4.4. Power supply

The power supply regulators of the DVM were fed by a single phase 115 V, 60 Hz mains transformer. This transformer is replaced by a smaller three phase 115 V, 400 Hz transformer. Also the filter capacitors are replaced with high quality electrolytic types with less capacity, 1000 µF instead of 4000 µF . Minor changes had to be made on the regulator board in order to maintain stability with low environmental temperature.

5.4.5. Display

A projection type in-line display was applied in the DVM. This display is removed from the DVM housing. A smaller type display mounted on the operators panel is connected instead, through a connector provided on the rear of the DVM.

5.5. Parallel to series converter

The parallel to series converter (PSO) required in the data collection system performed the task of a systems programmer. This is because it not only serializes the digital DVM data but also generates input commands for several parts of the instrumentation system. These commands are time related and derived from the reference clock. A functional block diagram of the PSO is given in fig. 48, complete scheme is presented in fig. 49.

5.5.1. Operation of the parallel to series converter

Every 2.5 msec when the conversion complete puls from the DVM is true, the 16 bit word is loaded into a register. During recording, however, each group of 16 bits is preceded by a characteristic mark of 9 bits. Consequently one complete measurement contains 25 bits. The characteristic mark needs some explanation.

The 9 bits are divided into two groups. The first 6 bits are fixed to one "0" and five "1" s, thus: 011111. The three bits of the second group are either "0" or "1".

The five "1" s on a row produce the recognizable pattern, because in the data in BCD code never five times "1" on a row can occur. The three bits of the second group contain information regarding the contents of the following 16 bits of information. So each of the 20 measurements forming one cycle of 50 msec duration contains a characteristic mark and measurement data. Because 19 analog channels are recorded, the 20th time slot is available for administrative data, consisting of the cycle number of a manual selected manoeuvre number. Selection of this number is performed by two thumbwheel switches on the operators panel. The BCD code (8 bits) is loaded into the register every 100 msec. The cycle number is generated by the cycle counter producing 16 bits BCD coded information, also loaded into the shift register every alternate 100 msec. Consequently manoeuvre number and cycle number are loaded alternately in the register resulting in the recording of odd cycle numbers and of recording manoeuvre number when the cycle number is even. To distinguish between cycle number and manoeuvre number information, the last group of 3 bits of the characteristic mark are used as follows

characteristic mark		meaning of the 16 bits of recorded data
first 6 bits	last 3 bits	
011111	111	cycle number
011111	110	manoeuvre number
011111	101	zero input data
011111	100	zero pressure input data
011111	011	measurement data

As mentioned earlier in the description of the pressure transducer box and of the analog filters, provisions are made to measure zero offset of the pressure transducers and of the analog filters. The different modes of measurement are selected by a thumbwheel switch on the operators panel. The actual selected mode is identified in the characteristic mark as can be seen in the foregoing table.

The information loaded in the shift register is shifted in serial form to a divider circuit under control of the 10 kc clock. This clock is inhibited during 6 clock periods to produce the first 6 bits of the characteristic mark. In the divider circuit the signal path is splitted into two outputs, output M_0 changes when the input bit is "0", whereas M_1 changes when the input bit is "1". In this way there is always a change either in M_0 or in M_1. Two current driver amplifiers connected to M_1 and M_0 supply the write head coils of the tape recorder.

5.5.2. Recorder start/stop command

Whenever a start command is generated with the recorder switch on the operators panel, the recorder starts running immediately although without recording, the signals M_0 and M_1 being inhibited. After a certain time, adjustable with jumpers and set to 3 sec, the recording process is started with a characteristic mark followed by the cycle counter contents.

The delay time of 3 sec is required by the recorder to attain its normal operation speed. When a stop command is generated, the recorder keeps on running until the cycle is completed and M_0 and M_1 are again inhibited.

5.5.3. Control circuits

During the recording process no parity checks are carried out and no channel numbers are recorded. Some faulty conditions however are detected. Whenever a negative analog signal appears at the input of the DVM, the information is useless. The negative sign of the DVM is used to set a flip-flop which turns on an indicator lamp on the operator's panel.
When a positive signal greater then the allowed 9999 mV appears at the input, both the "8" bit and the "2" bit of the most significant decade are turned on. Those two bits are gated and fed to the lamp drive flip-flop. A check is made by comparison of the conversion complete pulse of the DVM and the clear pulse of the shift register. As there are no additional latch circuits in the DVM, only reliable information is obtained when the conversion process is finished, which is indicated by the conversion complete signal. The output of this comparison circuit is used to light a second indication lamp on the operator's panel. Both check flip-flops can be reset by a single push button switch.
A 1 Hz signal is derived from the cycle counter when the recording mode is selected and the cycle counter is cleared. This signal is used to increment an electromechanical counter on the operator's panel. Each count represents 20 cycles of recorded information so the indicated number is the total duration of the recording in seconds.

5.5.4. Ouput signals

The analog switches of the multiplexer are controlled by 19 output commands from the PSO. The control pulses of 2 msec duration are checked by a "watch dog" circuit with a preset time of 2.1 msec to prevent short circuiting of analog switches in the case of a malfunction in the timing circuits. The 19 outputs are of standard TTL level, properly interfaced in the multiplexer.
The control pulse for the sample/hold circuit, located in the DVM has a duration of 1.1 msec. This TTL level pulse generated every 2.5 msec, independent of the mode selected, is interfaced in the PSO to the required level, 0 V-hold and 6 V-sample. The conversion start pulse (50 μsec) generated also every 2.5 msec, is delayed 50 μsec after the sample/hold command, to cope with the turn-off time and aperture uncertainty of the sample/hold circuit. As already mentioned in the description of the pitch-rate gyro, a 400 Hz precision square wave signal is derived from the reference clock. This TTL level signal is interfaced in the analog signal conditioning box on printed circuit A5.

5.5.5. Construction

All logic circuits used are TTL series 5400. The compnents are located on 5 printed circuit boards placed in slotted metal case. A mother board provides the necessary cross connections between the boards. The box measures 20 x 20 x 6 cm^3 and is provided with sockets for connection to the DVM, the multiplexer, the operators panel the tape recorder and the power supply. The required power supply voltages 5 V, ± 15 V and 22 V are derived from the "digital power supply".

The interconnection of the PSO with peripheral parts is shown in fig. 50.

5.6. Tape recorder

The tape recorder used in the instrumentation system is a NAGRA IV D. This recorder, shown in fig. 51, designed to operate within specifications at low temperature is however an audiorecorder. The servo tape moving system proved to be highly accurate under adverse conditions.
The digital information available in series form from the PSO is recorded with the "zeros" on the upper track and the "ones" on the lower track.
Standard triple play ¼" tape is used on 6" reels allowing for an effective recording time of approximately

half an hour. A tape speed of 15" per sec resulted in a bit density of 667 bits per inch. The recorder is powered by 2.5 Ah nicad batteries carefully recharged after each flight. The internal full track play-back amplifier is used to drive a small indicator mounted on the operators panel. This indicator proved to be very useful for the observer during flight in giving a final indication of the recording process.
The recorder is placed on a tray and was easily put in the instrumentation pod through the access door, fig. 52.
Two connectors are provided, one for the digital input and indicator output and one for the start/stop command of the tape transport system.

5.7. Analog signal conditioning box

In the analog signal conditioning box several different circuits are placed. Some of them have been already mentioned in the transducer chapter.
The complete wiring diagram of the box with the various interface boards is given in fig. 53.
The circuits placed in the box are:
- Recorder operation control.
- Power supply for accelerometers.
- Engine speed measurement interface.
- Total temperature measurement interface.
- Reference clock interface.
- Heading change measurement interface.
- Angle of attack measurement interface.
- D.c. power supply ±15 V.

5.7.1. Recorder operation control

The function of this circuit is to provide an easy means for the operator to select the various recording modes of the system. The recorder on/off switch and the mode selector switch are mounted on the operator panel.
The four possible recording modes are:
- Balance (zero filter input).
- Zero pressure (short circuit of pneumatics).
- Manoeuvre (recording of manoeuvre data).
- Automatic (Balance and Zero in sequence).

The circuit, fig. 54, is realized with relais and two simple timing networks. The operation of the circuit is as follows, with reference to fig. 55.
When the recorder on/off switch is placed in the "on" position a re corder start command is fed to the parallel to series converter.
Dependent on the position of the mode selector, balance data, zero pressure data or manoeuvre data is recorded. With the mode selector switch the position of the pneumatic valves and of the balance relais is determined. In addition the last three bits in the characteristic mark, generated in the parallel to series converter are set.
When the momentary recorder switch is pushed a recorder start pulse is generated with a duration time of 5 sec, yielding a recording time of 2 sec, due to the start up time of the recorder. If the mode selector is in the "automatic" position two recorder start commands are generated in sequence. During the first recording time "balance data" is recorded and during the second recording time "zero pressure data" is recorded.
The pneumatic valves, in the pressure transducer box, are always in the "zero pressure" position except when the "manoeuvre" mode is selected and the recorder switch is placed in the "on" position.

5.7.2. Accelerometer power supply

Accelerometer power supply voltages ± 15 V and ± 28 V are made with voltage regulators, type LM305, for the positive voltages and LM304 for the negative voltages. The circuit diagram is in fig. 56. The circuits operate in the current limiting mode, hence are short circuit proof. Supply ground is not connected to signal reference ground in the signal conditioning box, but to the signal returns in the accelerometer box, for relatively long leads are necessary to connect the power supply with the accelerometerbox dorsally mounted on the aircraft.

5.7.3. Engine speed, measurement interface

This signal conditioning unit, a frequency to voltage converter is already described in 4.4. of the transducer chapter.

5.7.4. Total temperature measurement interface

This signal conditioning circuit, a high gain differential amplifier and bridge supply is described in 4.5. of the transducer chapter.

5.7.5. Comparator 40 kc clock

A differential comparator LM211 with high input impedance is used to "isolate" digital and analog ground in the analog signal conditioning box. The scheme is given in fig. 57. The 40 kc signal is used for the engine speed frequency to voltage converter. The printed circuit board also contains some components used for the heading change measurement to be described next.

5.7.6. Heading change measurement interface

The synchro of the Sperry S3A directional gyro is demodulated with a synchro to d c converter. This converter is of a design based on the well known principle of the R-C phase-shift converter. The converter, the scheme given in fig. 58, accepts the three wire output of a 26 V, 400 Hz synchro and produces an output from 0 to 10 V for 180° of synchro armature rotation.

The synchro stator is connected to a triple differential input stage Q_1, Q_2 and Q_3. Outputs of Q_1 and Q_3 are connected to two tuned R-C networks, which produce 45° phase-shift at 400 Hz. The buffered outputs of Q_4 and Q_{12} are subtracted from a fraction of the Q_2 output. Outputs of Q_5 and Q_{13} are 400 Hz sin waves with constant amplitude independent of synchro armature rotation. Relative phase-difference however is twice the rotation angle of the synchro. Two 2nd order low pass filter Q6 and Q8 reduce unwanted high frequency contents of the signals. Comparators Q_7 and Q_9, switching at the zero crossings of the inputs produce two square waves, signals p and q, used to set and reset a flip-flop. The set time of this flip-flop is proportional to synchro input angle. An analog switch, Q_{14} and Q_{15}, fed by the 10V reference voltage and controlled by the flip-flop output produces an output square wave with constant amplitude. Low-pass filtering with Q_{10} and Q_{11} results in a d c voltage proportional to synchro input angle. Due to the application of two R-C networks twice full scale output is attained for each revolution of the synchro, hence a 180° anomaly is introduced by the converter.

To prevent fast signal changes from 10 V to zero, or from zero to 10 V at 180° and 360°, the slaving torque motor of the gyro is used in the following way, see fig. 57 and fig. 59.

The converter output signal is filtered and buffered with Q_{12}. Comparator Q_{13}, with hysteresis, switches around a -5 V level, corresponding to 90° or 270° of synchro output angle. The comparator controls R18 with the output stage Q_5. Contacts of R18 change the phase of the voltage applied to the control winding of the slaving torque motor, the fixed phase being energized permanently. So when the instrumentation

system is switched on, the gyro being in any arbitrary position, starts slaving to either the 90° or 270° position. Possible output signals are shown in fig. 60a and b. Consequently during flight the gyro is forced to maintain its position relative to the aircraft. During the recording of measurement data, the switched torque motor supply is interrupted by R19 controlled by the recorder on/off switch. Henceforth, during recording the directional gyro is a free gyro and heading change measurement always starts approximately from the 90° or 270° position.
The synchro to d c converter is placed in slot A6 or the analog signal conditioning box, the comparator circuit in slot A5.

5.7.7. Angle of attack measurement interface

This interface A6 is equal to the heading change measurement interface. Full scale output of the synchro to d c converter is attained by 180° of synchro rotation. Due to the 2 to 1 gear ratio in the transducer, the vane input angle is limited to ±45°. The anmomaly introduced, four times zero output for one revolution, causes no problem for angle of attack measurement.

5.7.8. D c power supply

To supply various circuits in the box a ± 15 V, 100 mA readily available power supply type C15-1D from Delta Electronics is used. The supply board is placed in slot A10 and connected to a 115 V, 400 Hz mains transformer.

5.8. Operators control panel

Switches and displays required for operation and monitoring of the instrumentation system are mounted on a small panel, installed in the cockpit on starboard side. Fig. 61 shows the panel installed.
In the following a summary is given of the different components and their purposes, in fig. 62 the complete scheme of the panel is shown.

5.8.1. Manual channel selector

With a 24 position switch, each of the 19 analog channels, the multiplexer output or the 10 V reference voltage can be selected. The analog value of the selected signal is digitized at a rate of 400 measurements per sec and displayed.

5.8.2. Display unit

The display unit applied, a one plane projection type read out, is connected to the decimal outputs of the DVM. A ± sign indicator and a 5-digits display are installed on the panel.

5.8.3. Pressure transducer box mode indicator

An electromagnetic indicator is used to display the position of the pneumatic valves in the pressure transducer box. The indicator is activated by micro switches which sense the actual position of the rotary valves. Both "end" positions "measure" and "zero pressure" are active so even a power failure or a jamming valve is indicated by the passive mid-position of the indicator.

5.8.4. Record mode selector and manoeuvre counter

The mode slector and the manoeuvre counter are part of an assembly of three thumbwheel switches. Two are used to generate, in BCD code, the manoeuvre number to be recorded. The third switch is the mode selector with the positions:

- balance
- zero pressure
- manoeuvre
- automatic.

With this switch the appropriate relays of the recorder operation control are activated.

5.8.5. Cycle counter

A resettable electromechanical counter is applied to display the number of cycles recorded. Actually the counter is an elapsed time indicator, for it is connected to the 1 sec pulse output of the parallel to series converter. Each count on the indicator equals 20 cycles of recorded data.

5.8.6. Modulation indicator

This indicator is a small movingcoil instrument connected to the rectified output of the play-back amplifier of the NAGRA recorder.
Both, upper and lower, channels are read by a full track head, nevertheless a single track failure is easily indicated by this simple device.

5.8.7. Recorder switches

Recorder on/off is performed with a toggle switch for continuous operation and a push-button switch for momentary operation. The latter activates the recorder in the modes "balance", "zero pressure" and "automatic". A small green light is "on" when the recorder is switched in "run" by the recorder relay of the recorder operation control circuit.

5.8.8. Error indicator

As mentioned in the description of the parallel to series converter, three error conditions are detected. The indication on the panel is with two small red lights. A push-button switch is provided to reset the error detection circuits in the converter.

5.8.9. Signal lights, switches and connectors

A set of four white lights is mounted to show the operation of the temperature regulators of the accelerometer box, the pitch rate gyro, the pressure transducer box and the altimeter (static pressure) box. An amber "press to test" light is connected to the aircraft's 28 V bus. The instrumentation system's master switch, which activates the master relay is also located on the panel. Another, spring activated, switch is provided for the release of the "mouse trap" holding the trailing cone tube during take off.
Furthermore, for ease of installation, 5 sockets are provided, shown in the scheme presented in fig. 62, for the connection with the power supply, the DVM, the parallel to series converter, the analog signal conditioning box and the multiplexer.

5.9. Power supplies

Three main power supplies are used in the instrumentation systems.

5.9.1. Reference supply

A reference buffer supply is required because the total current drawn from the 10 V reference is approximately 40 mA. This is mainly caused by the low resistance values of the yaw-rate gyro potentiometer and the altimeter potentiometer, 800 resp. 450 ohm. Without a reference supply or a reference sensepoint close to these two units, at least An-10 wires should have been required to deal with resistance variations due to temperature changes. The possibility of a sensepoint is considered, however not applied for rather long (~ 15 m) sense wires would be required, next to the problem of the common ground of the system. Instead a simple reference buffer amplifier, see fig. 63, with its own power supply regulator is applied.
Consequently the 10 V reference point is not the reference unit in the filterbox, but a post of a terminal strip mounted in the radio compartment of the aircraft. To this point equipment is connected requiring 10 V reference.

5.9.2. Dual 15 V supply

This power supply is used only for the amplifiers in the filter box. The unit consists of two complementrary 15 V series regulators, fig. 64. The maximum available current is 2A at 15 V from each supply. Foldback current charateristics applied not only make the outputs short circuit proof but also lower the output voltage during possible overload. Both circuits are coupled in order to achieve a balanced output in any circumstance. Remote sensing of the voltage at the junction strip in the filter box is applied. The mains transformer is three phase 115 V, 400 Hz for easy ripple free rectification. A and C phase are protected with fuses.

5.9.3. 5 V power supply

This power supply provides 5V, ±15 V and 22 V for the parallel to series converter and 5 V, 6 V, 12 V and -15 V for the multiplexer. Output voltages are series regulated except for the 22 V which is only used for relay power.The scheme of the power supply is shown in fig. 65.
Foldback current charateristic is applied for the 5 V digital supply together with an SCR controlled "crowbar" overvoltage protection circuit. The maximum output current equals 1.6 A. This power supply also has a three phase mains transformer, provided with different secundary windings supplying the regulator circuits. A and C phase are fuse protected.

6. INSTALLATION IN THE AIRCRAFT

The parts of the instrumentation system placed in the underwing pod are from front to rear: the recorder, the power supplies, the multiplexer, the parallel to series converter, the analog to digital converter and the analog signal conditioning box. Fig. 66 shows the instrumentation drawer.
Terminal strips are mounted on the bottom side of the drawer and provided for interconnection of the different parts. The wiring diagram of the pod is given in fig. 67. From the terminals a cable harness is conducted to the top connector just in front of the mounting pylon. This connecter is the only electrical connection with the aircraft. In the aircraft the cabling is conducted through the leading edge of the wing to the fuselage. The harness is terminated on several 58 pin MRA-50 connectors in the radio compartment. From this compartment connections are made, through the pressure bulkhead, to the operators panel in the cockpit. In the radio compartment are furthermore installed the vertical gyro, the directional gyro, the altimeter (pressure transducer) with servo amplifier and the reference buffer. The complete aircraft wiring diagram is given in fig. 68. The pressure transducer box is placed in the starboard ammunition bay. In this bay also provisions are made for the connections of the pneumatic tubing from the trailing cone probe, the exhaust pitot and static probes, the nose pitot tube and the wing static- and pitot tube.
Under the spline fairing are placed the accelerometer box, the pitch and yaw rate gyro's and the pitch rate amplifier demodulator.
The required electrical power for the instrumentation system is derived from a 750 VA, 115 V, 400 Hz three phase rotary inverter, Leland MGH-182-100, installed underneath the ammunition box. Power distribution strips, power relay and circuit breakers are placed in the ammunition compartment on port side.
Dependent on the type of flight either the angle of attack transducer or the pitot tube is installed on the nose boom.
Locations of system parts are given in fig. 69..
Before the functional testflight was performed the following activities are carried out:

- Calibration of the elevator- and of the stabilizer-deflection transducer.
- Calibration of the angle of attack transducer.
- Alignment of the accelerometer box.
- Measurement of dynamic characteristics of the pneumatic system.

6.1. Calibration of the elevator- and of the stabilizer deflection transducer

Elevator deflection transducer calibration was carried out with a precision inclinometer (Optical Winkel Libelle, Miller) placed on a mounting frame attached to the left elevator half, perpendicular to the rotation axis of this elevator half. Because elevator deflection is measured relative to the horizontal stabilizer it was necessary to trim the stabilizer in horizontal position. To this end the "Hunter" was jacked horizontally with the X- and Y-reference axes. The reference datum points in the wheel bay were used together with a long fluid filled tube held against wing tip reference points. The zero position of the elevator was established with a three point precision template placed on marked points of the horizontal stabilizer. As already mentioned in the transducer chapter, the play in the control mechanism showed to be significant, about 0.08°. Calibration points were adjusted with a special screw-jack placed on the ground and hooked up to elevator trailing-edge.
The stabilizer deflection transducer was calibrated with the same inclinometer, now put perpendicular to the stabilizer rotation axis. With a precision clamp the elevator half with the inclinometer was rigged to the stabilizer. Calibration points now were adjusted, using the electrical trim actuator, to this end the elevator push-rod had to be removed.

6.2. Calibration of the angle of attack transducer

Because angle of attack measurement was added after the calibration of transducers and instrumentation system in the laboratory was carried out, the angle of attack transducer had to be calibrated installed on the nose boom. The calibration was performed with a small inclinometer,attached to a framework, which was clamped directly to the vane shaft. The transducer housing had to be fixed rigidly during calibration, to exclude uncertainties due to bending of the approximately 2 m long nose boom. This method however did not prove to be satisfactory, so more calibrations were carried out with the transducer removed from the aircraft, with the aid of an extender cable between the nose boom connector and the transducer.

6.3. Alignment of the accelerometer

Alignment of the accelerometer box with reference to the X_B-Y_B plane of the aircraft was no problem and was carried out with an inclinometer, the aircraft being jacked horizontally. Alignment of the X_d- and Y_d-reference axes (rotation around the Z_d-axis of the accelerometer box was carried out with an optical aiming device. The following procedure, see fig. 70, was executed.
The aircraft was jacked horizontally with the X_B- and Y_B-axes. After removal of a part of the spline fairing a straight ruler was attached to the accelerometer box in alignment with the Y_d-axis of the box. The aiming device placed at some distance was aligned with the ruler and point P was marked on the hangar wall. Next the aiming device was placed with its rotation axis just over the center of the accelerometer box. Mark P was displaced over 90° to point Q, also marked on the wall. Furthermore a straight line was drawn on the floor parallel to the aircraft's X_B-axis. This line, precisely drawn under the aircraft's centerline was extended to the wall and point R was marked. Consequently the horizontal distance between Q and R devided by the distance from the accelerometer box to point R, equaled the misalignment angle φ_D (rad) of the accelerometer box.
Measurements carried out resulted in a φ_D of 0.0017 rad.
No further attempts have been made to realign the box, necessary correction is taken in account during flight test data processing.

6.4. Measurement of dynamic characteristics of the pneumatic system

The pressure transducers of the instrumentation system were used to determine the dynamic characteristics of the pneumatic systems. Each pneumatic system consisted of the measurement probe, the connecting tube and the transducer located in the pressure transducer box. The step response measurements were carried out as follows, see also Fig. 71. The particular measurement probe was closed with a special piece of equipment allowing for connection of a suction/pressure pump. A small piece of glass tube being part of the connection could be closed by a valve. With the pump system pressure was applied to about 80% of the range of the transducer, then the valve was closed and the glass tube was broken. The response to the step change of pressure was displayed on a memory scope. The time base trigger was taken, from a simple circuit, at the moment of glass tube destruction.
Measurements were carried out only under sealevel conditions, change of absolute pressure and of temperature were not taken into account.
Some results are shown in fig. 72.
The dynamic characteristics are approximated with a first order system and a time delay.

7. CONCLUDING REMARKS

With the instrumentation system described, the validity of the flight test method to derive performance, stability and control characteristics from measurements in non steady flights for high performance aircraft is demonstrated.

Some remarks concerning the instrumentation however have to be made:

- The precision of the elevator deflection measurement was degradated by the play in the control assembly far more than was expected. A possible solution would have been the use of two deflection transducers, connected directly to the elevator halves and positioned in the horizontal stabilizer.
- The use of the static pressure change measurement system as described is no longer justified by the improvement of available high performance absolute pressure transducers.
- The measurement of zero input related output of differential pressure transducers improves the obtainable precision.
- Acceleration sensitive pressure transducers should be placed in the most favourable direction.
- Accuraccy (stability) of transducers is improved by stabilizing the environmental conditions.
- Extensive leak tests have to be carried out on pneumatic systems, especially on those which connect probes located in the engine exhaust.
- Delay times in pneumatic systems should be measured not only at sea level preessure, but also on lower absolute pressure levels.
- Although final calibration of the angle of attack transducer is carried out during data processing, the static calibration (scale factor and zero-reference angle) should be carried out with maximum achievable accuracy.
- Increasing the sample frequency of the data collection system, would allow for a higher filter cut-off frequency, resulting in a more linear phase characteristic in the frequency range of interest.
- The implementation of digital filters in onboard systems, with more precisely controlled filter characteristics should be investigated.

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Table 1. Variables measured with the instrumentation system.

channel number	measured variable	
1	q	rate of pitch
2	A_x	specific force along X-axis
3	A_y	specific force along Y-axis
4	A_z	specific force along Z-axis
5	ψ	change in heading
6	r	rate of yaw
7	n	engine speed
8	T	total temperature
9	Δp_4	cone static pressure - reference pressure
10	Δp_5	nose static pressure - reference pressure
11a	Δp_6	cone static pressure - nose static pressure
11b	EGT	exhaust gas temperature
12	Δp_1	vessel pressure - reference pressure
13a	α	angle of attack
13b	Δp_{s_j}	exhaust static pressure - reference pressure
14	Δp_{t_j}	exhaust total pressure - reference pressure
15	δ_e	elevator deflection angle
16	p_{ref}	reference pressure (absolute)
17	q_c	total pressure - reference pressure
18	φ	angle of bank
19	i_h	stabilizer trim angle

Table 2. Desired and achieved accuracies (% RMS).

Channel number	Measured variable	Desired accuracy	Achieved accuracy	
			not corrected for zero shift	corrected for zero shift
1	q	0.022	-	0.013
2	A_x	0.02	-	0.013
3	A_y	0.02	-	0.015
4	A_z	0.02	-	0.012
5	ψ	0.55	-	0.06
6	r	0.5	-	0.3
7	n	0.1	-	0.015
8	T	0.1	-	0.04*
9	Δp_4	0.5	0.3	0.24
10	Δp_5	0.5	0.2	0.12
11a	Δp_6	0.5	0.3	0.17
11b	EGT	0.5	-	0.35
12	Δp_1	0.5	0.23	0.07
13a	α	0.5	-	0.32
13b	Δp_{s_J}	0.5	0.11	0.08
14	Δp_{t_J}	0.5	0.13	0.12
15	δ_e	0.3	-	0.26
16	P_{ref}	0.2	-	0.05
17	q_c	0.5	0.11	0.11
18	φ	0.5	-	0.15
19	i_h	1.0	-	0.78

* without the dual 50Ω p_t sensor.

Table 3. Inertial transducers types and ranges.

channel number	measured variable	transducer type	transducer range
1	q	Honeywell GG87B7	± 23°/sec
2	A_x	Donner 4810	± 10 m/sec^2
3	A_y	Donner 4310	± 5 m/sec^2
4	A_z	Donner 4810	± 100 m/sec^2
5	ψ	Sperry S3A	360°
6	r	SFIM J14	± 7°/sec
18	φ	Sperry HGU-B	± 90°

Table 4. Pressure transducers types and ranges.

channel number	measured variable	transducer type	transducer range
9	Δp_4	A.C.B. H5010	± 200 kgf/m^2
10	Δp_5	A.C.B. H5010	± 200 kgf/m^2
11a	Δp_6	A.C.B. H5010	± 200 kgf/m^2
12	Δp_1	Statham PM6TC	± 700 kgf/m^2
13b	Δp_{s_j}	Statham PM6TC	± 7000 kgf/m^2
14	Δp_{t_j}	Statham PM6TC	± 17,500 kgf/m^2
16	p_{ref}	Kelvin HKTG1902	11000 kgf/m^2
17	q_c	Statham PM6TC	± 7000 kgf/m^2

Table 5. Position/deflection transducers.

channel number	measured variable	transducer type	transducer range
13a	α	synchro 7E911-1	± 45°
15	δ_e	potentiometer,	-9 to +21°
19	i_h	Beckman 6603	± 2.5°

Table 6. Engine speed and temperature transducers.

channel number	measured variable	transducer type	transducer range
7	n	a.c. generator	8150 RPM
8	T	Rosemount 102W2W	-100 to +200°C
11b	EGT	B and H 187A-80	0 to 1200°C

Table 7. Typical integrating gyro characteristics Honeywell GG87B7.

Angular momentum	1×10^5	grf cm^2/sec
Damping coefficient	333,000	dyne cm sec/rad
Polar moment of gimbal inertia	200	grf cm^2
Characteristic time	0.6	m sec
Gyro transfer function	3.75	V/rad
Sensitivity (pick off)	12.5	V/rad
Gyro gain	0.3	rad/rad
Sensitivity (torquer)	825	°/hr/mA
Maximum torquing rate	82500	°/hr
Operating temperature	355	K
Spin motor excitation	25/400	V/Hz

FIG. 1: HAWKER HUNTER MK VII WITH TRAILING CONE TUBE.

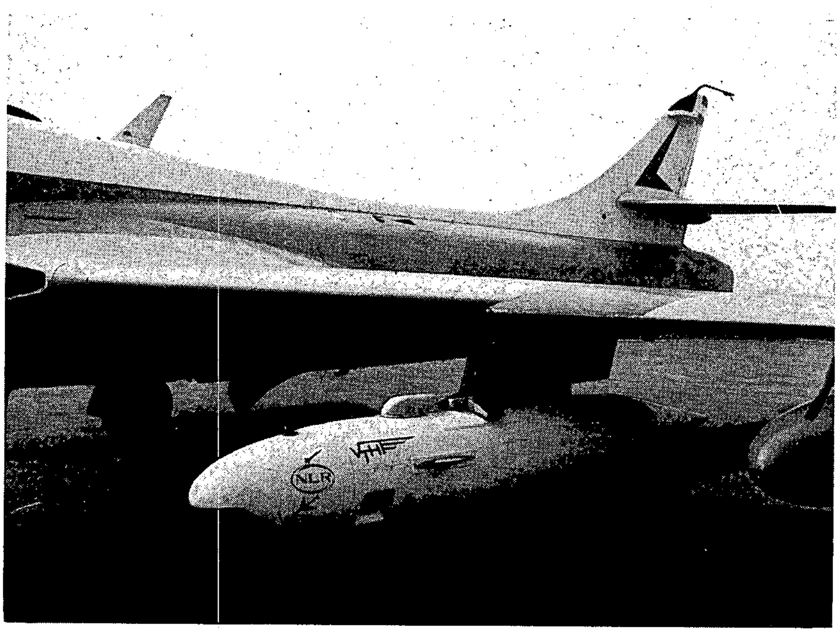
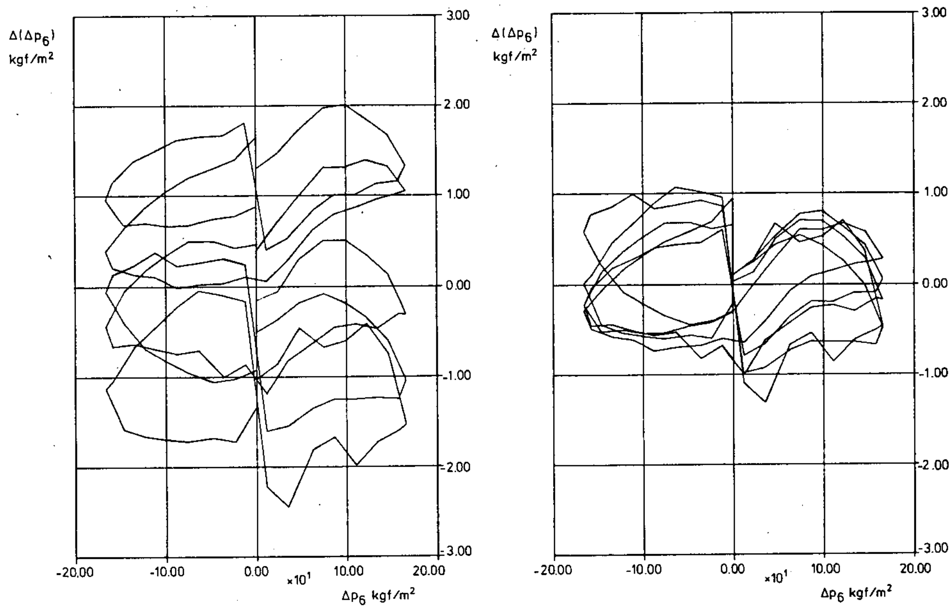


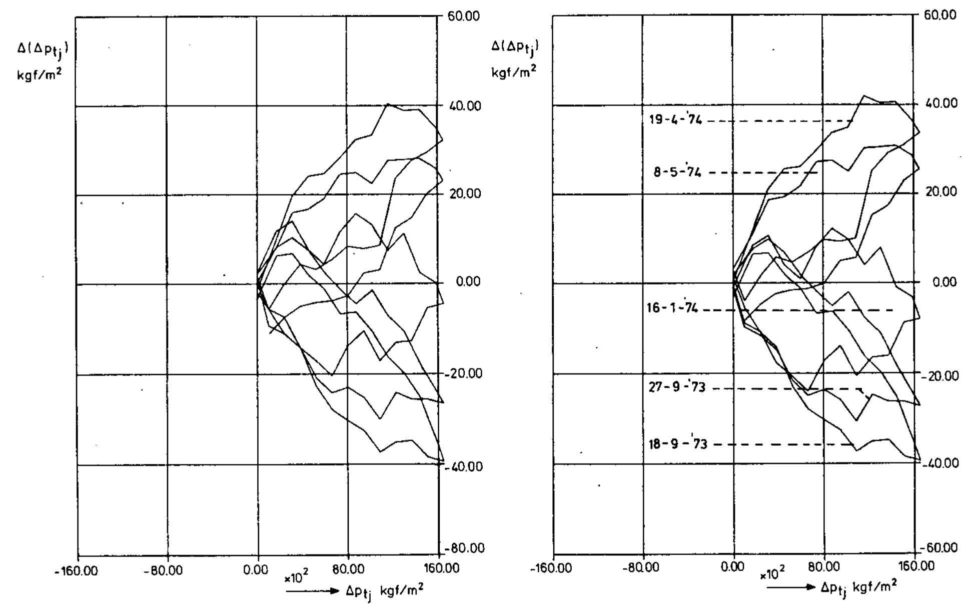
FIG. 2: HAWKER HUNTER MK VII WITH INSTRUMENTATION POD.



a. NOT CORRECTED FOR ZEROSHIFT

b. CORRECTED FOR ZEROSHIFT

FIG. 3: DEVIATIONS OF FIVE COMBINED CALIBRATIONS OF A PRESSURE TRANSDUCER (Δp_6)



a. NOT CORRECTED FOR ZEROSHIFT

b. CORRECTED FOR ZEROSHIFT

FIG. 4: DEVIATIONS OF FIVE COMBINED CALIBRATIONS OF A PRESSURE TRANSDUCER (Δp_{t_j})

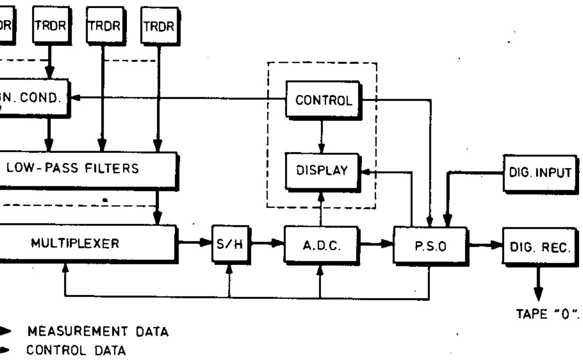


FIG. 5 : GENERAL ARRANGEMENT OF THE INSTRUMENTATIONSYSTEM.

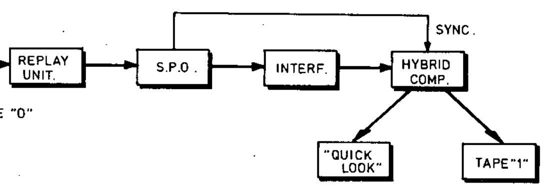


FIG. 6 : GENERAL ARRANGEMENT OF THE GROUND EQUIPMENT.

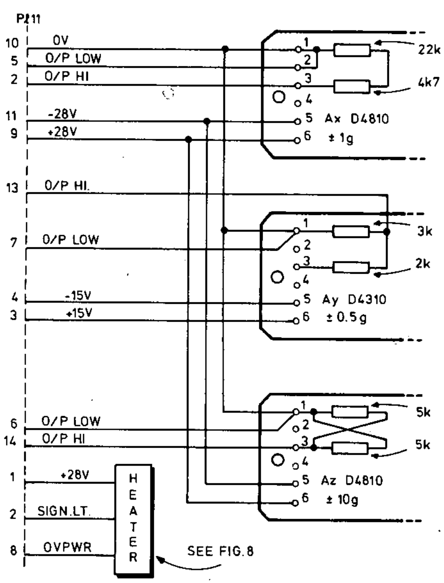


FIG. 9 : CONNECTION SCHEME OF ACCELEROMETERS.

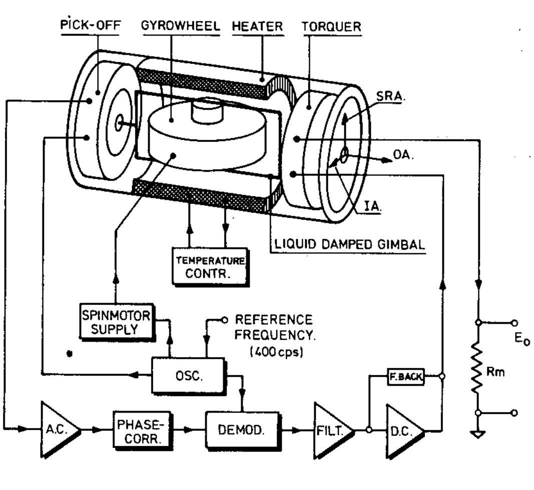


FIG.10: PRINCIPLE OF PITCH RATE MEASUREMENT.

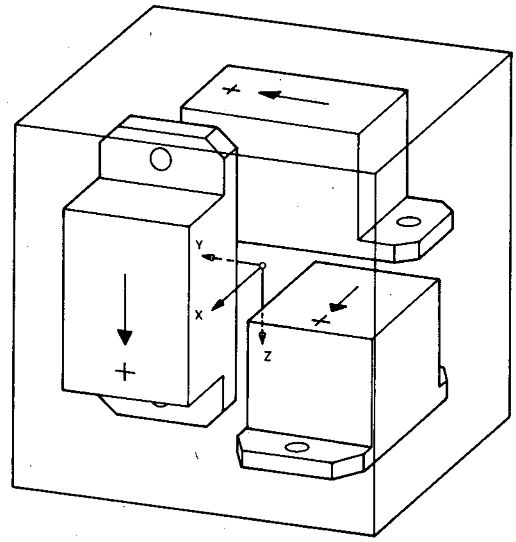


FIG. 7 : ARRANGEMENT OF THE THREE ACCELEROMETERS.

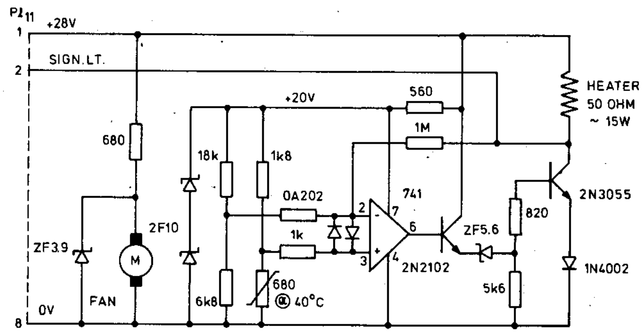


FIG. 8 : TEMPERATURE REGULATOR ACCELEROMETERBOX.

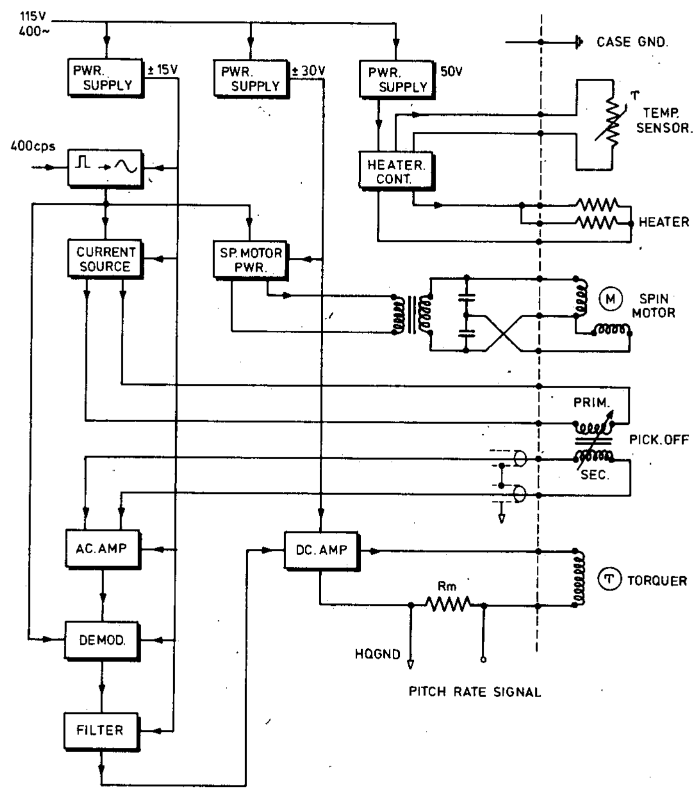
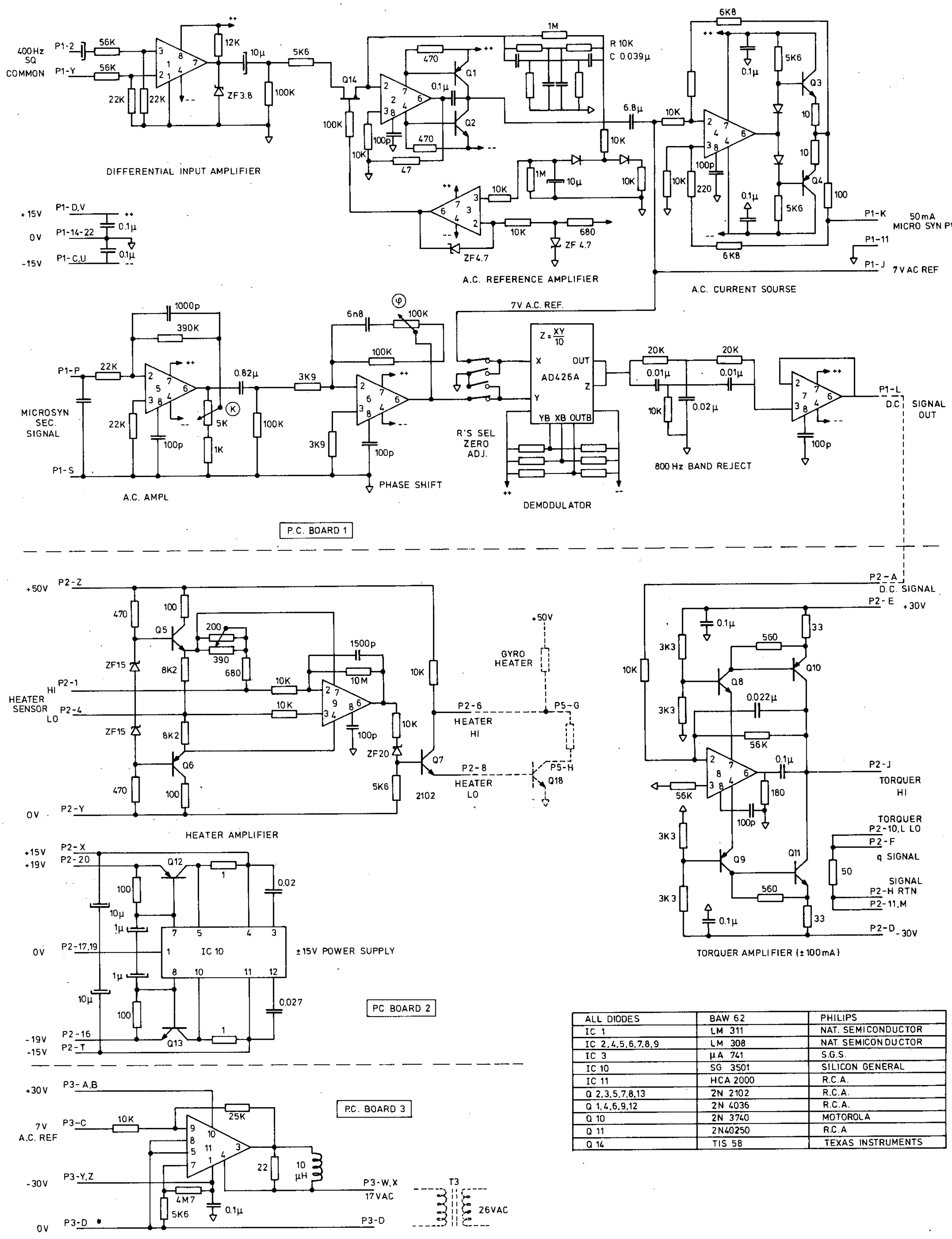
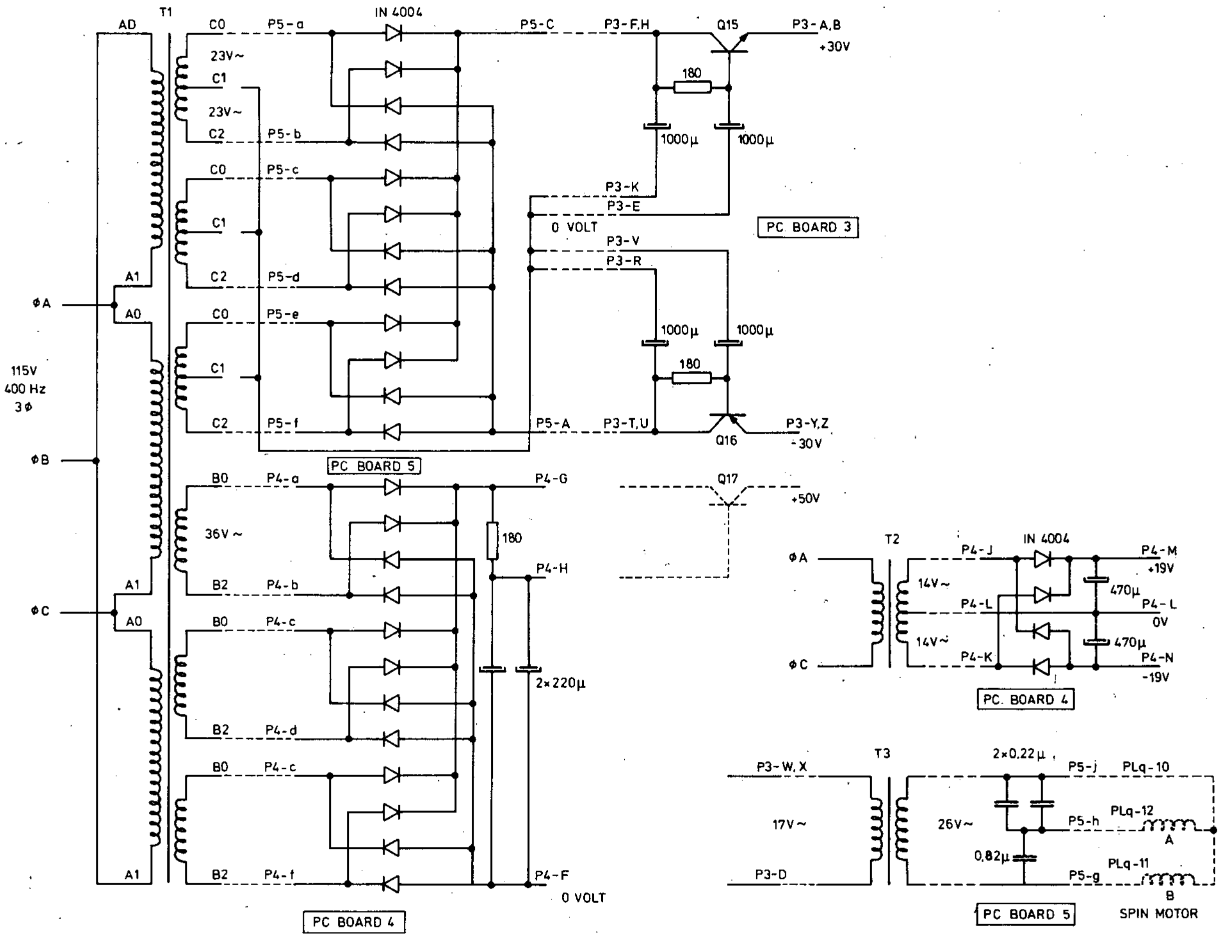


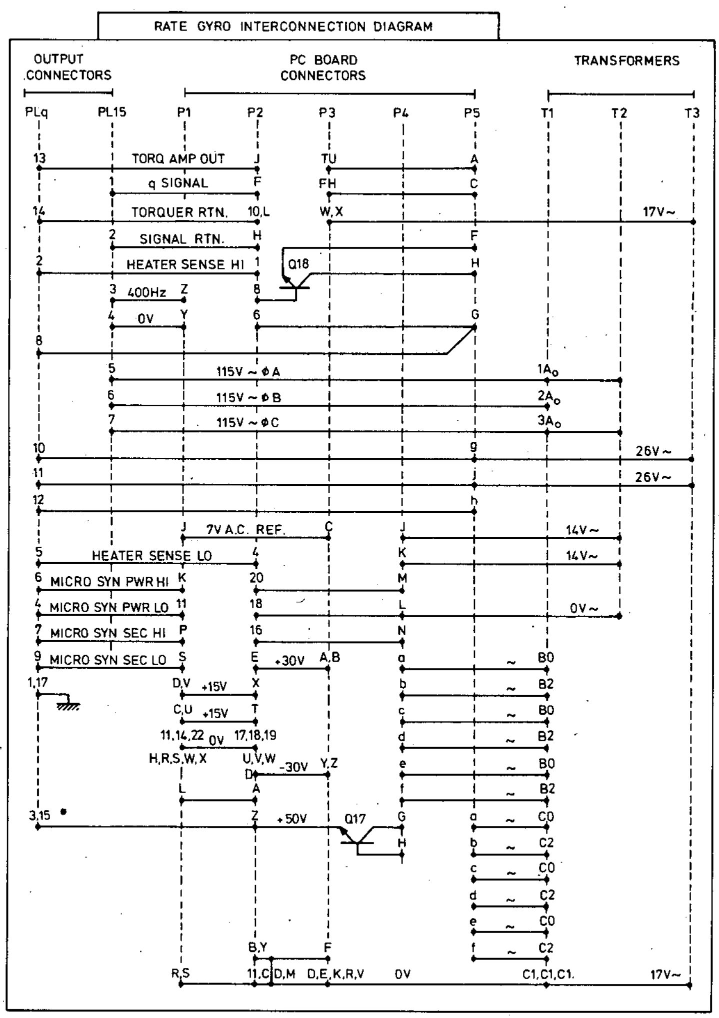
FIG. 11 : FUNCTIONAL DIAGRAM OF PITCHRATE GYRO ELECTRONICS.

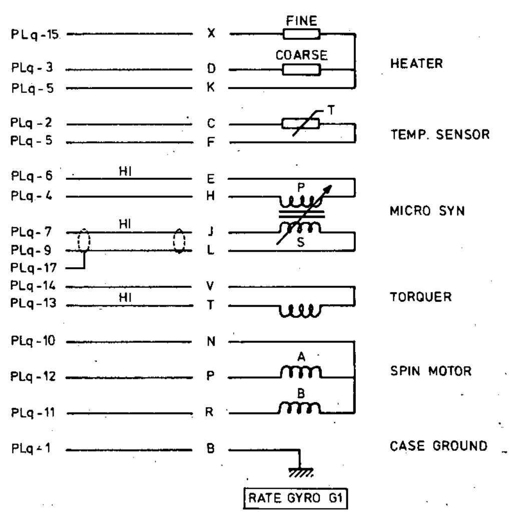


ALL DIODES	BAW 62	PHILIPS
IC 1	LM 311	NAT. SEMICONDUCTOR
IC 2,4,5,6,7,8,9	LM 308	NAT. SEMICONDUCTOR
IC 3	μA 741	S.G.S.
IC 10	SG 3501	SILICON GENERAL
IC 11	HCA 2000	R.C.A.
Q 2,3,5,7,8,13	2N 2102	R.C.A.
Q 1,4,6,9,12	2N 4036	R.C.A.
Q 10	2N 3740	MOTOROLA
Q 11	2N40250	R.C.A
Q 14	TIS 58	TEXAS INSTRUMENTS

FIG. 12a : RATE GYRO ELECTRONICS.







Q15,17	2N40250	R.C.A.
Q16	2N3740	MOTOROLA
Q18	2N3055	R.C.A.
G1	MINI RIG GG87B7	HONEYWELL

FIG. 12b: PITCH RATE GYRO ELECTRONICS.

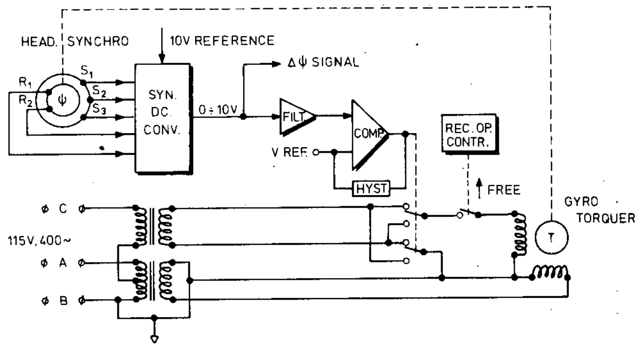


FIG. 13 : PRINCIPLE OF HEADING CHANGE MEASUREMENT.

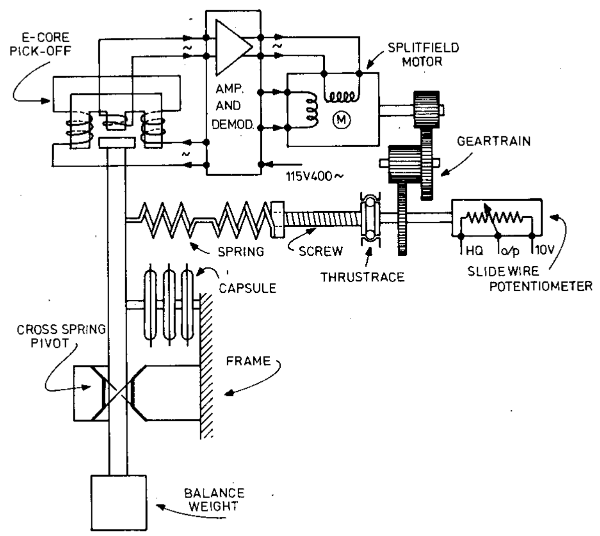


FIG. 14 : SCHEMATIC DIAGRAM FORCE BALANCE ABSOLUTE PRESSURE TRANSDUCER.

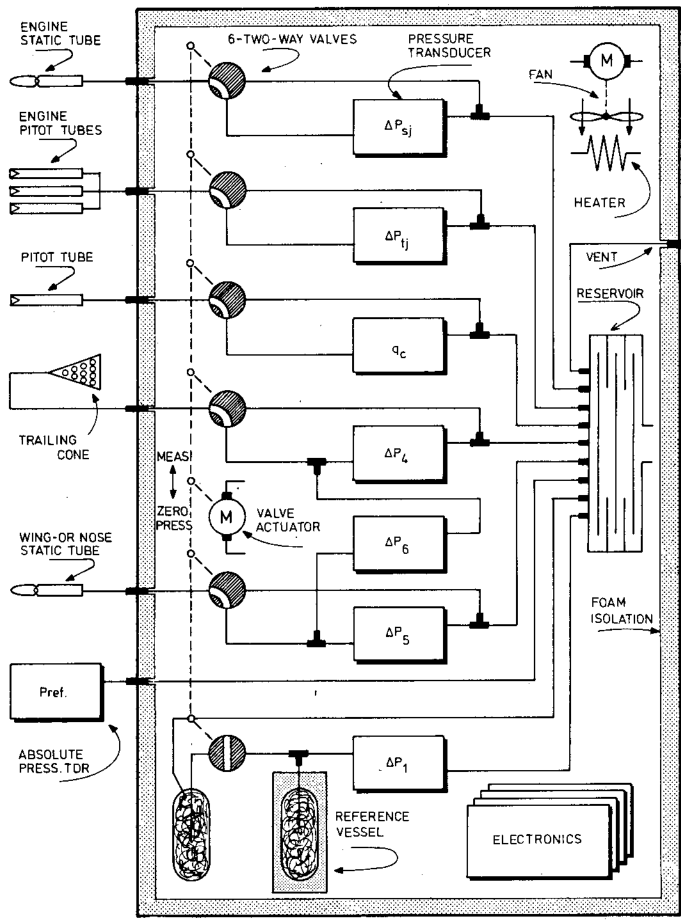


FIG. 17 : CONNECTION SCHEME OF PRESSURE TRANSDUCER BOX.

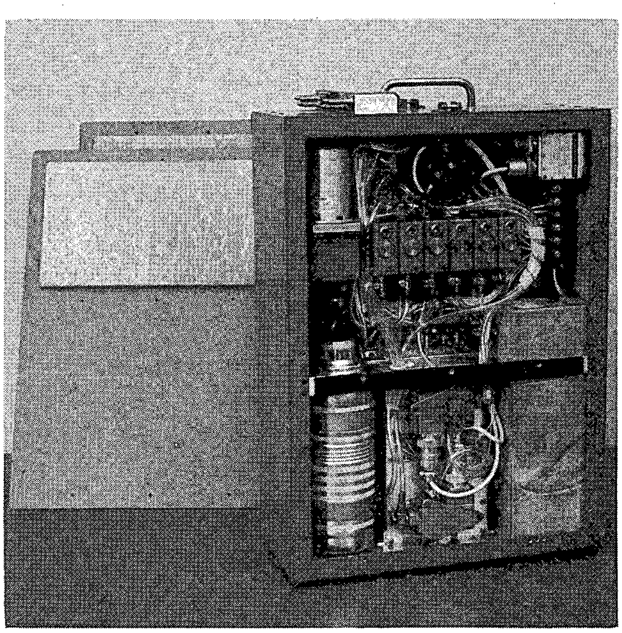


FIG. 15 : PRESSURE TRANSDUCER BOX. (VALVES AND VESSELS)

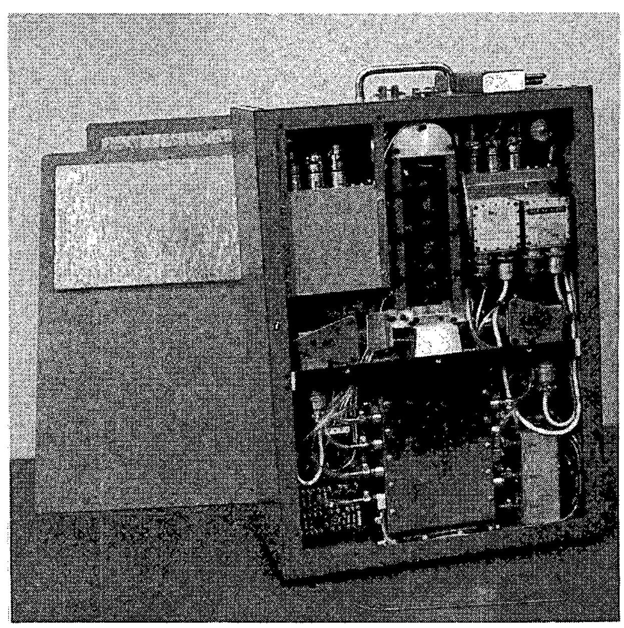


FIG. 16 : PRESSURE TRANSDUCER BOX. (HEATER AND RESERVOIR)

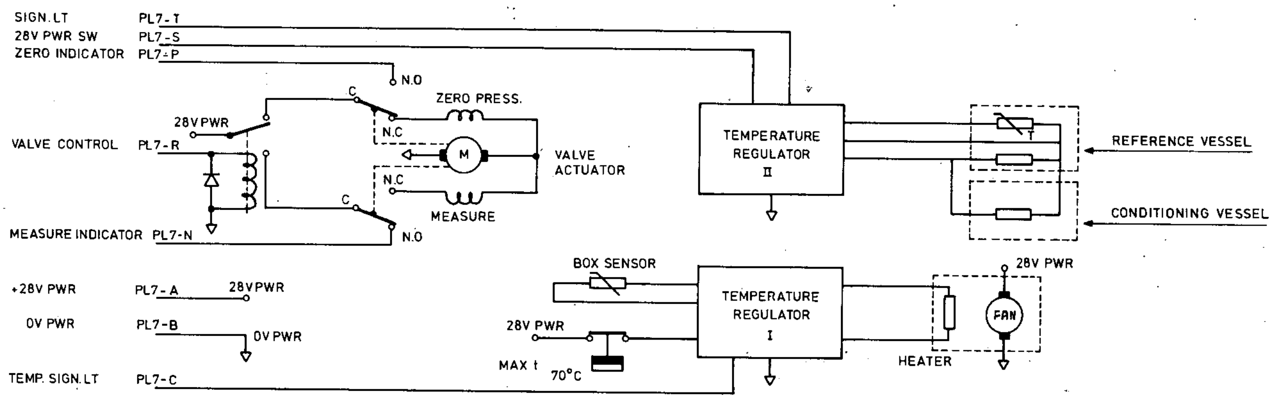


FIG. 18 : PRESSURE TRANSDUCER BOX, VALVE- AND TEMPERATURE CONTROL.

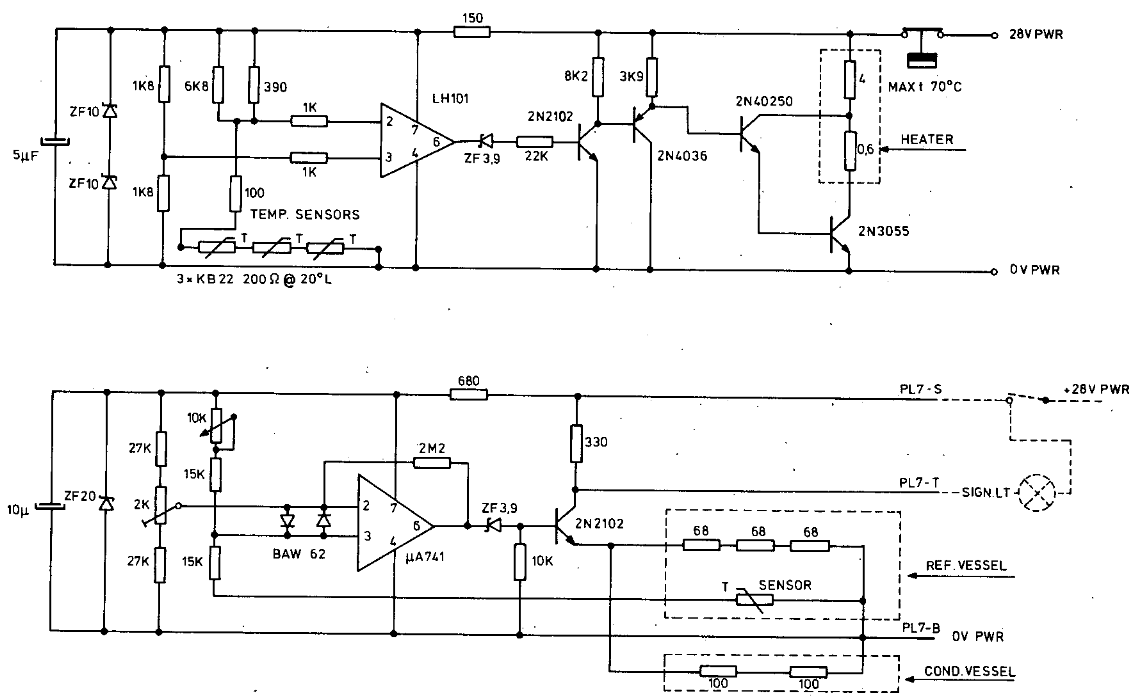


FIG. 19 : TEMPERATURE REGULATORS OF THE PRESSURE TRANSDUCER BOX.

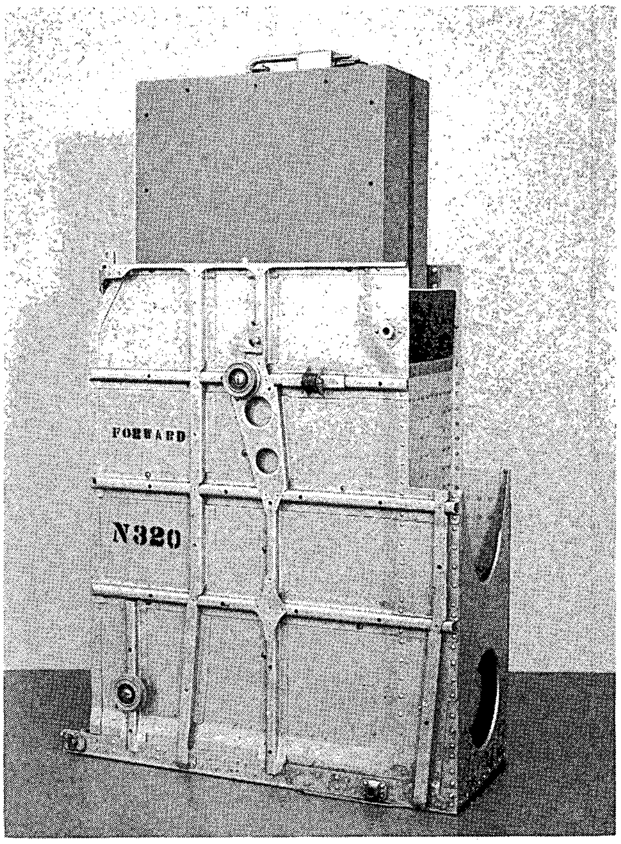


FIG. 20: PRESSURE TRANSDUCER BOX IN THE AMMUNITION CONTAINER.

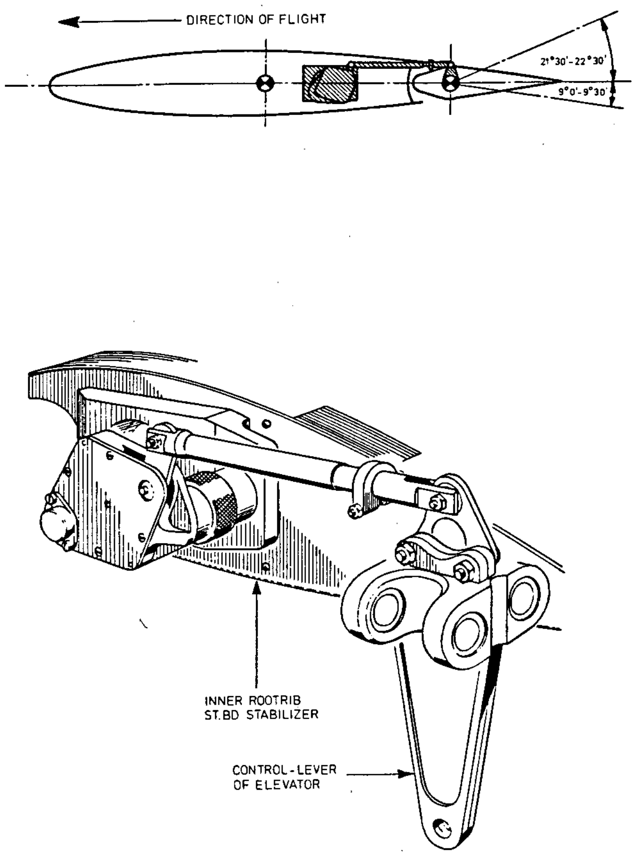


FIG. 21: POSITION OF ELEVATOR DEFLECTION TRANSDUCER

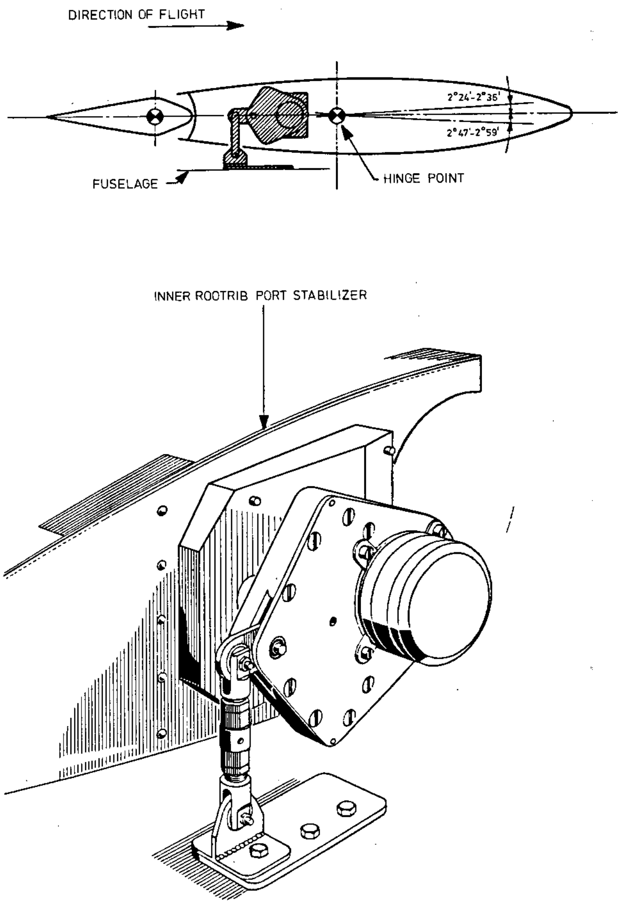


FIG. 22: POSITION OF STABILIZER TRIMANGLE TRANSDUCER.

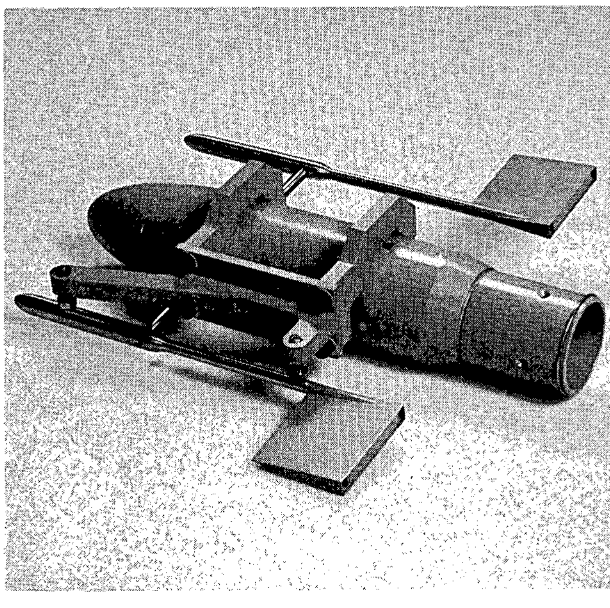


FIG. 23: ANGLE OF ATTACK TRANSDUCER.

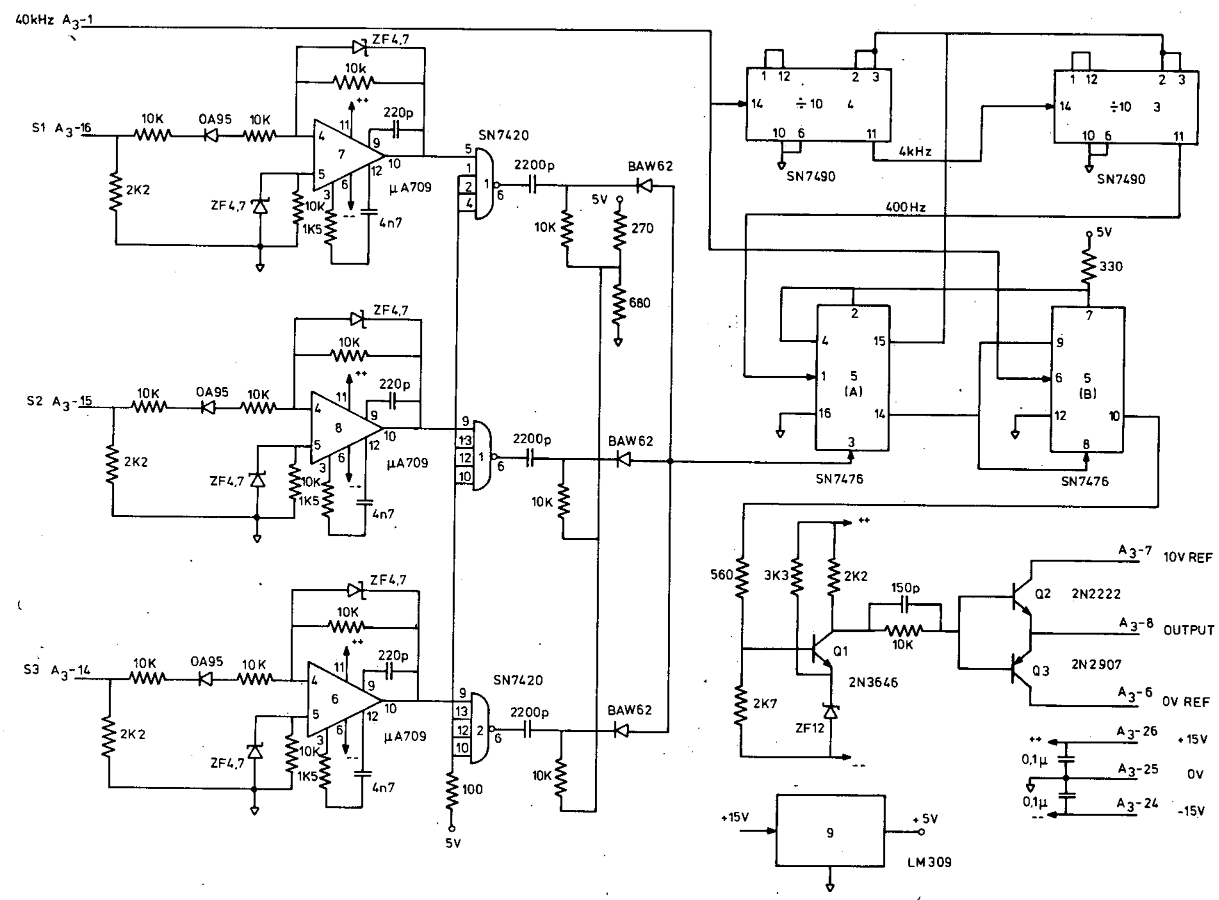


FIG : 24 : ENGINE SPEED FREQUENCY/DC CONVERTER. (SLOT A3)

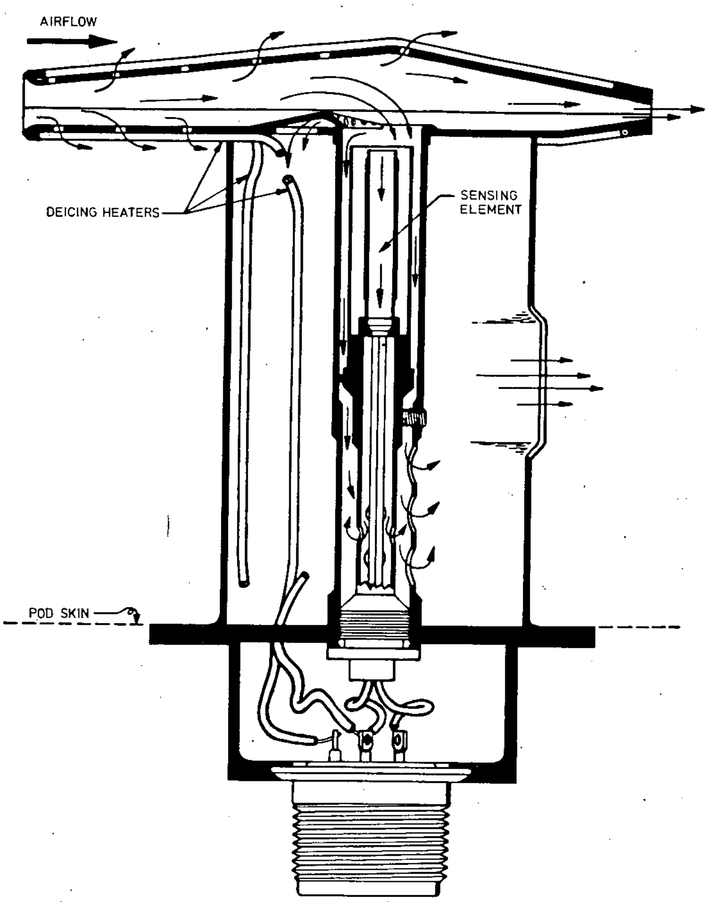


FIG. 25 : TOTAL TEMPERATURE PROBE (ROSEMOUNT ENG. COMP LTD)

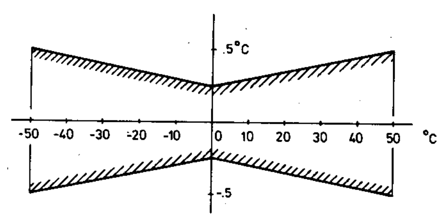


FIG. 26: STANDARD TOLERANCE OF TOTAL TEMPERATURE SENSOR, MODEL 102.

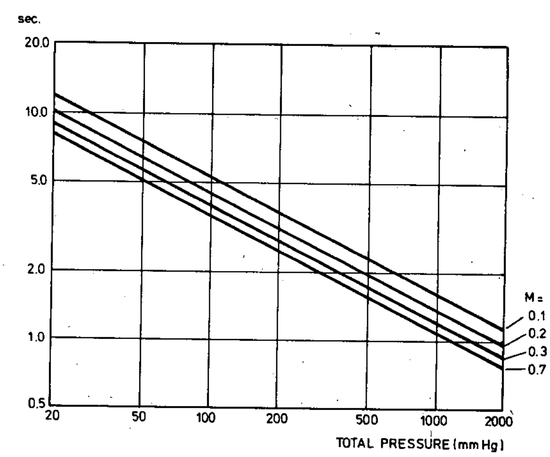


FIG.27 : TIME CONSTANT OF TOTAL TEMPERATURE SENSOR. (SEALED ELEMENT).

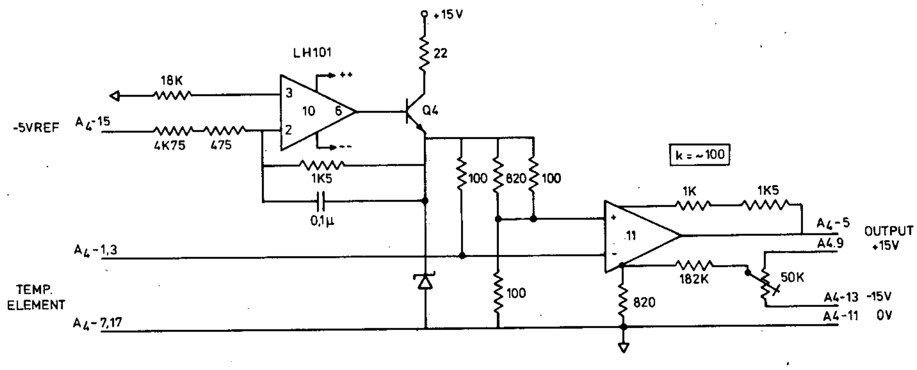


FIG. 28 : TOTAL TEMPERATURE MEASUREMENT CIRCUIT.
(SLOT A4)

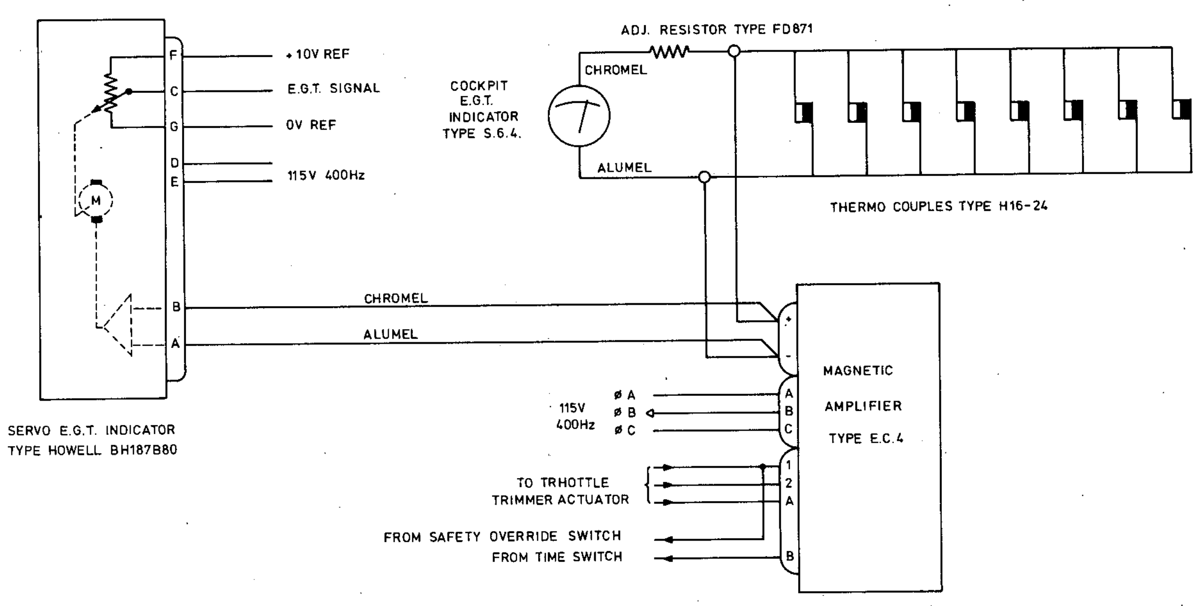


FIG. 29 : CIRCUIT DIAGRAM E.G.T. MEASUREMENT.

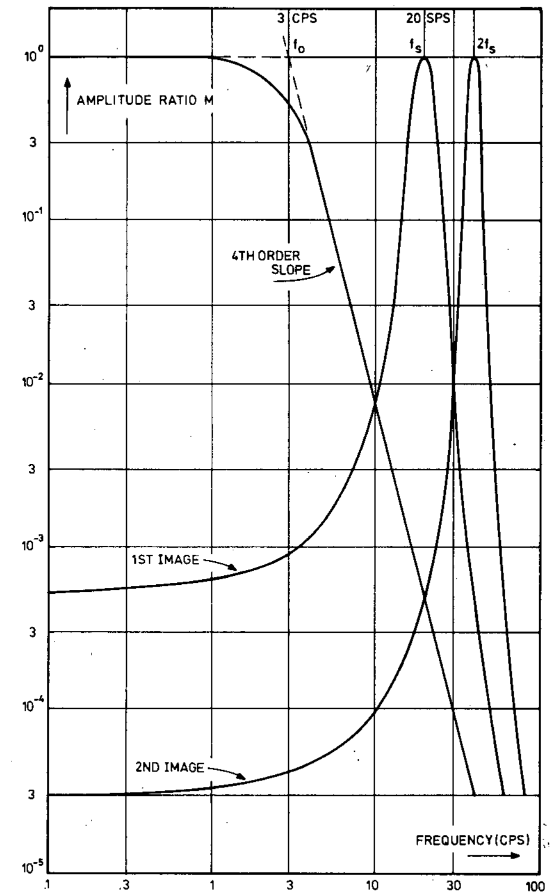


FIG. 30 : IMAGES PRODUCED BY SAMPLING OF 4TH ORDER DATA

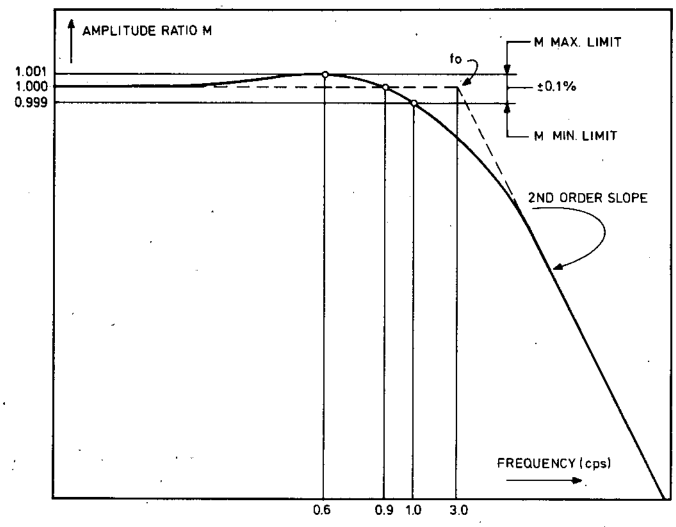


FIG. 31 : FREQUENCY CHARACTERISTIC OF 2ND ORDER FILTER.

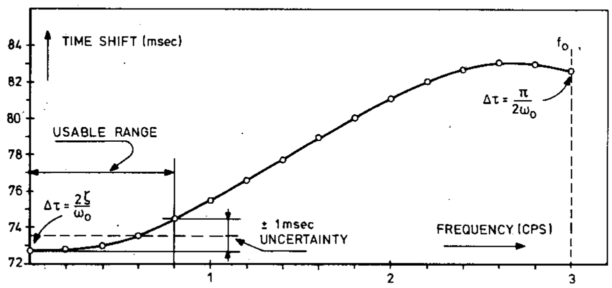


FIG. 32 : TIME SHIFT CHARACTERISTIC OF 2ND ORDER FILTER.

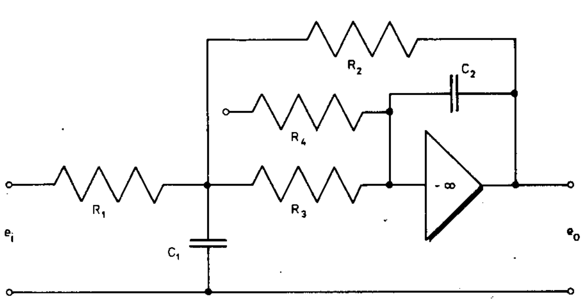


FIG. 33 : BASIC COMPONENTS OF 2ND ORDER FILTER.

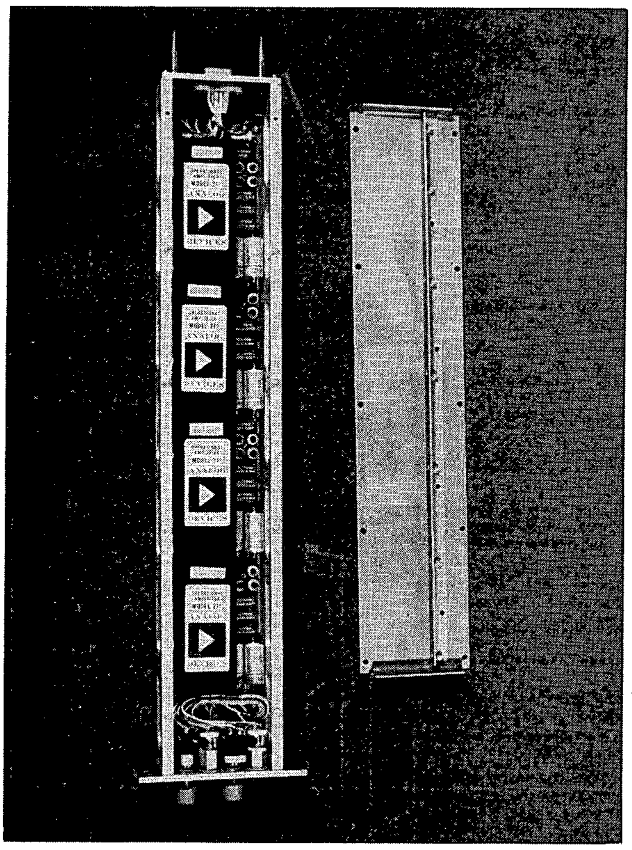


FIG. 34 : ELECTRONIC FILTER MODULE.

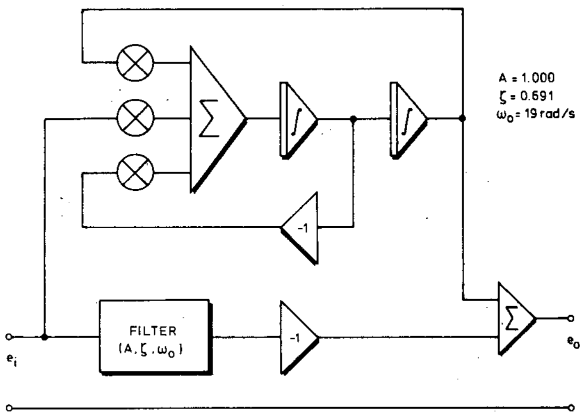


FIG. 35 : ANALOG COMPUTER ARRANGEMENT FOR FILTER ADJUSTMENT.

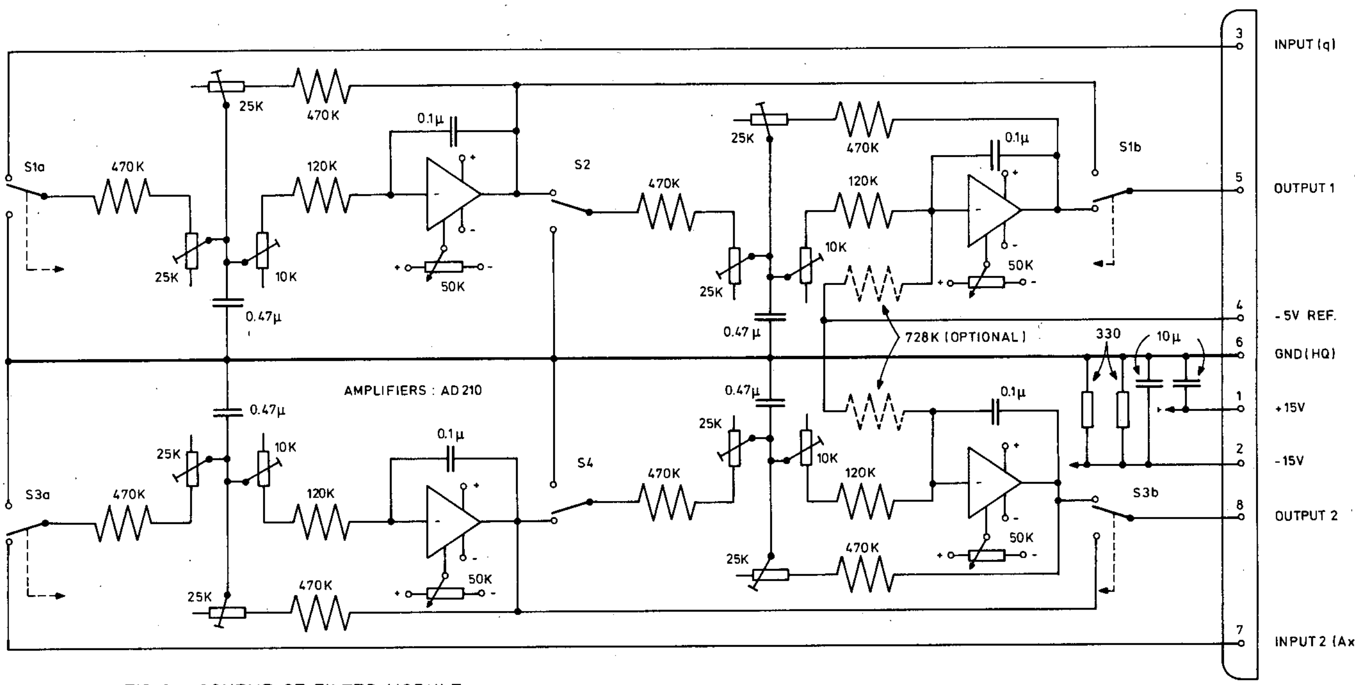


FIG. 36 : SCHEME OF FILTER MODULE .

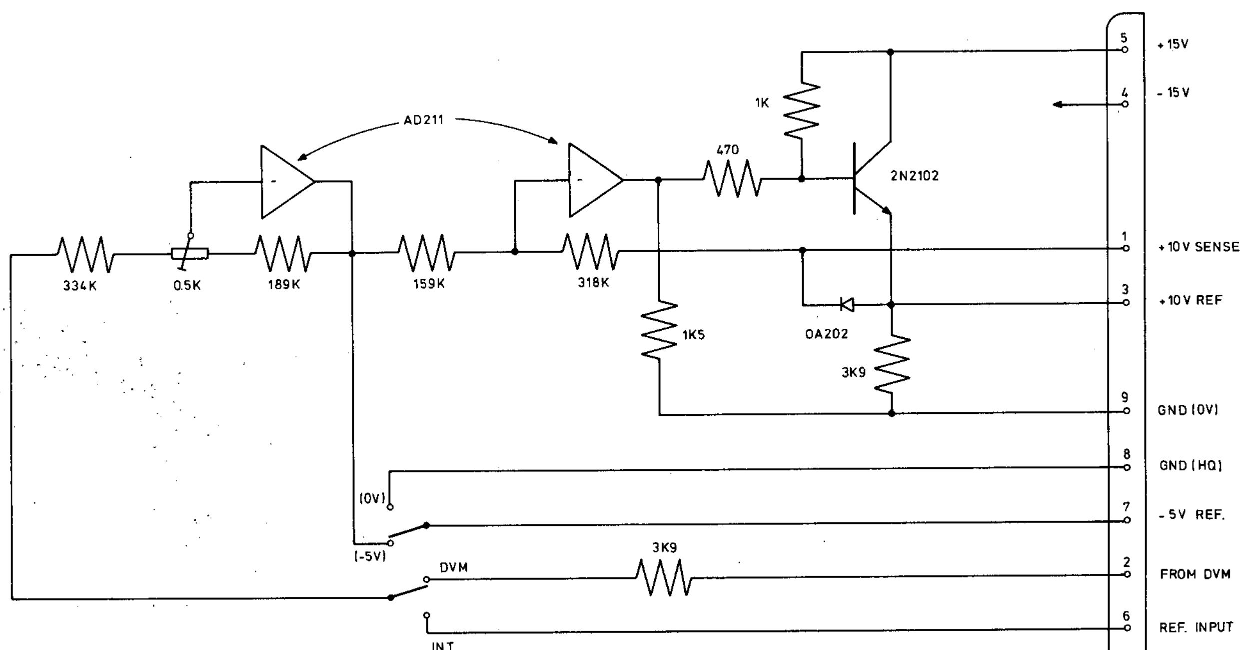


FIG. 37 : REFERENCE VOLTAGE UNIT.

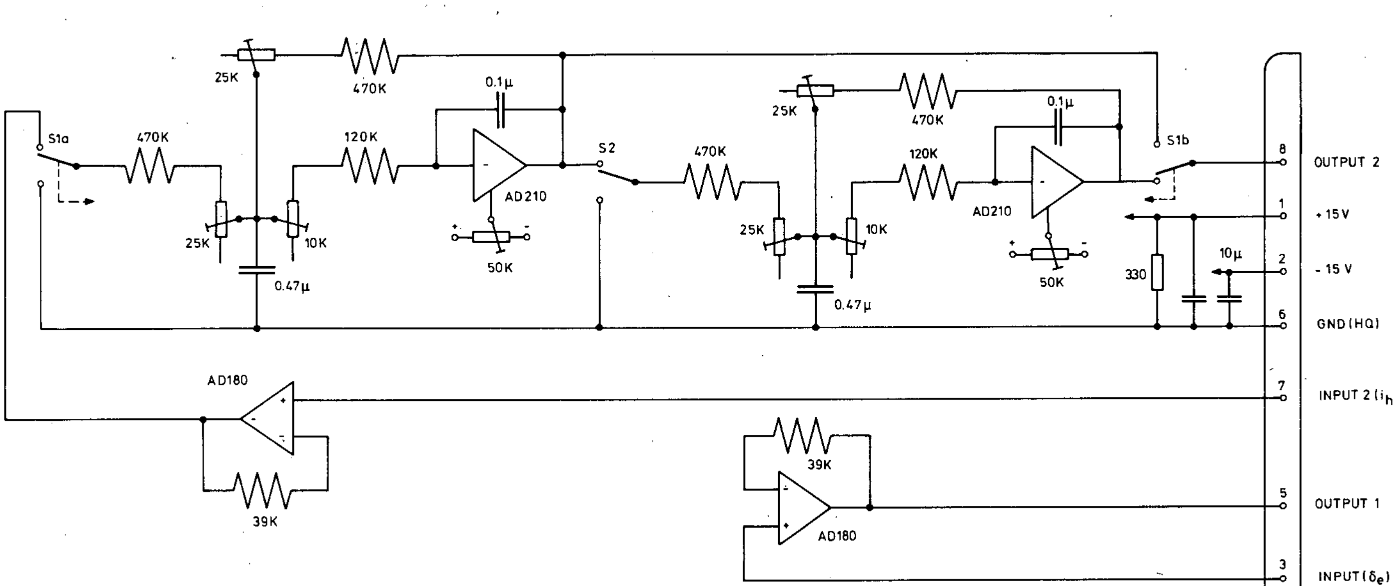


FIG. 38 : SCHEME OF FILTER UNIT WITH TWO "BUFFER" AMPLIFIERS.

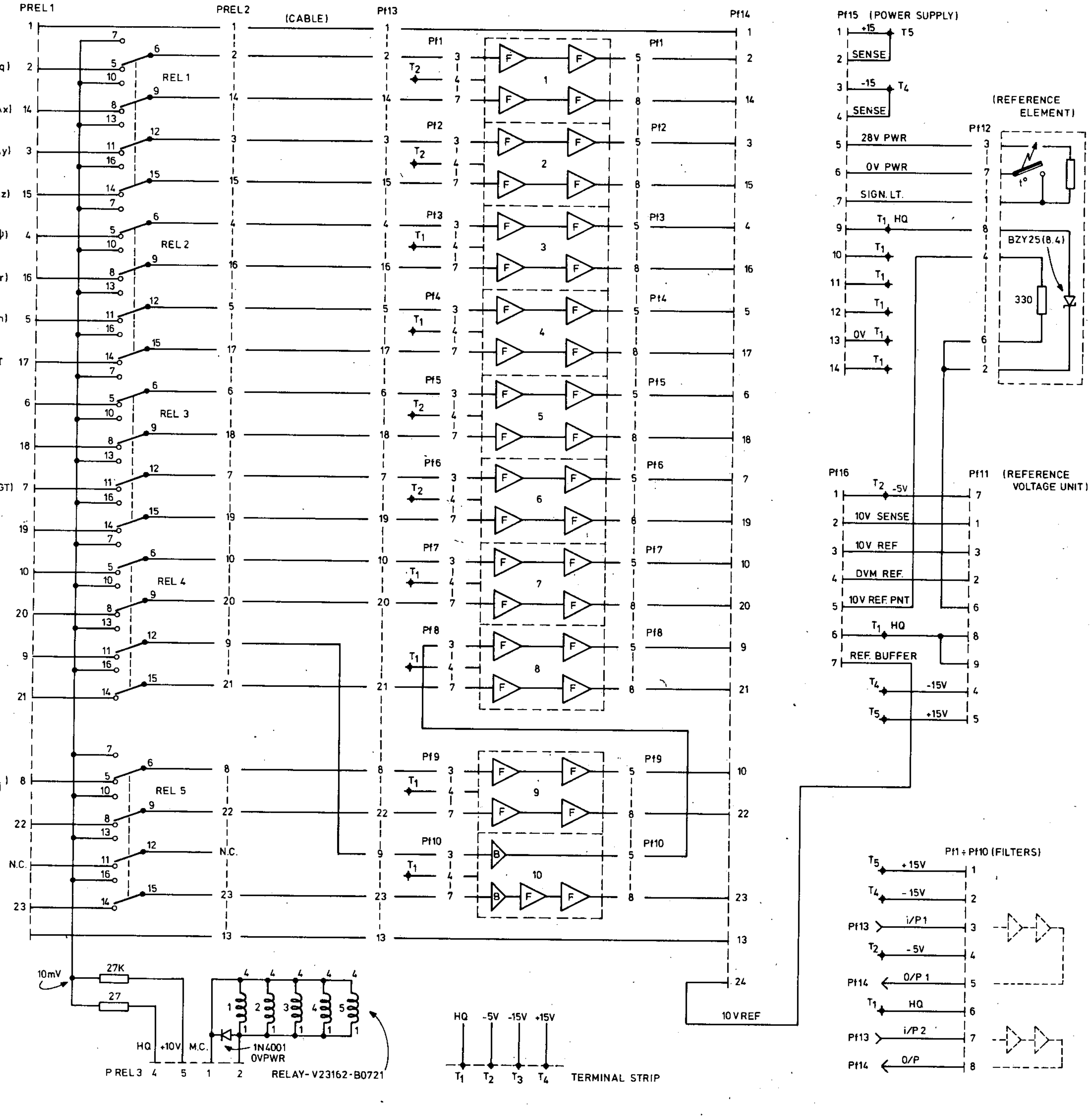


FIG. 39: WIRING DIAGRAM OF FILTERBOX AND OF RELAYBOX.

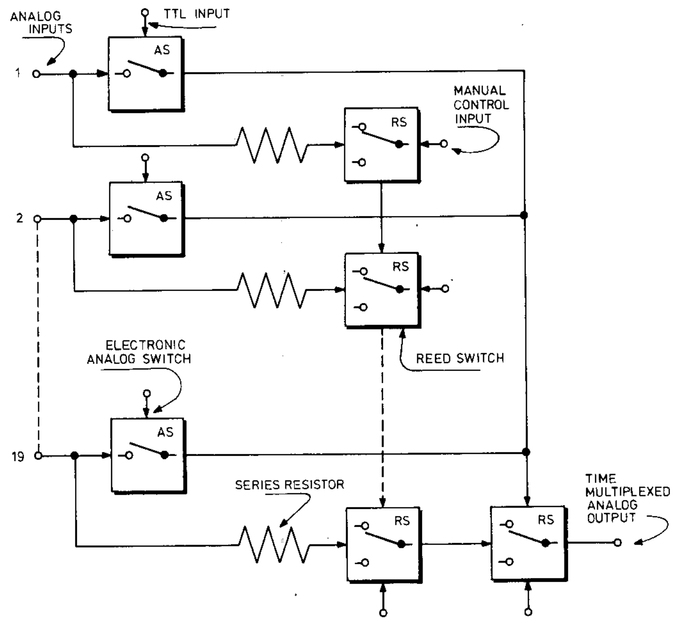


FIG.40: PRINCIPLE OF ANALOG MULTIPLEXER.

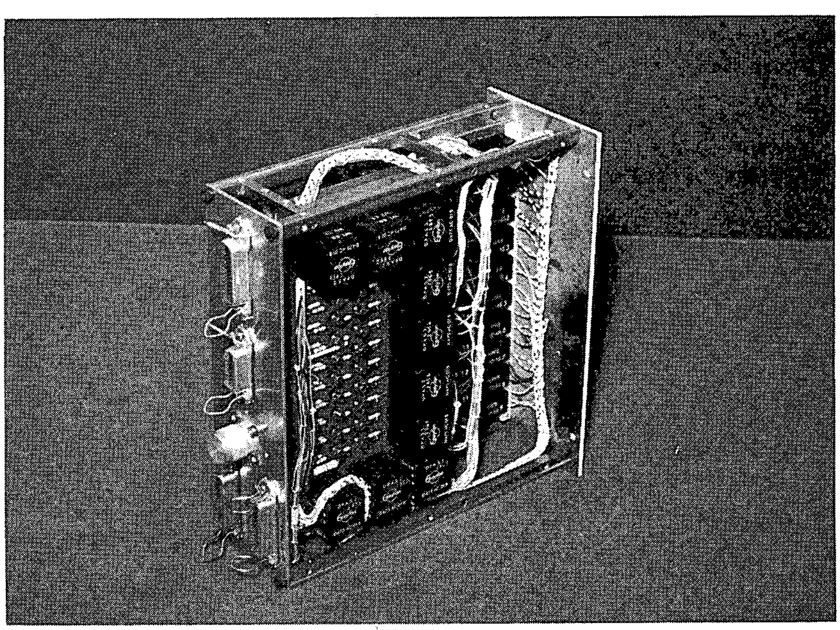


FIG. 41: MULTIPLEXER.

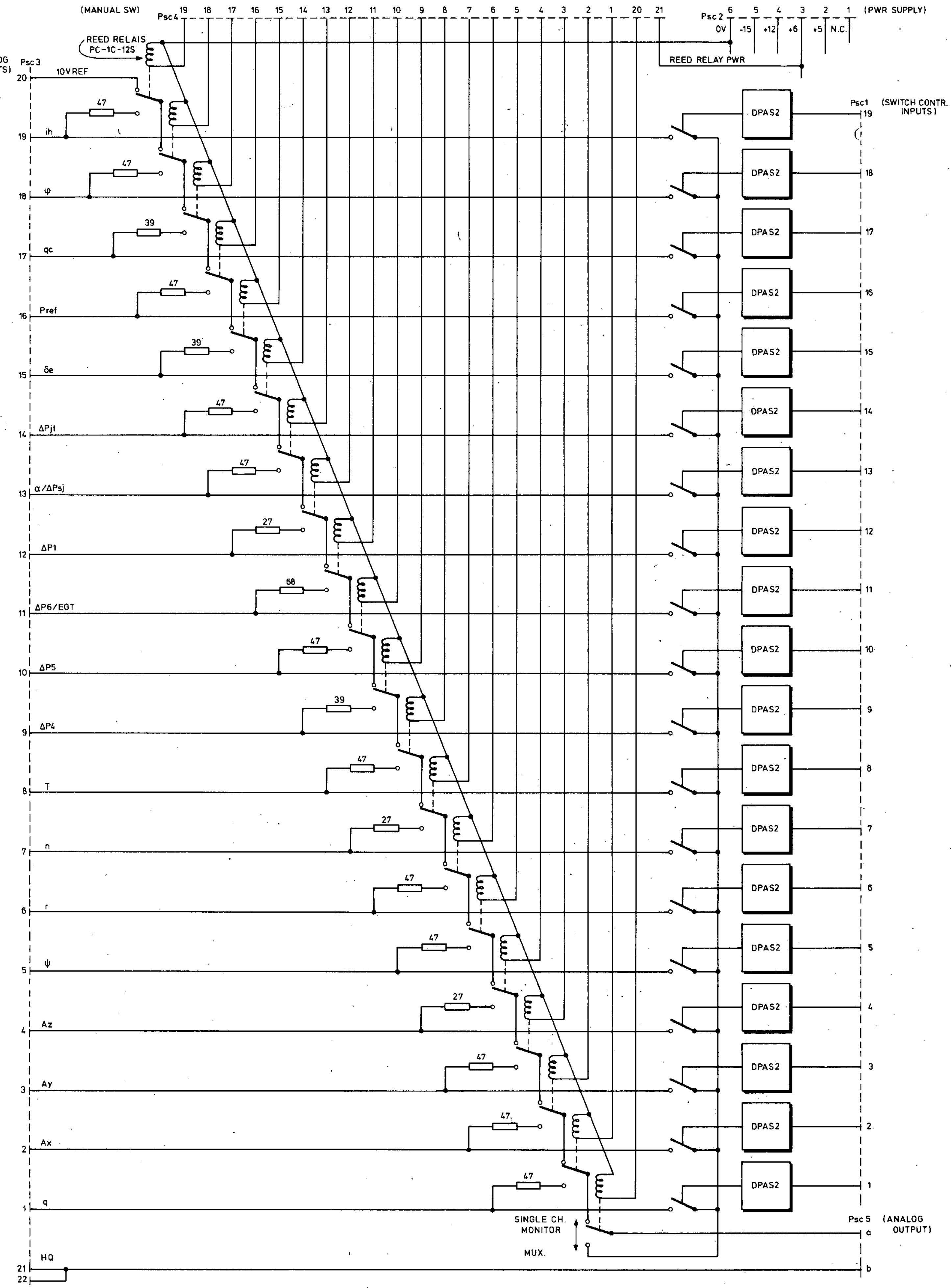


FIG. 42: SCHEME OF THE MULTIPLEXER.

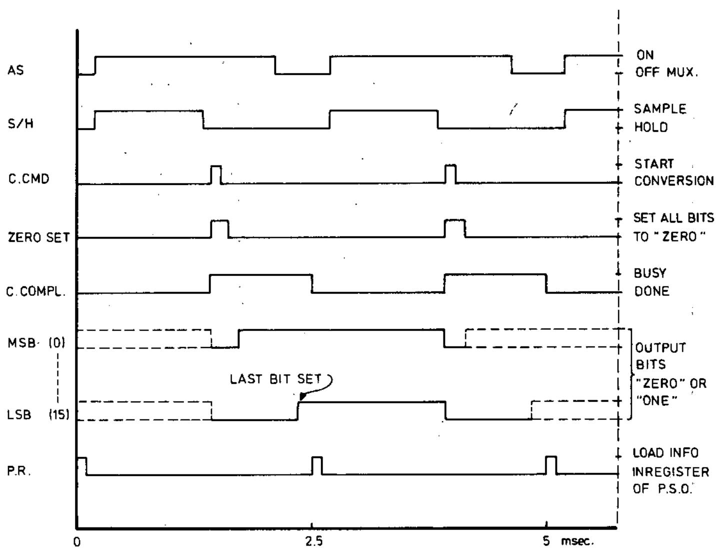


FIG. 43 : SIMPLIFIED TIMING DIAGRAM OF THE APPLIED DVM.

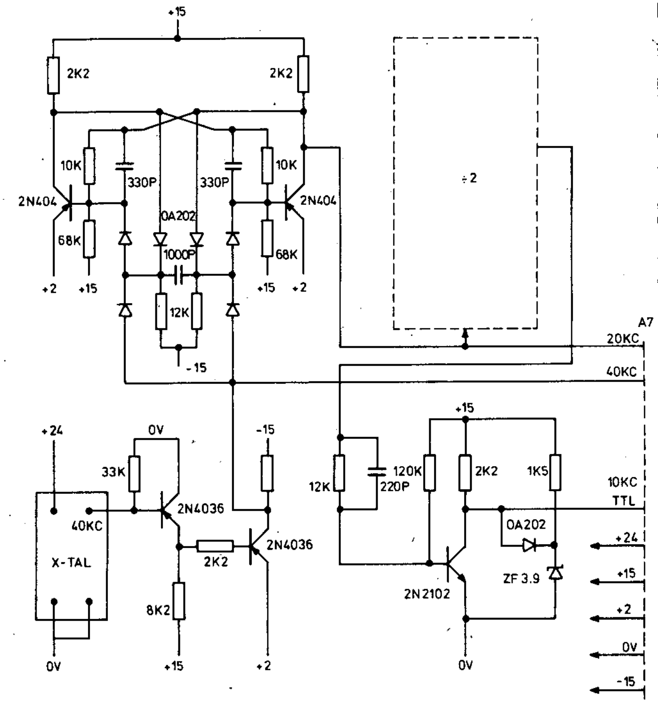


FIG. 44 : SCHEME OF REFERENCE CLOCK IN DVM. (SLOT A7)

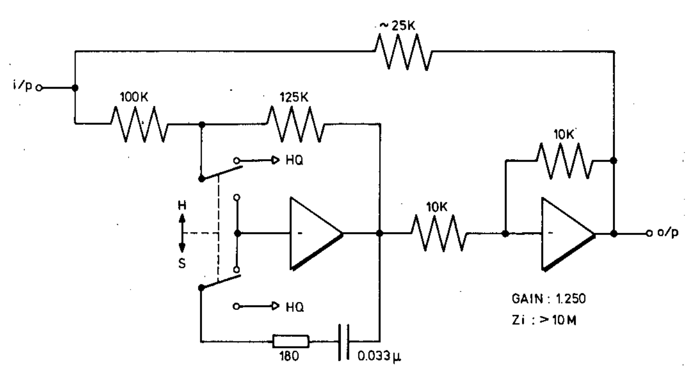


FIG. 45 : PRINCIPLE OF SAMPLE / HOLD CIRCUIT IN THE DVM.

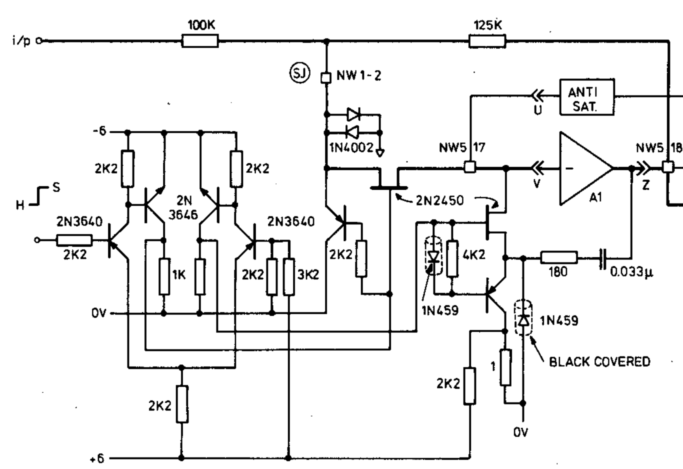


FIG. 46 : SCHEME OF SAMPLE HOLD CIRCUIT.

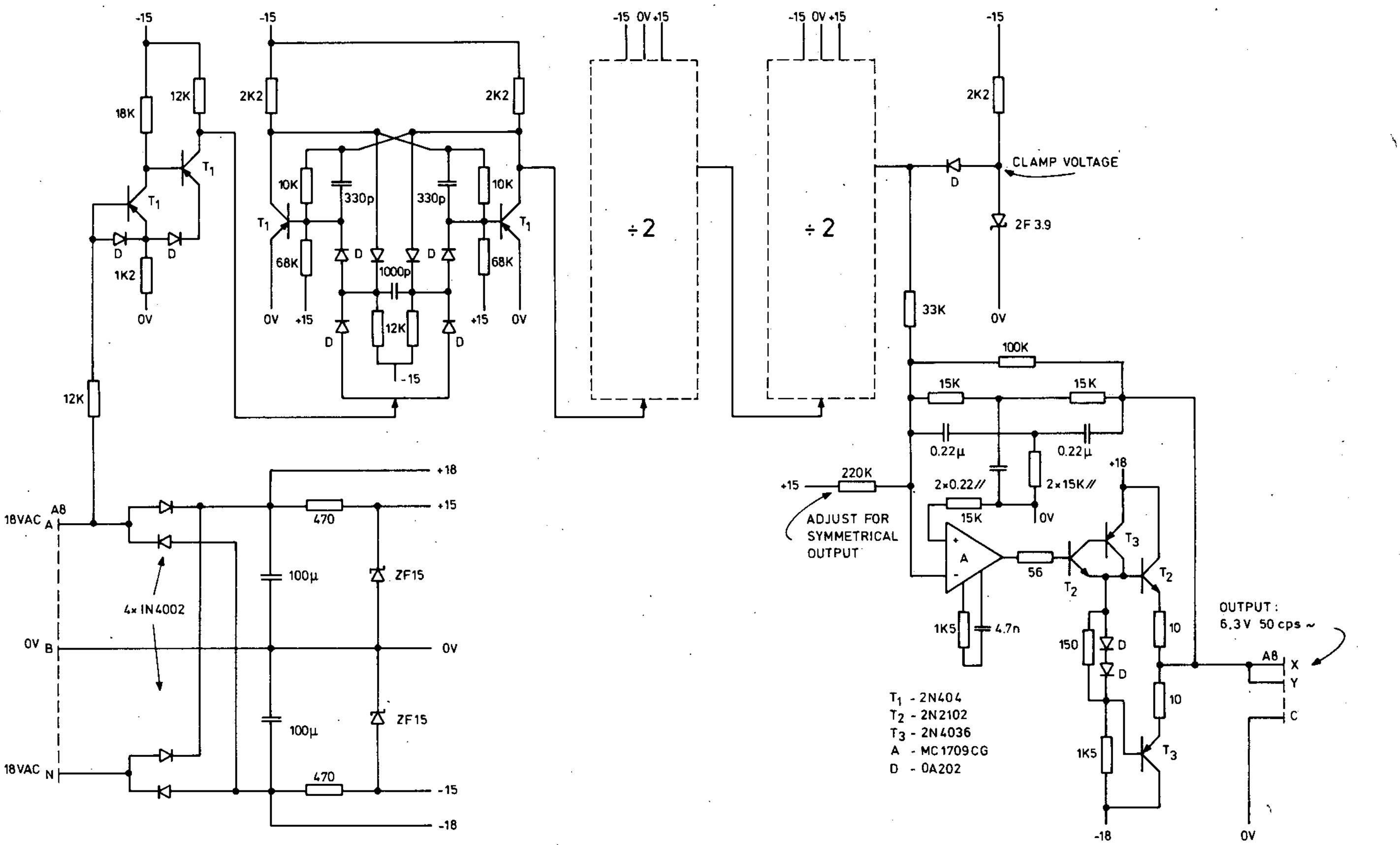


FIG. 47 : SCHEME OF DVM REFERENCE VOLTAGE CHOPPER POWER SUPPLY (SLOT A8)

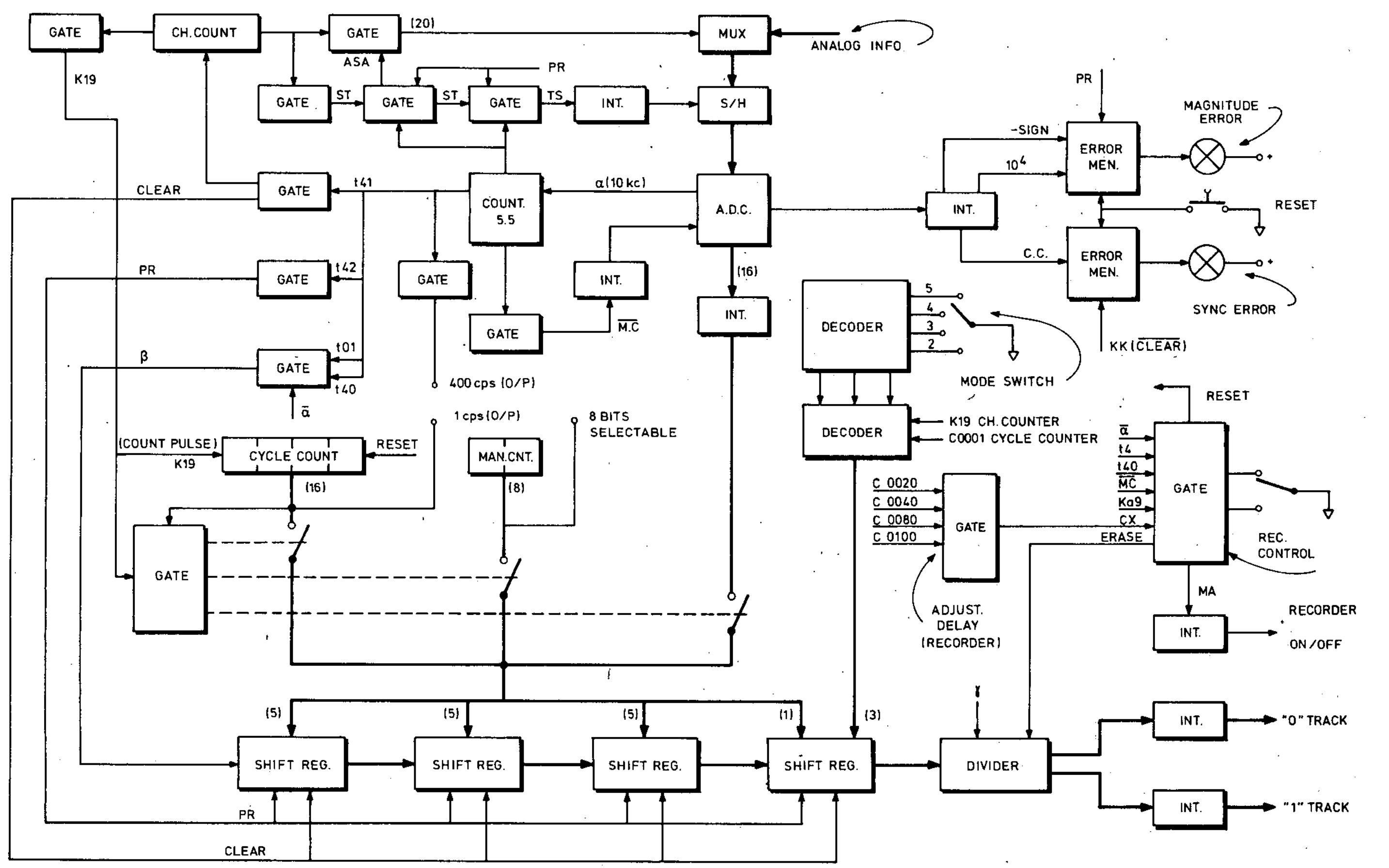


FIG. 48: BLOCK DIAGRAM OF THE PARALLEL TO SERIES CONVERTER.

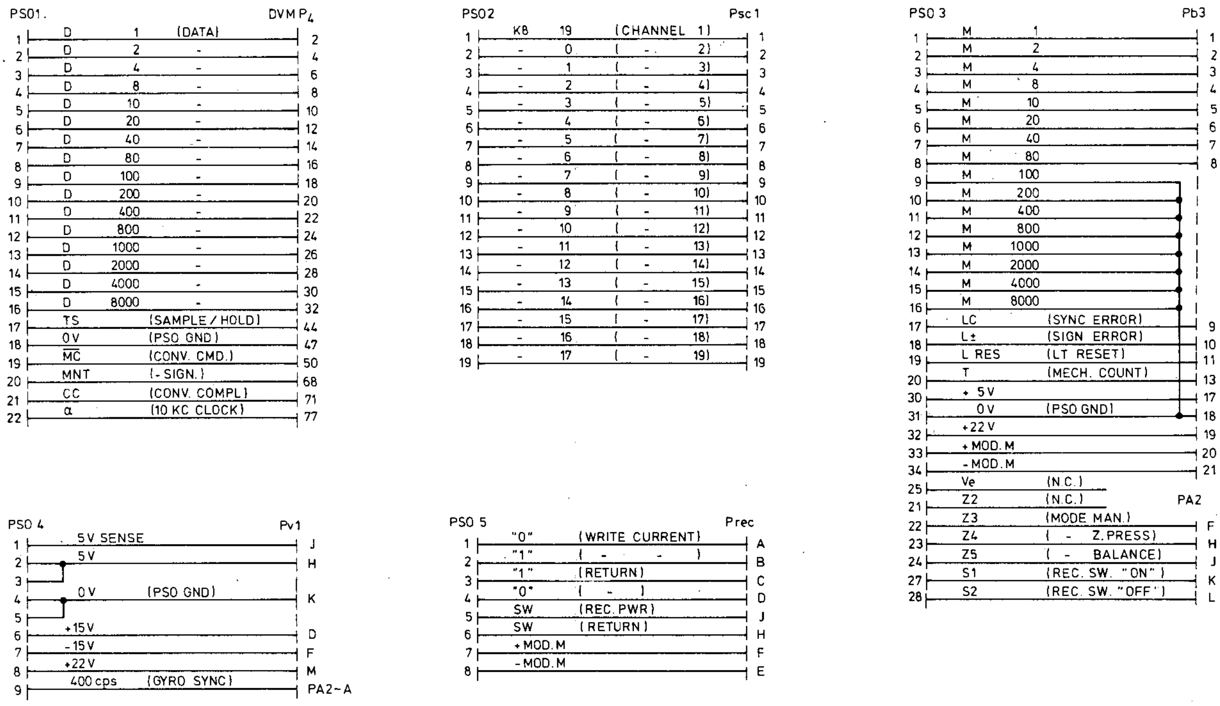


FIG. 50: INTERCONNECTION OF PARALLEL TO SERIES CONVERTER. (PSO).

FIG. 51: NAGRA IV D RECORDER

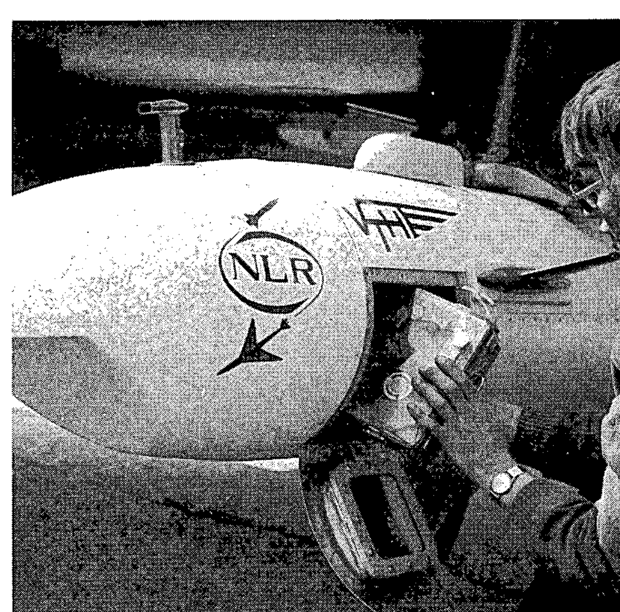


FIG. 52 : RECORDER PLACED IN THE INSTRUMENTATION POD.

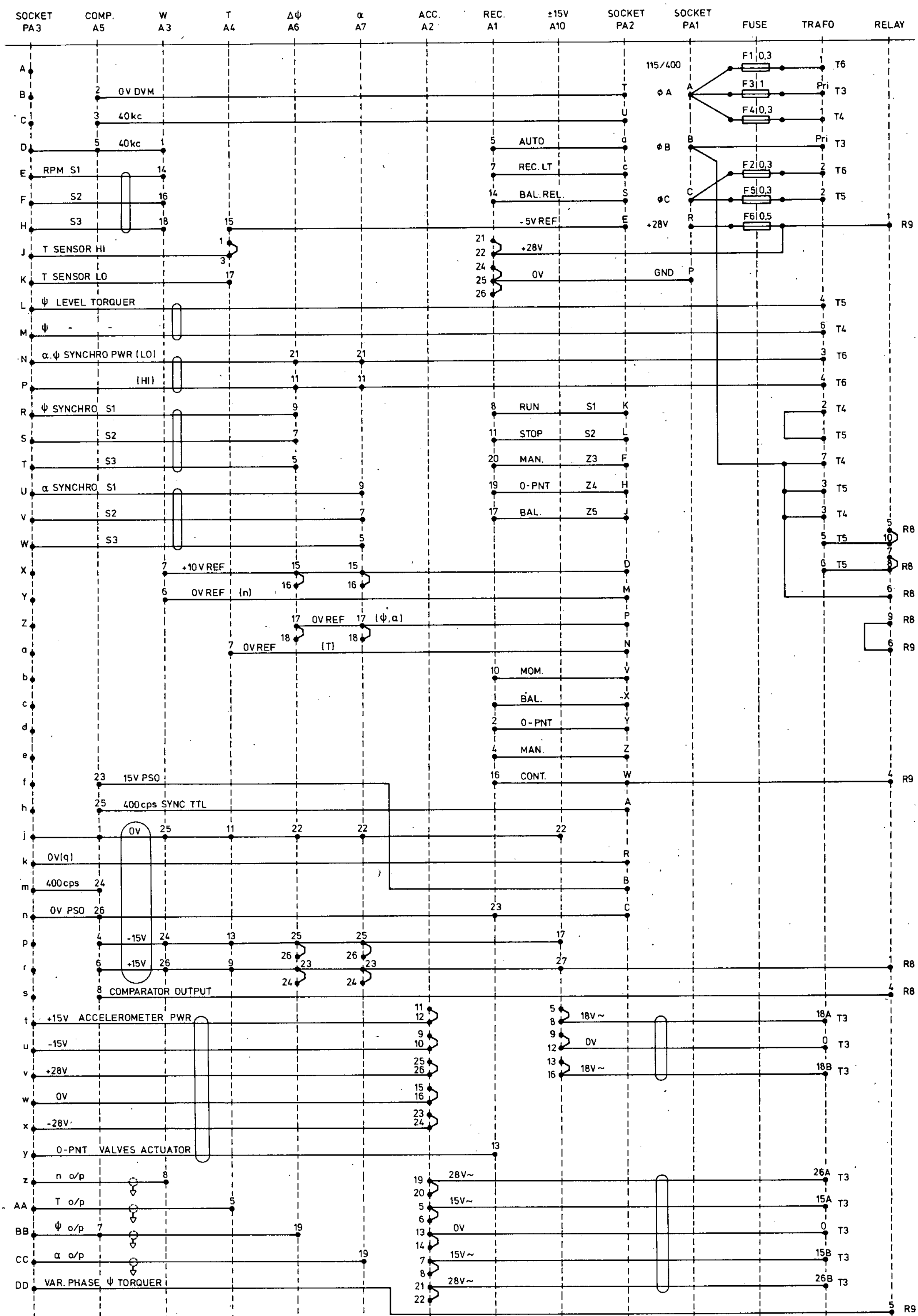


FIG. 53: WIRING DIAGRAM OF ANALOG SIGNAL CONDITIONING BOX.

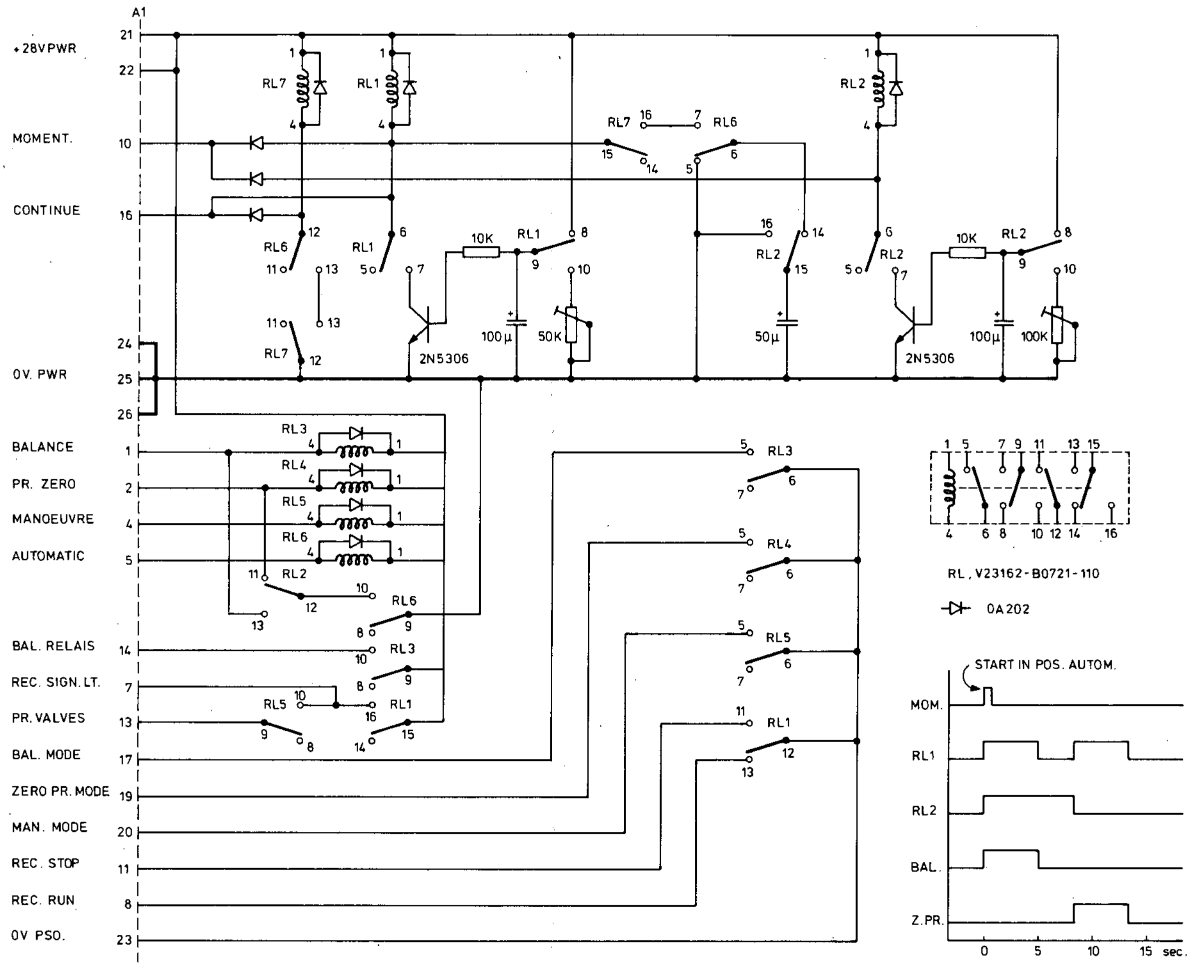


FIG. 54 : SCHEME OF RECORDER OPERATION CONTROL (INTERFACE BOARD A1)

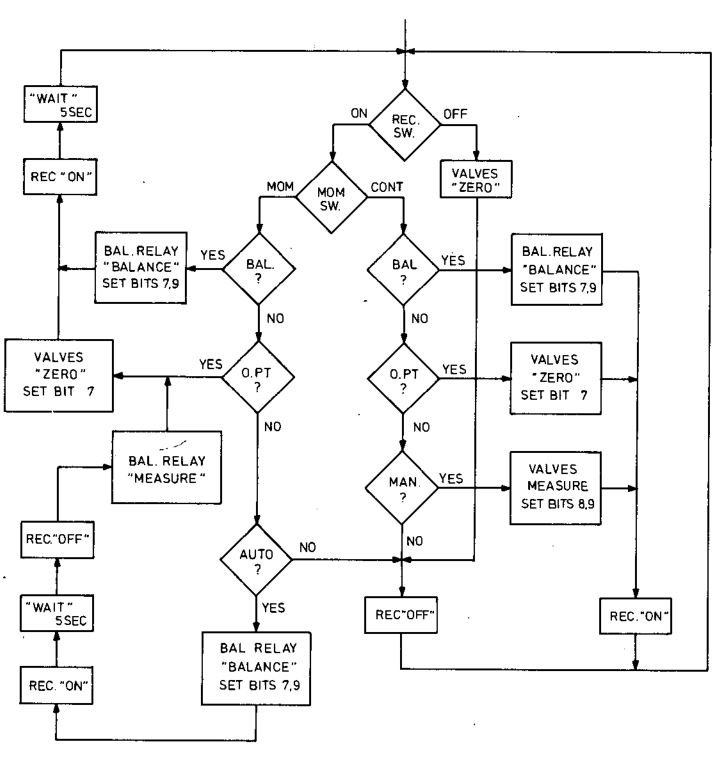


FIG. 55 : FLOW CHART RECORDER OPERATION CONTROL.

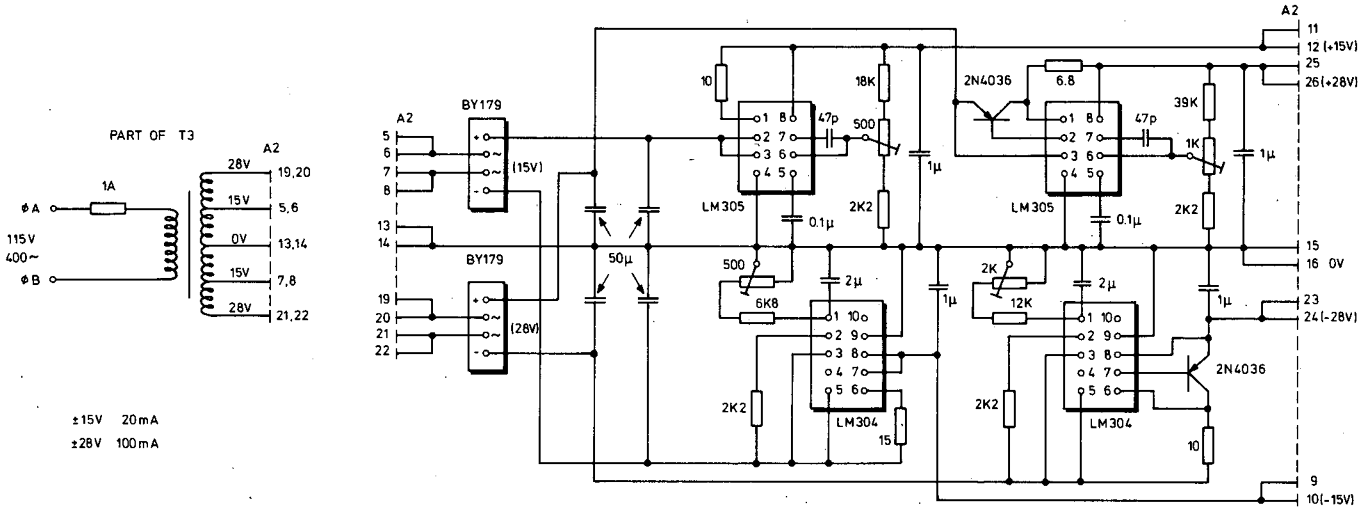


FIG. 56 : ACCELEROMETER POWER SUPPLY, ±15 AND 28V. (INTERFACE BOARD A2)

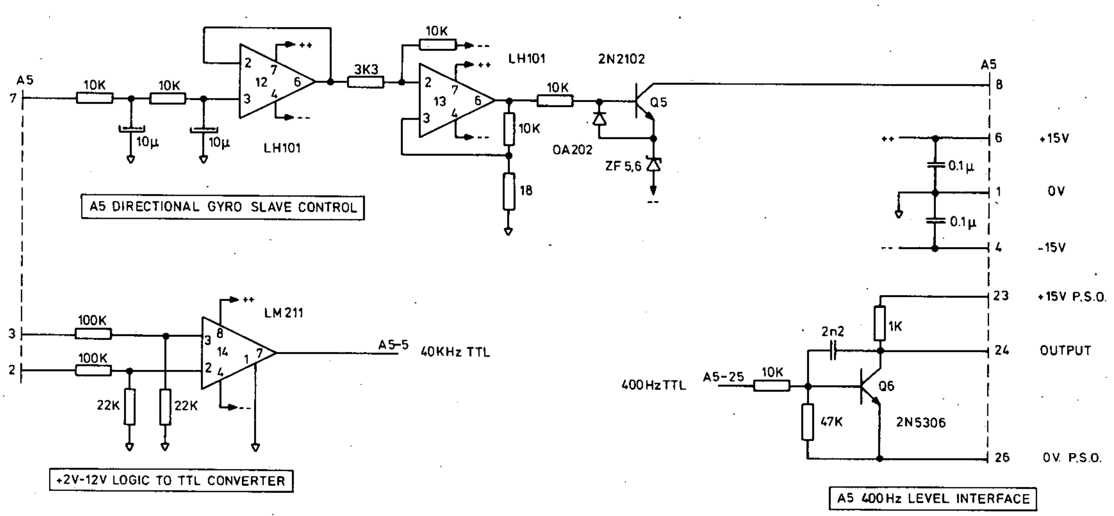


FIG. 57: MISCELLANEOUS BOARD. (A5)

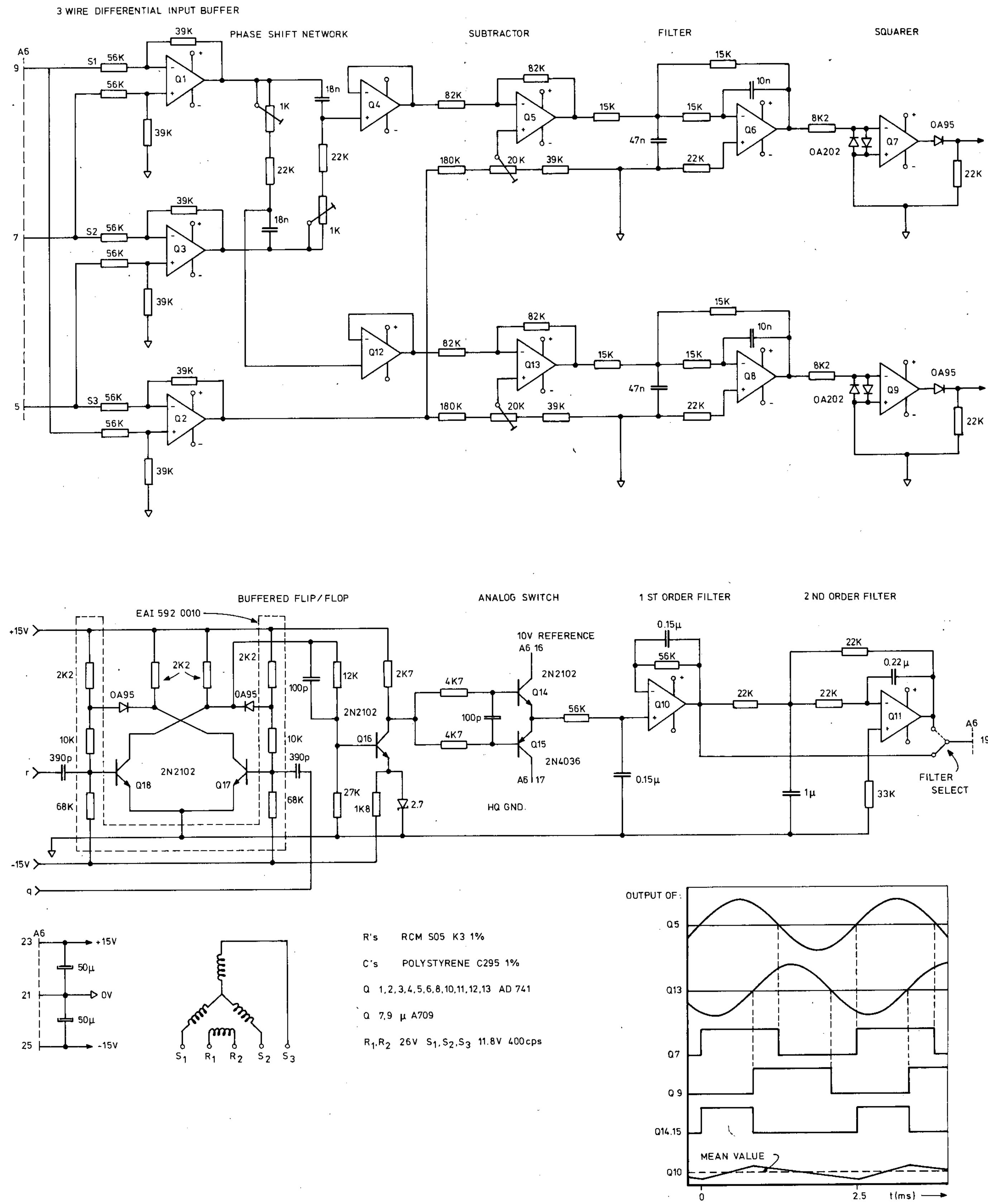


FIG. 58 : SCHEME OF THE APPLIED SYNCHRO TO DC CONVERTERS.
(INTERFACE BOARDS A6 AND A7)

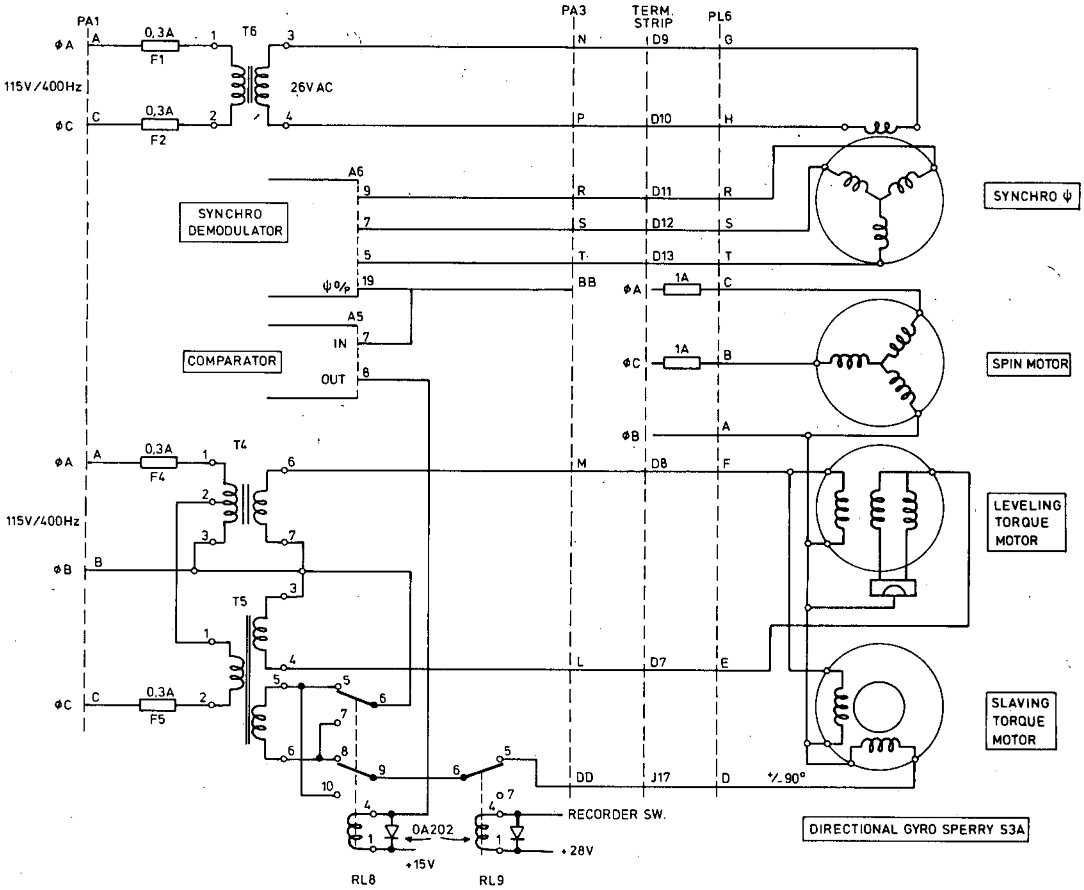


FIG. 59 : SCHEME OF DIRECTIONAL GYRO FOR HEADING CHANGE MEASUREMENT.

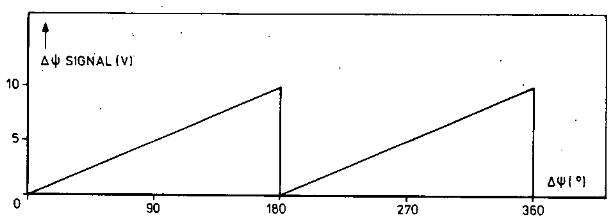


FIG. 60a : OUTPUT RELATION OF HEADING CHANGE MEASUREMENT SYSTEM.

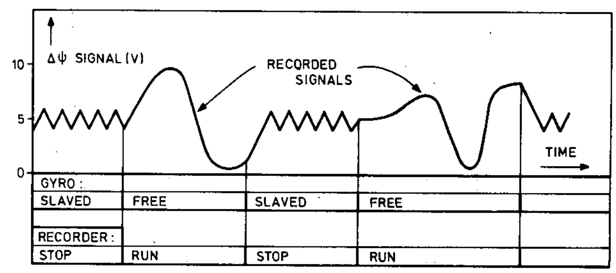


FIG. 60b : OUTPUT SIGNAL OF HEADING CHANGE MEASUREMENT SYSTEM DURING FLIGHT.

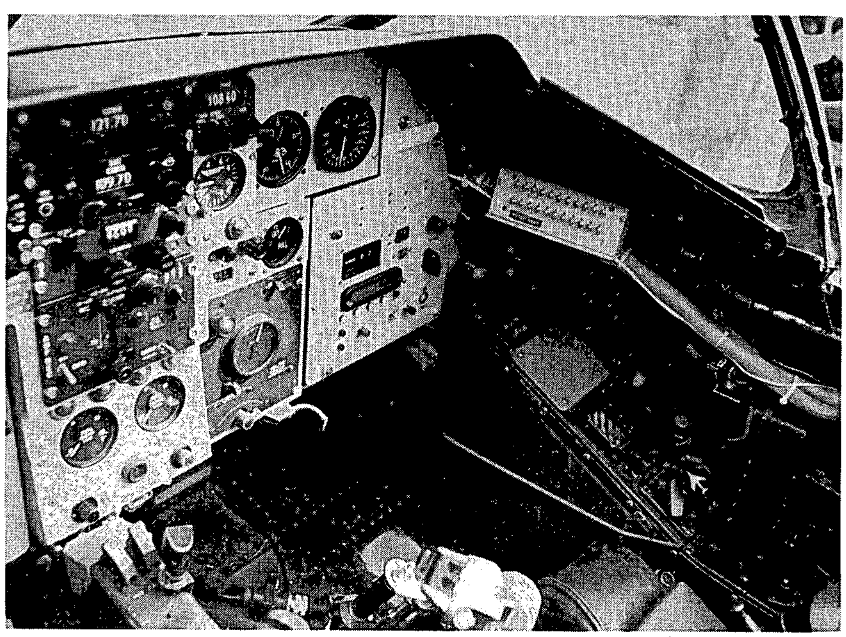


FIG. 61 : OPERATORS CONTROL PANEL.

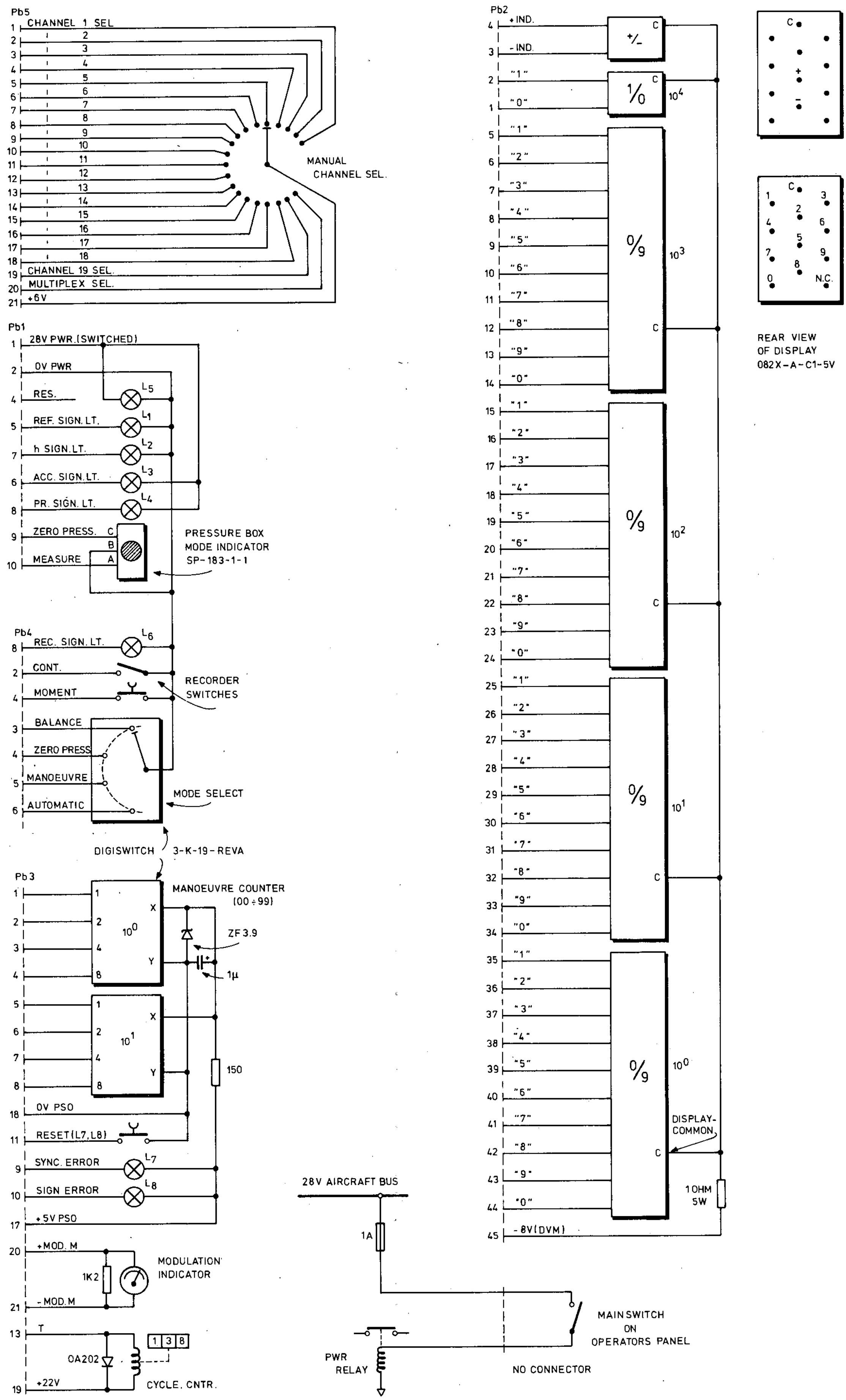


FIG. 62 : CONNECTION SCHEME OF OPERATORS PANEL.

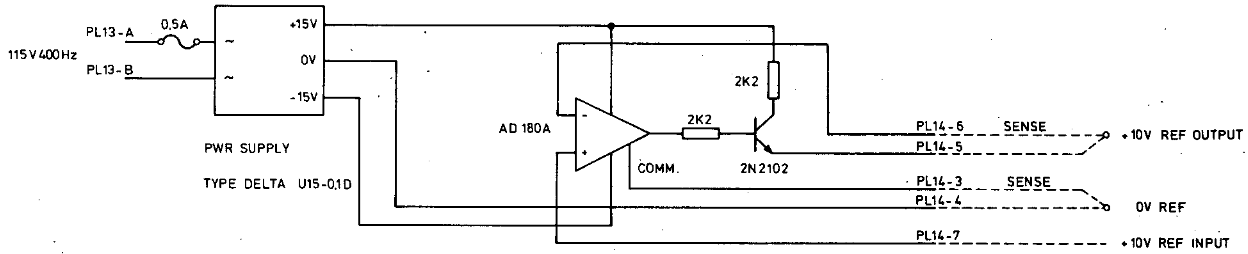
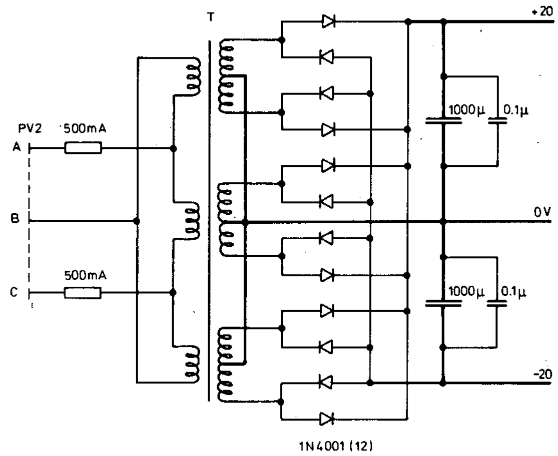


FIG. 63 : REFERENCE VOLTAGE BUFFER AMPLIFIER.



T_1 - 2N2102

T_2 - 2N4036

T - PR 3×115V 400 cps
SEC. 6×18V

I max 2×2.4 A @ 15V

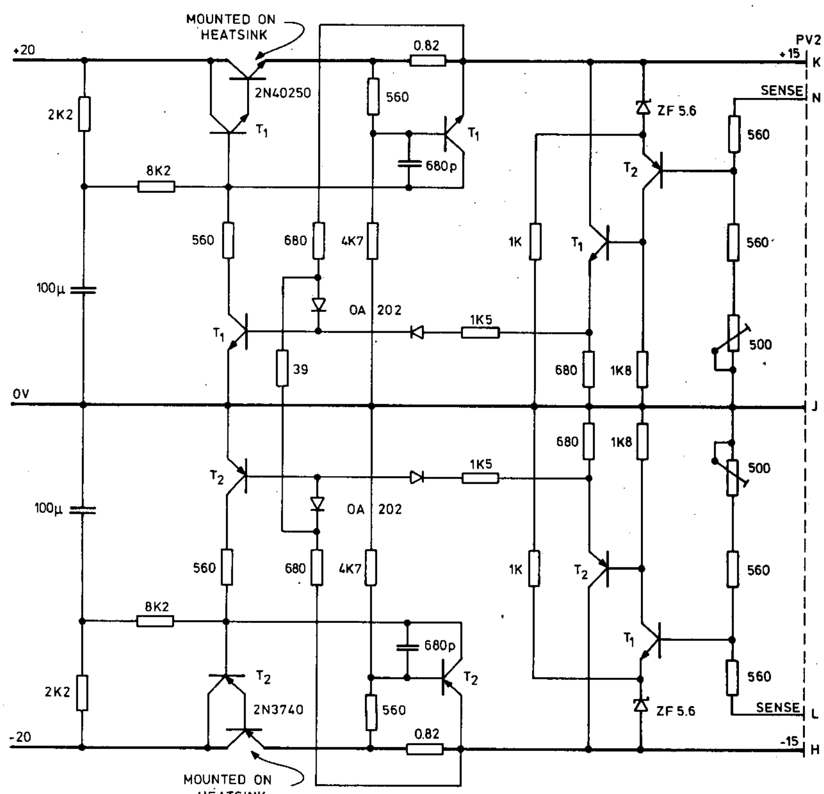


FIG. 64 : SCHEME OF DUAL 15V FILTER POWER SUPPLY.

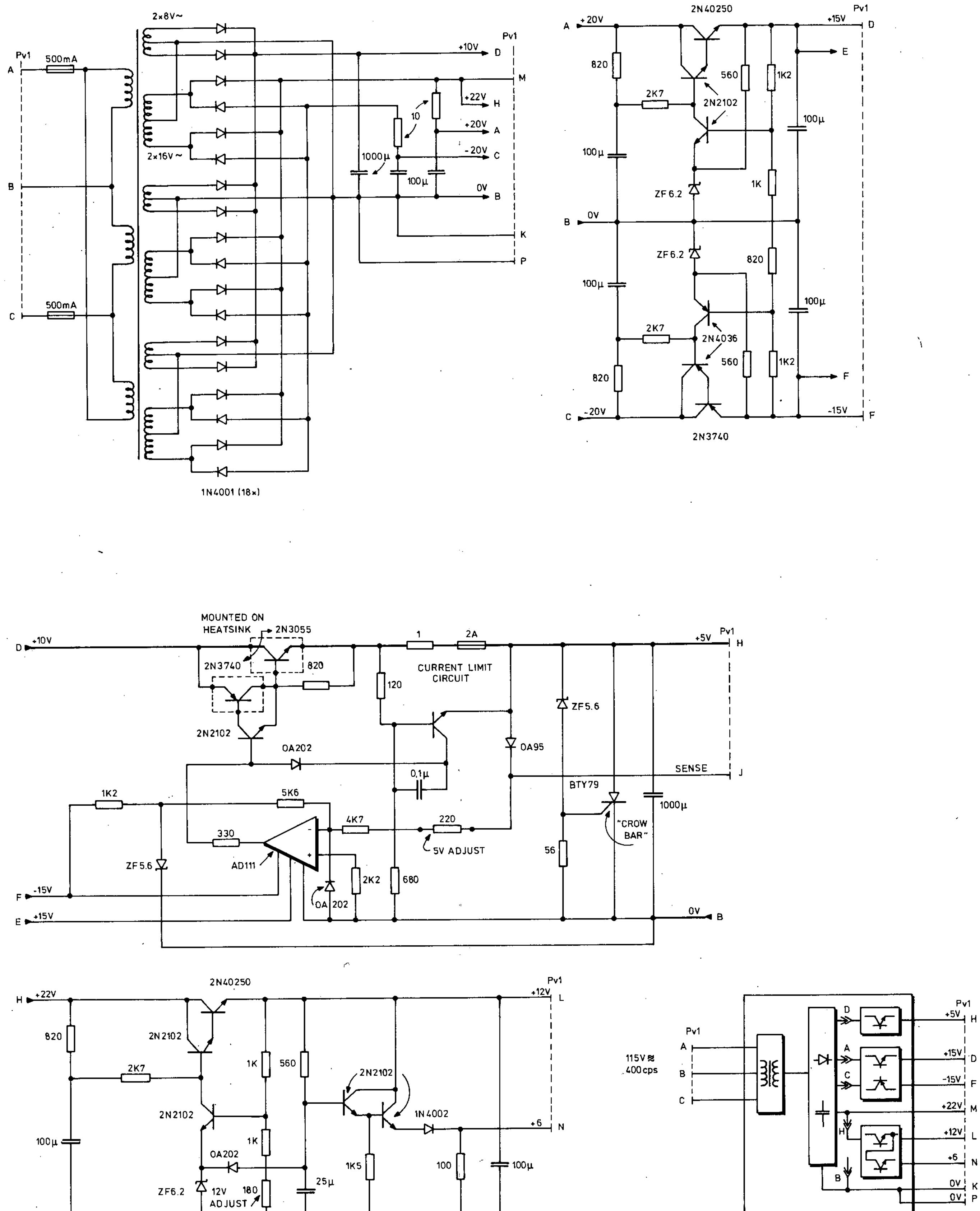


FIG. 65: SCHEME OF "DIGITAL" POWER SUPPLY.

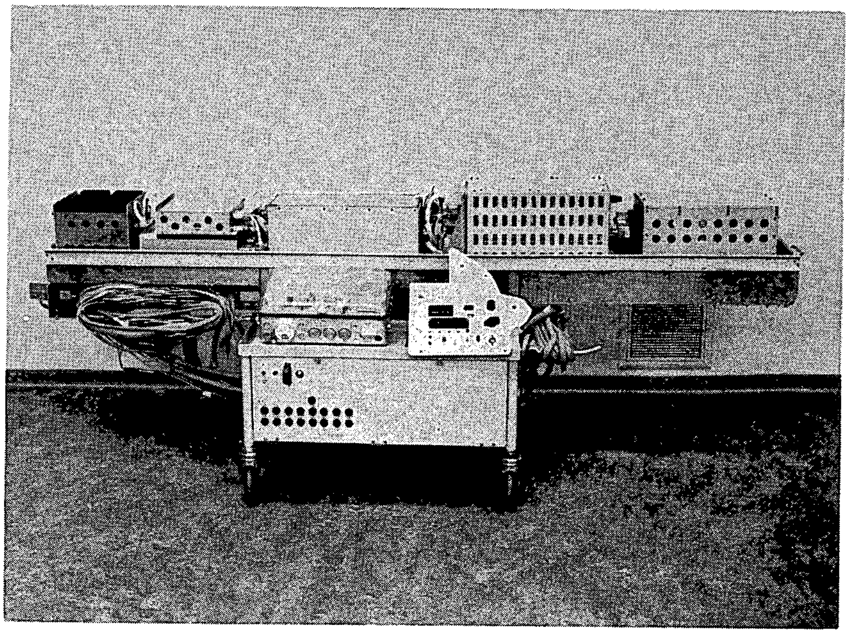


FIG. 66: INSTRUMENTATION DRAWER.

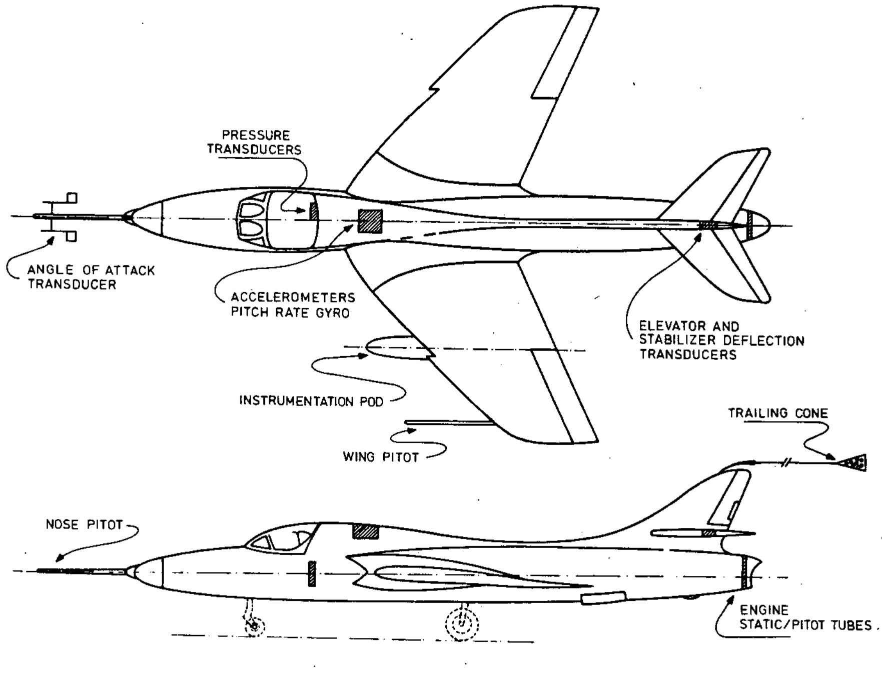


FIG. 69: TRANSDUCER LOCATIONS IN THE HUNTER MK VII.

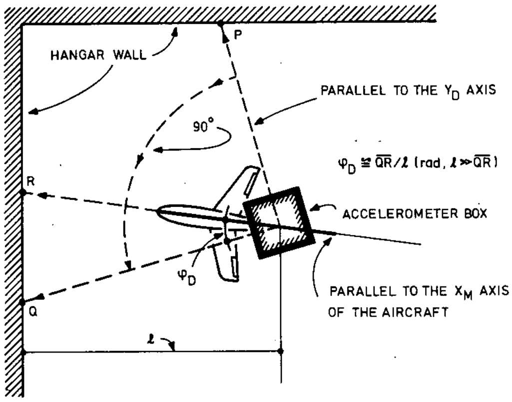


FIG. 70 : ACCELEROMETER BOX ALIGNMENT PROCEDURE.

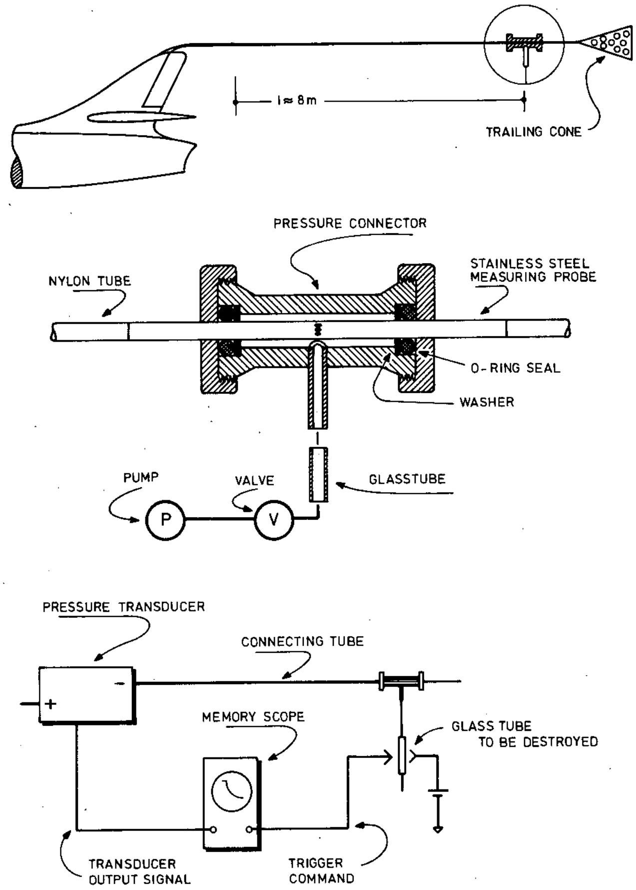


FIG. 71 : PRINCIPLE OF TIME DELAY MEASUREMENT.

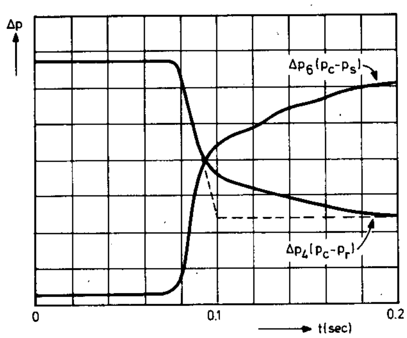


FIG. 72a : RESPONSE OF DIFFERENTIAL PRESSURE TRANSDUCERS
CONNECTED TO THE TRAILING CONE.

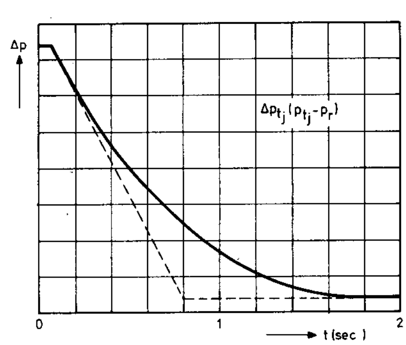


FIG. 72b : RESPONSE OF DIFFERENTIAL PRESSURE TRANSDUCER
CONNECTED TO THE JET PIPE PITOT TUBES.

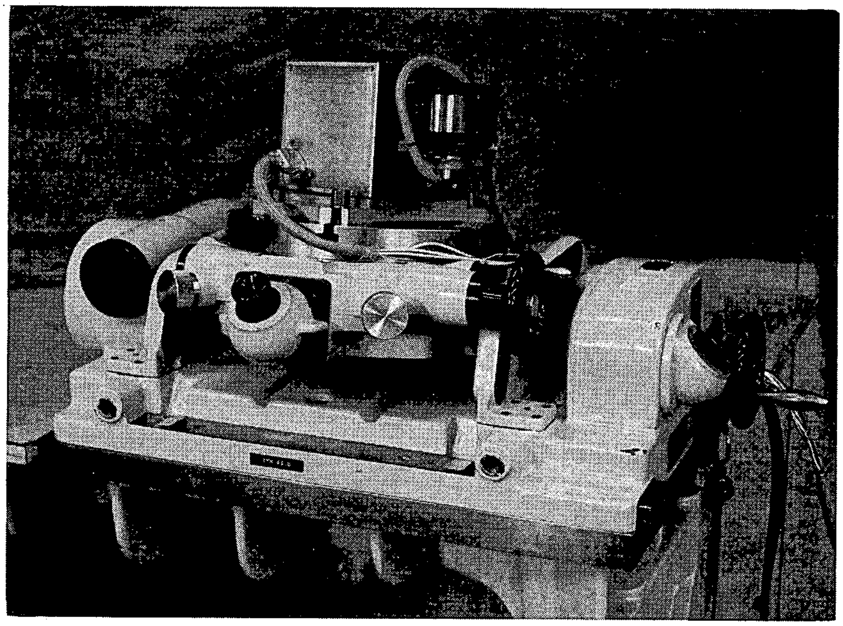


FIG. 73: CALIBRATION SET UP OF THE THREE ACCELEROMETERS.

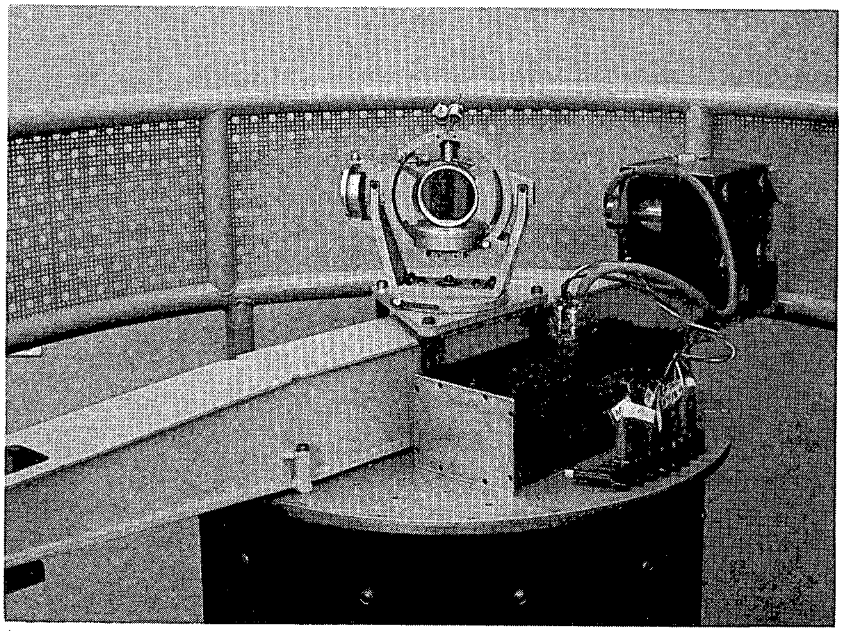


FIG. 74: CALIBRATION SET UP OF THE PITCHRATE GYRO.

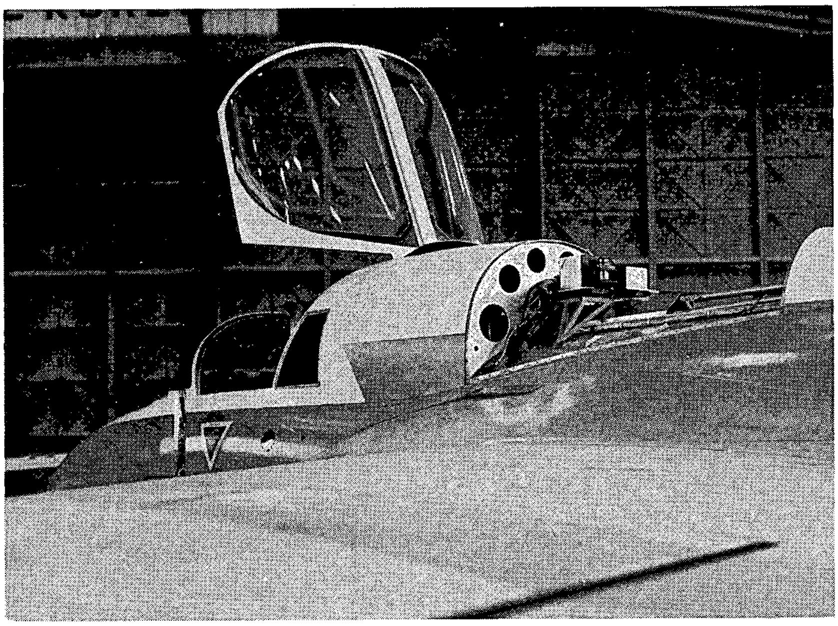


FIG. 75 : DORSAL MOUNTING OF ACCELEROMETERS AND PITCHRATE GYRO.

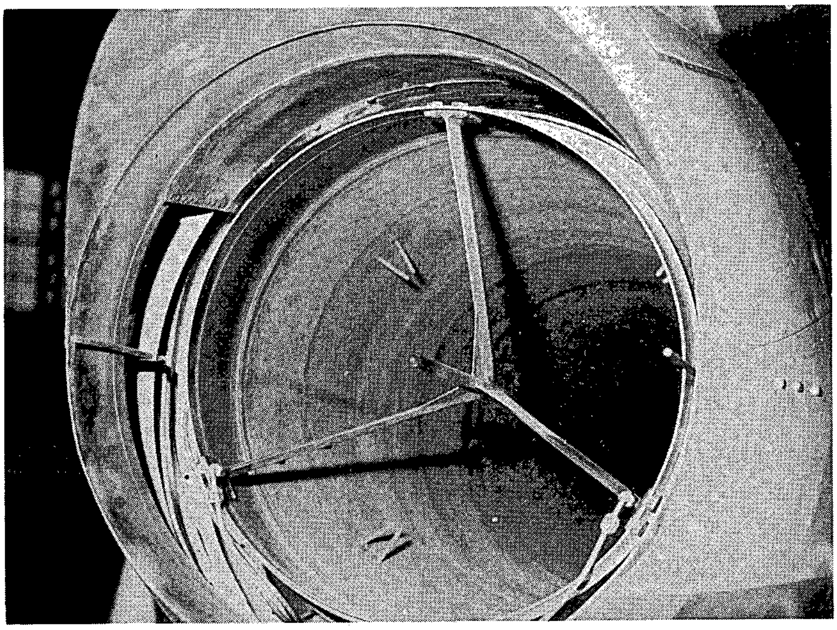
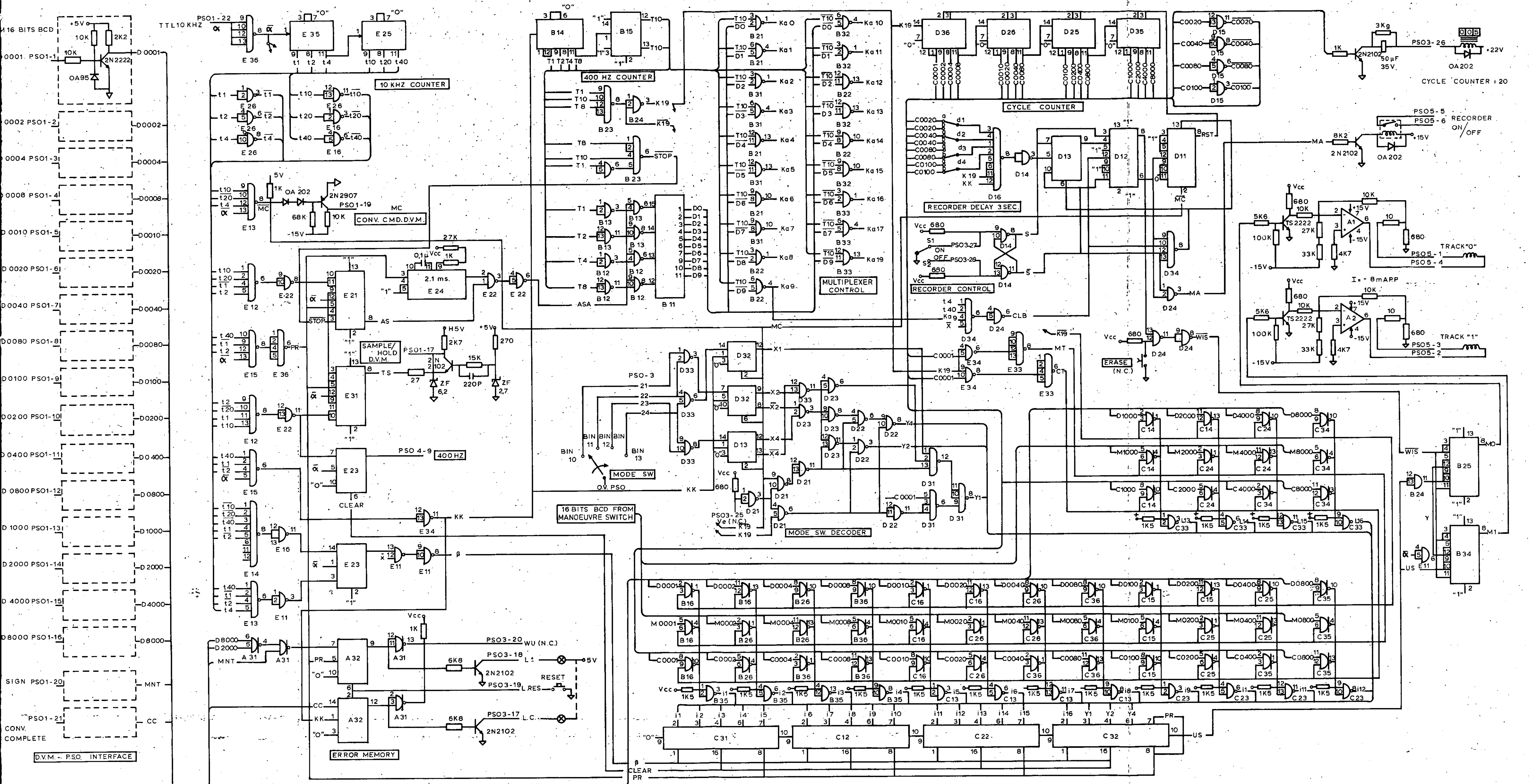


FIG. 76 : MEASUREMENT PROBES IN THE JET PIPE.



A1 µA 741											
A2 µA 741											
B11 SN 5442 N	B21 SN 5402 N	B31 SN 5402 N	C11	C21	C31 SN 8496 N	D11 SN 5472 N	D21 SN 5400 N	D31 SN 5410 N	E11 SN 5400 N	E21 SN 5472 N	E31 SN 5472 N
B12 SN 5400 N	B22 SN 5402 N	B32 SN 5402 N	C12 SN 8496 N	C22 SN 8496 N	C32 SN 8496 N	D12 SN 5472 N	D22 SN 5400 N	D32 SN 5473 N	E12 SN 5420 N	E22 SN 5400 N	E32
B13 SN 5400 N	B23 SN 5440 N	B33 SN 5402 N	C13 SN 5400 N	C23 SN 5400 N	C33 SN 5400 N	D13 SN 5473 N	D23 SN 5400 N	D33 SN 5400 N	E13 SN 5420 N	E23 SN 5473 N	E33 SN 5440 N
B14 SN 8490 N	B24 SN 5400 N	B34 SN 5472 N	C14 SN 5401 N	C24 SN 5401 N	C34 SN 5401 N	D14 SN 5400 N	D24 SN 5400 N	D34 SN 5420 N	E14 SN 5430 N	E24 SN 5421 N	E34 SN 5400 N
B15 SN 5473 N	B25 SN 5472 N	B35 SN 5400 N	C15 SN 5401 N	C25 SN 5401 N	C35 SN 5401 N	D15 SN 5400 N	D25 SN 8490 N	D35 SN 8490 N	E15 SN 5420 N	E25 SN 8490 N	E35 SN 8490 N
B16 SN 5401 N	B26 SN 5401 N	B36 SN 5401 N	C16 SN 5401 N	C26 SN 5401 N	C36 SN 5401 N	D16 SN 5430 N	D26 SN 8490 N	D36 SN 8490 N	E16 SN 5400 N	E26 SN 5400 N	E36 SN 5440 N

Fig. 49 SCHEME OF THE PARALLEL TO SERIES CONVERTER.