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The Drift of an Auto-Pilot Gyroscope Due to Prolonged Acceleration in the Skylark Rocket (Abridged Version)

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THE DRIFT OF AN AUTO-PILOT GYROSCOPE DUE TO PROLONGED ACCELERATION IN THE SKYLARK ROCKET

(Abridged version)

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SUMMARY

The method used to investigate the drift characteristics of a typical two-axis auto-pilot gyroscope when subjected to prolonged acceleration during flight in a Skylark rocket is described.

Rocket attitude obtained from the gyroscopes under test is compared with rocket attitude determined from other instrumentation systems to obtain gyro-drift. The principal reference system used to determine missile attitude is provided by three radio-interferometers fitted to the rocket. A secondary reference system comprises an aerial-survey camera which is installed in the vehicle and used to photograph ground markers.

The instrumentation referred to has been developed and flown in a series of Skylark rockets. The programme of firings is summarised and the accuracy of the instrumentation systems assessed. Ground equipments used to carry out pre-flight drift measurements on the gyroscopes under test are also briefly described.

Typical gyro-drift data obtained from flight results and measurements made before flight are given, together with conclusions based on data from twelve gyroscopes tested during the programme.

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CONTENTS

1 INTRODUCTION

Drift of the spin-axis of a two-axis gyroscope is due to error torques about the gimbal axes. These can be caused as follows:-

(a) Electromagnetic torques due to pick-offs on the gimbals.

(b) Windage effects from the spinning rotor.

(c) Friction in the gimbal bearings.

(d) Mass unbalance effects.

(e) Aniso-elasticity in the structure of the gyroscope.

(f) A random drift can also occur if temperature gradients exist in the structure of the gyroscope.

Error torques caused by (a) and (b) above are constant, but torques due to (c), (d) and (e) are all affected by acceleration of the gyroscope. The aim of the programme described in this Report was to investigate the effect of a prolonged and high acceleration on the drift characteristics of a typical missile auto-pilot gyroscope.

It is not easy to apply a unidirectional acceleration of several 'g' to a gyroscope in the laboratory. To overcome this difficulty Skylark rockets were used to apply a reasonably constant acceleration to a number of gyroscopes for a period of 30 to 40 seconds. The type of instrument selected for these experiments was a specially modified version of the two-axis gyroscope GW5 Mk.4 known as GW5 Mk.3. It was estimated that each gyroscope would show a total drift of approximately 50 arc minutes during the acceleration period of the Skylark rocket.

Drift of the spin-axis of the gyroscope can be determined by comparing missile attitude obtained from the gyroscope with missile attitude obtained from other sources. It was considered possible to measure the attitude of the rocket and therefore the frame of the gyroscope during flight to an accuracy of better than ±10 arc minutes by two other systems. The first system comprises three radio-interferometers which are fitted to the Skylark rocket and illuminated by two down-range transmitters during flight. In the second system a camera is fitted in the rocket and arranged to photograph accurately surveyed ground markers. Missile attitude during flight can then be derived from the recovered camera record. Data from the gyroscopes under test and the radio-interferometers are telemetered.

2 DESCRIPTION OF THE GYROSCOPES UNDER TEST

The GW5 Mk.3 gyroscope is a two-axis gyro which is manufactured by Reid and Sigrist Limited. The star-connected rotor requires a 3 phase 400 c/s 115 volt power supply of 8 watts (approximately) and has a normal running speed of 22800 rev/min. Each gimbal is fitted with an ac pick-off which is energised by a 36 volt 1300 c/s power supply. A voltage output proportional to gimbal position is obtained from each pick-off by means of a demodulator which is connected to the output of the ac pick-off and is provided with a reference signal from the 36 volt ac power supply. Movement of the gyro spin-axis is recorded during pre-flight laboratory tests by reflecting a beam of light from a mirror fixed to the inner gimbal onto a screen (see Fig.2). A window is provided in the cover of the gyroscope for this purpose (Fig.1). The drift rate of the gyroscope is specified by the manufacturer as being less than 15 deg/hr when tested in the earth's gravitational field (see Note (i)).

The gyroscope is basically the auto-pilot gyroscope GW5 Mk.4 with certain modifications to make it suitable for the gyro-drift experiments. These modifications are as follows:-

(a) The motorised caging mechanism of the instrument has been redesigned so that the spin-axis of the gyroscope can rotate through 360 degrees about the outer gimbal axis. This was necessary because the Skylark rocket may roll through more than 360 degrees during its flight (due to fin misalignment torques). Although the principal gyro-drift experiments have been carried out with Skylark rockets fitted with a roll-stabilised head (see section 4) the roll control unit was not available for some of the earlier instrumentation proving trials[1]. In addition, the 360 degrees of freedom about the outer gimbal axis allows the gyroscope to be tested on a turn-table in the

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Note (i)

The gyroscope is tested by the manufacturer with the base of the instrument horizontal to prevent lateral forces due to gravity components acting on the gyro during the test. The instrument is mounted on a horizontal table with the spin-axis pointing "mirror north" so that the earth rate component about the inner gimbal axis is eliminated. The table is then driven at a constant angular velocity of $\Omega \sin \eta$ about a vertical axis to eliminate the effect of earth rotation about the outer gimbal axis. In this manner an acceleration of 1 g is applied to the gyro in the required direction and the effect of earth rotation can be neglected provided the duration of the test does not exceed a few minutes (Ω = earth rate of 15 deg/hr, η = latitude in degrees).

laboratory[2]. Movement of the spin-axis of the gyroscope about the inner gimbal axis is limited to ±70 degrees (approximately) which is more than sufficient to accommodate the largest pitch and yaw angles of the rocket during powered flight.

(b) It was considered that the major part of the drift would be due to mass unbalance and would result in movement about the outer gimbal axis because each gyroscope is fitted in the rocket with the spin-axis of the rotor normal to the direction in which maximum acceleration occurs (Fig.3). An optical pick-off was therefore developed by G.E.C. Stanmore[3] which gave a more accurate measurement of drift about the outer gimbal axis than could be obtained from the ac pick-off.

A light alloy vane is attached to the outer gimbal of the gyroscope and arranged to pass between an illuminated slit and two photo-transistors fixed to the frame of the gyroscope (see Fig.4). Radial slits in the vane produce a train of pulses from the photo-transistors as the outer gimbal rotates through the datum position indicated by the positive pulse (see Fig.5). The radial slits in the vane occur at intervals of 2 degrees and extend ±16 degrees on either side of the datum position. Each slit produces a pulse having an amplitude of about 2 volts and a pulse-width at the half amplitude points of 10 arc minutes (approximately). It can be seen from Fig.6 that at the instant when a pulse occurs the position of the outer gimbal can be determined to an accuracy of ±1 arc minute. The pick-off does not give a continuous output but it does give an accurate indication of the rotation of the spin-axis of the gyroscope about the outer gimbal axis at intervals of two degrees near the caged position. The large pulses at the beginning and end of the pulse train (Fig.5) are caused by the edges of the vane.

(c) To improve the repeatability of the drift characteristic of the gyroscopes when tested in the laboratory the working clearances in the inner and outer gimbal bearings are kept to a minimum when the gyroscopes are assembled[2].

(d) The rotor bearings are preloaded to reduce the warm-up time required before repeatable drift results can be obtained from the gyroscopes when tested in the laboratory[2].

3 INSTRUMENTATION SYSTEMS FOR DETERMINING MISSILE ATTITUDE

As already indicated in section 1 the drift of the GW5 Mk.3 gyroscope when subjected to acceleration is determined by comparing missile attitude obtained from gyroscopes of this type with missile attitude determined by using two other instrumentation techniques; a radio-interferometer system and a photographic method. A description of all three instrumentation systems is given in the following sections.

3.1 Gyroscopes under test

Two GW5 Mk.3 gyroscopes are fitted in the interferometer-bay of the test vehicle (see section 6 and Fig.17) with their spin-axes perpendicular to the longitudinal axis of the missile as shown in Fig.3. The lower gyroscope (position 1) is arranged to measure attitude of the head of the missile about the yaw and roll axes whilst the upper gyroscope (position 2) is arranged to measure attitude about the pitch and roll axes. The outer gimbal pick-offs measure roll attitude in the case of both gyroscopes under test. The voltage outputs from the photo-electric pick-offs on the outer gimbals, and from the demodulators associated with the ac pick-offs on the inner and outer gimbals are telemetered by an R.A.E. sub-miniature telemetry sender. (Telemetry Sender 1, see section 5.)

The ac pick-offs provide a continuous record of the attitude of the head of the missile but with limited accuracy. A more accurate measurement of attitude about the roll axis is obtained from the photo-electric pick-offs, but as already explained (see section 2) these measurements are only available at certain angles i.e. every 2 degrees. A programmed oscillation about the roll axis, having an amplitude of 5 degrees, is therefore superimposed on the roll-stabilised head of the missile. Since the period of the oscillation is 4 seconds a datum pulse (positive, see Fig.5) is obtained every 2 seconds as the illuminated slit on the frame of the gyroscope moves past the datum slot in the vane fitted to the outer gimbal (Fig.4). Two pulses on either side of the datum pulse (Fig.5) are also transmitted from each photo-electric pick-off during a complete oscillation of the head of the missile as the light source scans the slotted vane. A pulse is therefore transmitted from the photo-electric pick-off on the outer gimbal of each gyroscope every 0.5 second, giving the position of the head of the missile about the roll axis to an accuracy of ±1 arc minute at each of these instants in time.

The accuracy of the telemetered data obtained from both types of pick-off is discussed in more detail in section 8.1.

The attitude of the missile is measured in the launcher[4], so that the caged positions of the gyroscopes are known and the total drifts from launch can be computed. Gyro-drift rates however, are determined from some arbitrary datum when the rocket has reached an altitude of not less than 5000 ft. The reasons for this are as follows:-

(a) Drift of the gyroscopes is affected by other factors in addition to acceleration during the launch phase, such as:

(i) Movement of the spin-axes on uncaging the gyroscopes.

(ii) Drift during the period between uncaging the gyroscopes and launch.

(iii) Shock and vibration as the rocket moves along the launcher rails (length 90 ft approximately).

(b) Data obtained from the interferometers and camera at rocket altitudes below about 5000 ft are not sufficiently accurate.

3.2 Radio-interferometers

3.2.1 Principles of the interferometer system

An elementary radio-interferometer is shown in Fig.7. It comprises two receiving aerials which are spaced by a distance ℓ known as the base length. If a plane polarised electromagnetic wave is incident upon the interferometer so that the direction of propagation makes an angle θ with the base of the interferometer, then assuming that the distance between the aerials is negligible compared with the distance between the interferometer and transmitter we have from Fig.7.

$$A_2B = \ell \cos \theta \text{ cm}$$

where ℓ = interferometer base length (cm)

also A_2B = $2\pi n \pm \phi$ radians of electrical phase

where ϕ = phase difference at the two aerials (rad)

and n = number of whole wavelengths

therefore,

$$2\pi \left(\frac{\ell \cos \theta}{\lambda}\right) = 2\pi n \pm \phi$$

where λ = wavelength of signal from transmitter (cm)
therefore,

$$\cos\theta = \frac{\lambda}{\ell}\left(n \pm \frac{\phi}{2\pi}\right) \quad . \tag{1}$$

Hence the angle θ which the interferometer base makes with the sight-line from the transmitter can be obtained from a knowledge of ϕ and n.

Differentiating ϕ with respect to θ in expression (1) gives the magnification factor of the interferometer as follows:

$$\frac{d\phi}{d\theta} = \frac{2\pi\ell}{\lambda} \sin\theta \quad . \tag{2}$$

It can be seen from expression (2) that small variations in angle θ are magnified by the interferometer to give large changes in the electrical phase angle ϕ, e.g. if $\ell = 10\,\lambda$ and the interferometer base is normal to the sight-line from the transmitter so that $\sin\theta = 1$ then the magnification is about 60 (one minute of arc becomes one degree of electrical phase). Expression (2) was used to plot the graph shown in Fig.8 which shows how the magnification factor is reduced as the angle θ becomes less than 90 degrees. Provided θ does not fall below 65 degrees during flight, the magnification factor will not drop below 57. It can be seen from section 8.2 that this range of magnification is acceptable as the error introduced by the telemetry will then have little effect on the overall accuracy of the interferometer system.

At this magnification ambiguities occur in the output from the interferometer receiver at intervals of approximately 6 degrees at normal incidence. A second method of attitude measurement is therefore necessary to resolve these ambiguities and in the case of the gyro-drift experiments this is provided by the gyroscopes under test as the drift of the gyros is normally much less than the interval between ambiguities in the outputs from the interferometer receivers, i.e. the telemetered voltage from the ac pick-offs on the gyroscopes can be used to resolve the value of the integer n in expression (1). In practice the gyro outputs indicate which six degree section of the graph shown in Fig.8 is to be used when the interferometer records are analysed.

If a receiver is designed to give an output which is a function of the phase difference ϕ between the signals received at the two aerials of an interferometer then it can be seen from expression (1) that the angle θ between the interferometer base and the sight-line from a transmitter can be obtained. If the interferometer base is arranged to be parallel to the longitudinal axis of a missile then the interferometer angle θ will define the angle which the missile axis makes with the sight-line from the transmitter. The missile axis will therefore lie along the surface of a cone of semi-angle θ having the sight-line from the transmitter to the rocket as axis. If the interferometer is illuminated by a second transmitter then a second output from the interferometer receiver will define the position of the missile axis on the surface of a second cone. The intersection of the two cones gives the attitude of the missile axis with one ambiguity which can be eliminated from a knowledge of the missile trajectory. If a second missile-borne interferometer, with its base length lying in a plane which is perpendicular to the missile axis, is illuminated by either of the two transmitters referred to above, the rotation of the missile about its longitudinal axis can be determined and the attitude of the missile is completely defined.

In order to determine the continuously changing directions of the sight-lines from the transmitters to the missile it is necessary to record the missile trajectory and survey the positions of the two transmitters accurately in terms of range co-ordinates.

3.2.2 Description of the interferometer system

The interferometer system for the Skylark rocket operates at X-band (9.5 Gc/s approximately) and was designed and built at the Applied Electronics Laboratories, G.E.C. Stanmore. The "interferometer-bay" of the missile which is fitted with the interferometer aerials, waveguides and receivers is shown in Fig.9. Aerial A_0 is common to the longitudinal interferometer $(A_0\ A_2)$ and the lateral interferometer $(A_0\ A_1)$. The base length of each interferometer is 12 inches, which is approximately 10 λ at the operating frequency of the interferometer system.

Fig.3 shows the bearings of Transmitter 1 (frequency 9560 Mc/s) and Transmitter 2 (frecuency 9540 Mc/s) relative to the interferometer aerials when the missile is in the launcher at Woomera. Both transmitters are about 25 miles down-range from the Skylark launcher and approximately 25 miles to the left and right (Transmitters 1 and 2 respectively) of the down-range bearing through the launcher. The polar diagram of the aerial system of each transmitter is horizontally polarised and has a beam width of about 8 degrees in azimuth and 38 degrees in elevation. This is sufficient to cover the whole of the missile flight path during the propulsion phase. A null in the polar diagram, three degrees below the horizontal, minimises the effect of ground reflections at the interferometer receivers in the missile. Uninterrupted sight-lines exist between the longitudinal interferometer aerials and both transmitters for a missile roll angle of 90 degrees. Transmitter 2 also illuminates the lateral interferometer aerials throughout the same 90 degrees of roll.

Each transmitter is fitted with a power amplifier (VX 3247 klystron, air and water cooled) which amplifies the signal generated by a cavity stabilised oscillator. The oscillator frequency can be set to an accuracy of ±100 kc/s and the maximum power output from the transmitter is 900 watts CW. Fig.10 shows the interior of one of the six ton, air-conditioned trailers in which each transmitter is installed. The positions of the transmitters and the Skylark launcher in terms of range co-ordinates are given in Appendix A. In the case of the transmitters, all measurements are made from the range origin to the throat of the horn aerial. The position of the launcher is taken as a point on the ground at the centre of the tripod structure which supports the launcher.

A block diagram of a single-channel interferometer receiver is shown in Fig.11. The signals received at the two aerials from a ground based transmitter are out of phase by an angle ϕ which depends on the missile attitude. Both signals are converted to an intermediate frequency by mixing with a local oscillator signal. A microwave single side-band low frequency modulator is used to add 800 c/s (approximately) to the local oscillator signal fed to the mixer associated with aerial A_0, while the signal from the second aerial A_2 is mixed directly with the local oscillator signal. The modulator also generates phase reference signals at the modulation frequency of 800 c/s.

The signals from the two mixers are added and the result is an I.F. carrier which is amplitude modulated at the frequency of the single side-band modulator. This is detected after amplification to give a signal at the modulator frequency (800 c/s approximately). A narrow-band low frequency amplifier follows the detector to improve the signal to noise ratio at the phase sensitive rectifiers. The phase difference between the signal from the detector and the reference signal generated by the single side-band modulator is the phase difference which exists between the signals received at the two aerials (A_0 and A_2). The phase difference is obtained from two phase-sensitive rectifiers in which the signal from the receiver is compared with sine and cosine functions of the reference signal. The voltage outputs from the phase-sensitive rectifiers are proportional to the sine and cosine of the phase angle ϕ.

A second receiver channel is added to the A_0 A_2 interferometer by duplicating the equipment from the I.F. amplifier onwards (Fig.12) because angles relative to the sight-lines from two transmitters (1 and 2) are required from the longitudinal interferometer. The centre frequencies of the two I.F. amplifiers connected to this interferometer are separated by 20 Mc/s to avoid interference. One receiver operates at an I.F. of 60 Mc/s and the other at 40 Mc/s.

The lateral interferometer which is formed by the common aerial A_0 and aerial A_1 (Fig.9) measures the angle of the interferometer base relative to the sight-line from Transmitter 2. A third receiver which operates at an I.F. of 40 Mc/s is connected to this interferometer.

The complete interferometer system is summarised in Table 1.

Table 1

Transmitter			Interferometer receiver			
No.	Site	Frequency Mc/s	No.	Base position on missile	Aerials	Receiver I.F. Mc/s
1	Coondambo	9560	H.21	Longitudinal	A_0 A_2	60
2	P.5	9540	H.22	Longitudinal	A_0 A_2	40
2	P.5	9540	H.12	Lateral	A_0 A_1	40

Two voltage outputs are obtained from each of the three interferometer receivers tabulated above. One is proportional to $\sin \phi$ and the other is proportion to $\cos \phi$ so that the angle ϕ is completely defined throughout a phase change of 360 electrical degrees for each interferometer (see Note (ii)). The six voltage outputs from the phase sensitive rectifiers are telemetered on separate channels of telemetry sender 1. The maximum voltage output from the phase sensitive rectifiers is about 5 volts peak to peak and the interferometer system as a whole has been designed to give usable voltage outputs of this order at missile altitudes of up to 100 000 feet.

Four signals from the reference generator are connected, in turn, to the input of each of the low frequency amplifiers at intervals during flight so that phase shifts which may occur in the low frequency amplifiers are checked. These signals are in phase and in anti-phase with the two reference signals supplied to the phase sensitive rectifiers, i.e. $e_r \sin \omega t$ and $e_r \cos \omega t$ (Fig.12).

The range of magnification factors obtained from the longitudinal and lateral interferometers during the propulsion phase of the rocket flight is shown by the limits indicated in Fig.8. These are based on the assumptions that the interferometers are fitted in a roll-stabilised head with a roll programme of ±5 degrees superimposed, and that the launcher bearing and aerial positions at the roll datum are as shown in Fig.3. The method used to determine missile attitude from this information is given in more detail in Refs.5 and 6.

3.3 Photographic method

3.3.1 General description

The attitude of a camera can be determined from a photograph taken by the camera in which two points of known co-ordinates can be identified[7]. If the camera is fitted in a missile and a system of accurately surveyed ground markers are arranged so that a series of pictures of different pairs of markers are obtained as the rocket moves along its trajectory, then missile attitude can be derived from each frame of the recovered camera record provided the missile position at each exposure is known in terms of range co-ordinates.

Note (ii)

When the telemetered outputs from each interferometer receiver are analysed the value of the phase angle ϕ is determined from the function of ϕ which is changing at the higher rate. i.e. If the value of ϕ is close to 0 or 180 degrees $\sin \phi$ is used, if closer to 90 or 270 degrees the value is obtained from $\cos \phi$.

If the sight-line from a ground marker to the camera makes an angle β with the optical axis of the camera, and if the focal length is known, the angle β can be determined by measuring the distance of the image from the centre point on the photograph which represents the optical axis of the camera. The optical axis can then be assumed to lie somewhere on the surface of a cone of semi-angle β having an axis given by the sight-line from the missile to the ground marker. A second angle and therefore a second conical surface in which the optical axis of the camera must lie can be obtained from the image of a second ground marker. The intersection of these two cones will give the attitude of the optical axis of the camera with one ambiguity which can be resolved from a knowledge of the vehicle trajectory or from the photograph itself. Since the position of each ground marker on the photograph fixes the angular position of the camera about its optical axis, the attitude of the camera can be resolved. Missile attitude follows from the attitude of the camera. The method of determining missile attitude from a camera record is given in more detail in Refs.6 and 7.

3.3.2 Camera installation and operation

The camera fitted in the vehicle (F117B aerial-survey camera) has a field of view of about 40 degrees in the planes which include the principal axes of the register plate (U_o and V_o Fig.13), and a field of view of approximately 55 degrees across the diagonals. The camera is fitted in the missile as shown in Fig.14 so that it photographs a line of ground markers through an aperture in the side of the vehicle during the propulsion phase. With the missile in the launcher the optical axis of the camera is arranged to lie in a vertical plane which includes the launcher bearing 313 degrees (see Fig.3). A typical camera installation is shown in Fig.15. An access panel is provided in the side of the "camera-bay" so that the film magazine can be changed without moving the camera.

The vehicle is fired in daylight and the camera runs continuously as a cine-camera. The method of photographing ground markers from the missile is shown diagrammatically in Fig.16 which includes some of the ground markers referred to in section 7 and Appendix A. The total running time of the camera is approximately 1 minute and one frame of the film is exposed every 0.5 second (approximately). The exposure time is kept to a minimum to reduce movement of each ground marker image across the negative, due to motion of the rocket, and so the fastest available shutter speed (1/400 second) is used.

With Kodak Plus X Aerographic film and a haze filter the smallest available aperture setting (f.22) was found to be appropriate for the ambient lighting conditions.

3.3.3 Camera calibration

To reduce the effect of film stretch and distortion in the optical system of the camera when the record is analysed, the camera is fitted with a glass register plate which is engraved with a 1 cm grid of reseau marks (see Fig.13). The register plate is immediately in front of the film, and so these marks appear on each negative and provide a set of reference points from which the effect of film stretch (which may occur due to tension in the film as it is "wound-on" in flight) can be compensated. Distortion in the optical system of the camera is also measured at each of these points before the camera is used (see Appendix B), so that a correction can be applied to the data obtained from the film record. The attitude of negatives obtained from the camera record must be correlated with rocket axes. For this, additional identification marks (i.e. letters A and S, Fig.13) are engraved on the edges of the glass register plate, and after the camera has been fitted in the camera-bay, the complete unit is mounted on a machined surface and the attitude of the register plate is measured using height gauges, a clinometer and a travelling microscope.

4 ROLL CONTROL UNIT

The Skylark test vehicle is a fin-stabilised rocket which can roll in flight at roll rates of up to 1 rev/sec. To carry out the gyro-drift experiments it is necessary to provide the test vehicle with a roll-stabilised head for the following reasons:-

(a) A unidirectional friction torque about the outer gimbal axis of a gyroscope due to the roll of a rocket about its longitudinal axis will cause a high precession rate about the inner gimbal axis. Although of general interest, the drift of a gyroscope under these conditions is not representative of the drift which will occur when the gyroscope forms part of an auto-pilot in a roll-stabilised vehicle.

(b) The sight-lines from the two ground based transmitters to the interferometer aerials on the missile must not be interrupted during powered flight by rotations of the vehicle about its longitudinal axis.

(c) The ground markers, which are photographed from the missile during the motor burning phase, are set out along the down-range bearing through the launcher (304° 42.7', see Fig.3). It is therefore necessary that this bearing shall remain within the field of view of the camera during the propulsion period. Since the effect of lens distortion is greater at the edges of the camera's field of view, it is also desirable that the down-range bearing 304° 42.7' shall remain somewhere near the centre of the field of view of the camera, i.e. adjacent to the U_o axis of the register plate.

The illuminated slit fitted to the frame of each gyroscope must continuously scan the slotted vane attached to the outer gimbal so that datum pulses are transmitted at frequent intervals (see section 3.1). This is achieved by superimposing a programmed oscillation about the roll axis of the roll-stabilised head, having an amplitude of 5 degrees and a period of 4 seconds. This particular amplitude and frequency of oscillation was chosen because it gives an adequate sampling rate from the pick-off without having an adverse effect on the other systems of attitude measurement.

The head of the missile is supported on a stub-axle as shown in Fig.17 so that it is free to rotate relative to the rocket motor. An electrically operated servo-motor is used to control the roll position of the head of the missile by providing a drive between the rocket motor, with its relatively high moment of inertia in roll, and the stabilised head. The roll datum of the control system is conveniently defined by the voltage output from the demodulator of the ac pick-off fitted to the outer gimbal of one of the gyroscopes under test. (Yaw gyro, position 1, Fig.3.) The initial design of the roll control unit is described in Ref.8. It was found necessary to increase the control torque available however, during the firing programme (see section 6).

To protect the telemetry and Doppler aerials when the missile is in the launcher, the head of the vehicle is prevented from rotating relative to the rocket motor by a key which locks the head to the motor. The key is tripped out of the locked position as the vehicle leaves the launcher. Roll control and the programmed oscillation of the head about the roll-axis of the missile does not commence until 3 seconds after launch.

5 TELEMETRY, DOPPLER AND M.T.S.

The complete head of the gyro-drift test vehicle is illustrated in Fig.17. Not all the instrumentation shown was carried by vehicles fired early in the programme (see section 6). In its final form, the rocket is fitted with two R.A.E. sub-miniature telemetry transmitters. A 24-channel 85 c/s multiplexing switch is included in sender 1 (frequency 453 Mc/s) which is arranged to telemeter the voltage outputs from the ac pick-offs and photo-electric pick-offs fitted to the two gyroscopes under test. Interferometer measurements transmitted by this sender include the voltage outputs proportional to $\sin\phi$ and $\cos\phi$ from each of the three interferometer receivers, and the skin temperature between the aerials of each interferometer (so that a correction can be applied for expansion of the base length due to the effect of kinetic heating). Telemetry channels are also provided for monitoring the operation of the roll control unit and for transmitting the voltage outputs from transducers such as rate gyroscopes and accelerometers.

Telemetry transmitter 2 (frequency 443 Mc/s) comprises a single-channel sender which is used to monitor the operation of the shutter of the camera fitted in the vehicle. A voltage pulse generated by an electro-magnet, each time the shutter operates, is telemetered so that it is possible to correlate each frame with a particular instant in time. The camera is also arranged to photograph a light source when the missile is stationary in the launcher so that the datum frame, as the rocket begins to move, can be located on the recovered camera record. (The camera is started at 8 seconds before launch.)

Since trajectory data for the experiment is only required during the motor burning phase which normally terminates at an altitude of 70000 ft (approximately), this can be obtained from optical trackers. A Doppler transponder has been fitted in some rockets however so that trajectory data could also be obtained from multi-station Doppler. To aid recovery of the instrumented head and camera, an M.T.S. beacon (see Note (iii)) is also fitted in the vehicle to give long range trajectory information and indicate the position of the impact point of the head.

6 SUMMARY OF FIRINGS

Two gyroscopes under test, an interferometer-bay and a parachute recovery system were fitted to the roll-stabilised head of each Skylark rocket

Note (iii)

The M.T.S. beacon forms part of a Missile Tracking System which is used at the Woomera rocket range. The system operates at a wavelength of 6.5 cm (approximately).

i.e. SL 27 to SL 32. Head separation from the rocket motor was arranged to take place at 168 seconds after launch in each case. Additional instrumentation fitted in the heads of these vehicles and the principal characteristics of the roll control units are given below in the order of firing (see Table 2).

Table 2

Vehicle number	Camera bay	Doppler transponder	M.T.S. beacon	Roll control unit	
				Max. torque lb ft (nom.)	Max. roll rate rev/sec
SL 28	Not fitted	Fitted	Fitted	9	1.00
SL 27	Not fitted	Not fitted	Fitted	18	0.50
SL 29	Fitted	Not fitted	Fitted	18	0.50
SL 30	Fitted	Not fitted	Fitted	18	0.50
SL 31	Fitted	Fitted	Fitted	27	0.33
SL 32	Fitted	Fitted	Fitted	27	0.33

During the firing programme it was found necessary to increase the maximum torque available from the roll control unit. The effect of increasing the torque was to reduce the bandwidth of the control system and therefore reduce the maximum roll rate of the rocket motor which could be cancelled by the roll control unit. To ensure that the roll rate of the motor did not exceed the figure given in Table 2, fin assemblies were selected for all vehicles fired after SL 28 to give them a predicted maximum roll rate of less than 0.1 rev/sec. Telemetry transducers such as rate gyroscopes and accelerometers which were also fitted in these vehicles are not tabulated above.

7 GROUND MARKERS FOR THE PHOTOGRAPHIC METHOD

Ten ground markers for the photographic method of measuring missile attitude were set out along the down-range bearing 304° 42.7' (see Fig.3) so that at least two markers would appear in each photograph. Two extra sites, 4A and 7A were also included as "Sequence Identification" markers so that each site could be identified on the recovered camera record. Each ground marker was in the form of a cross (white in colour) which measured 80 ft from tip to tip. The position of each site is given in Appendix A.

8 ACCURACY OF THE INSTRUMENTATION SYSTEMS

During the course of the work every effort was made to assess the expected accuracy of the systems of measuring missile attitude in order to ensure that the errors in the photographic and interferometer methods would be much less than the gyro-drift to be measured.

Drift of the spin-axis of the GW5 Mk.3 gyroscope during the acceleration period of the Skylark rocket was estimated to be about 50 arc minutes and the majority of the drift was expected to occur about the outer gimbal axis of the gyroscope (see sections 1 and 2). It was therefore necessary to ensure that the resolution of the instrumentation systems was considerably better than 50 arc minutes; particularly in the case of attitude measurement about the roll axis of the missile.

8.1 Gyroscope outputs

Two types of pick-off are fitted to the gimbals of the GW5 Mk.3 gyroscope as described in section 3.1.

(a) Photo-electric pick-off

The accuracy of the telemetered data obtained from the photo-electric pick-off fitted to the outer gimbal of the gyroscope depends upon the pulse shape shown in Fig.6. The voltage output from the pick-off is transmitted on three channels of telemetry sender 1 so that at least 10 samples of each pulse are telemetered. Since the base length of each pulse represents approximately 20 arc minutes, the peak of the pulse can be resolved to an accuracy of about ±1 arc minute.

(b) ac pick-off

Both the inner and outer gimbals of the gyroscope are also fitted with ac pick-offs and demodulators. The trajectory of the vehicle is such that a heading change in pitch of up to 12 degrees (approximately) can take place between the attitude of the missile in the launcher, when the gyroscopes are uncaged, and the attitude of the missile at motor burn-out. Full modulation of the telemetry system occurs at input signals of ±3 volts, so the sensitivity of the demodulators is set at 0.25 volt/degree in order that the system will not saturate before the end of the acceleration period.

The error due to non-linearity of the telemetry modulator is within ±1% of the full scale deflection of 24 degrees (160 kc/s to 130 kc/s), and reading errors associated with the interpretation of the telemetry record are also less than ±1%, therefore the accuracy of the data from this pick-off is within ±21 arc minutes. By considering several samples and smoothing the data it is possible to greatly reduce the reading error and an accuracy of approximately ±15 arc minutes is achieved.

8.2 Interferometer system

Both systematic and random errors occur in the measurements obtained from the interferometer system. Of the former, only the error due to the 3 milli-second time delay in the phase sensitive rectifiers (see Fig.12 and Ref.10) is thought to be of significance. Appendix C shows estimates of the 3 σ values of the random errors in the steady-state, assuming normal distributions. Telemetry non-linearity errors and reading errors of ±1% (see section 8.1) are each equivalent to attitude errors of ±2 arc minutes with the particular scalings adopted for the trials (see Note (iv)). The steady-state errors of the interferometer system (Appendix C) and the errors introduced by the telemetry therefore amount to approximately ±5 arc minutes.

The limited frequency response of the receivers has negligible effect on the accuracy of the voltage outputs from the two receivers connected to the longitudinal interferometer, because rocket heading angles change relatively slowly. Thus the maximum error in the telemetered data from these receivers is approximately ±5 arc minutes. However, the voltage outputs from the lateral interferometer receiver are affected by the higher angular rates caused by the programmed oscillation in roll. With an amplitude of 5 degrees and a period of 4 seconds, the maximum angular velocity of the head is 0.14 rad/sec (approximately). At this roll rate, the time delay of 3 milli-seconds in the phase sensitive rectifiers of the receiver causes an additional measuring error of 1.4 arc minutes. Therefore the total error (steady-state plus dynamic) in the data from the lateral interferometer may reach approximately ±6.5 arc minutes.

Note (iv)

Since the voltage outputs from the interferometer receivers give a full scale deflection of the telemetry system for a phase change of 180 degrees at the interferometer aerials, a 1% change in the telemetry signal represents a phase change of 2 degrees (approximately) or a change in missile attitude of 2 arc minutes. (A phase angle ϕ of about 45 degrees was considered where $\sin \phi$ and $\cos \phi$ are changing at the same rate.)

8.3 Photographic method

The main sources of error associated with the photographic method of determining missile attitude are as follows:-

(a) Spot size.

(i) Size of image on the film.

(ii) Elongation of image due to movement of the camera during exposure of the film.

(b) Distortion introduced by the optical system of the camera. (Effect reduced by calibration.)

(c) Film stretch caused by "wind-on" mechanism.

(d) Errors in the measurement of camera attitude in the missile.

(e) Errors in estimating the direction of sight-lines from ground markers to the missile.

(f) Refraction of sight-lines.

Spot size

To achieve maximum accuracy the image size on the film must be kept as near as possible to the minimum readable image.

(i) Size of image on film

The size of a ground marker which appears on the film record depends upon the angle subtended at the camera by the ground marker. This depends in turn upon the altitude of the rocket. A resolution of better than ±0.05 mm should be possible for all frames taken above an altitude of 15000 ft. Since the focal length of the Williamson 117B camera is approximately 15 cm the error in fixing the angular position of the image on the film record should be better than ±1 arc minute.

(ii) Elongation of image

Elongation of the image is due to angular motion of the optical axis of the camera during an exposure. With a duration of 2 milli-seconds for the exposure, the following expressions apply[7].

$$\text{Angular error due to forward velocity} = \frac{7V}{S} \text{ arc minutes} \qquad (3)$$

$$\text{Angular error due to roll} = 43R \text{ arc minutes} \qquad (4)$$

where V = forward velocity of missile (ft/sec)

S = length of sight-line (ft)

R = angular velocity of head (rev/sec).

Considering a forward velocity of 2000 ft/sec at an altitude of 15000 ft then from expression (3), the angular error due to forward velocity $\simeq$ 1 arc minute.

The maximum angular velocity of the camera due to the programmed oscillation superimposed on the roll-stabilised head (see section 8.2) $= \frac{0.14}{2\pi}$ rev/sec.

Hence from expression (4) the maximum error due to the angular velocity of the head $= \pm 43 \times \frac{0.14}{2\pi}$ arc minutes $\simeq \pm 1$ arc minute.

Distortion introduced by the optical system of the camera

The effect of distortion is reduced by using a grid of reseau marks on the register plate of the camera (see Fig.13) and calibrating the optical system of the camera at each of these marks as described in section 3.3.3 and Appendix B. The maximum error in the rectangular co-ordinates (U_o and V_o) of each reseau mark depends upon the accuracy of the marks on the master plate from which the register plate was engraved. In the case of the register plate fitted in the Williamson 117B camera, these errors do not exceed ±8 microns.

If it is assumed that the reseau marks provide a 1 cm reference grid when the film is analysed, the maximum error introduced at each reseau mark

$$= \pm \frac{0.008 \times 57.3 \times 60}{150} \text{ arc minutes}$$

$$= \pm 0.18 \text{ arc minutes.}$$

Film stretch

The reseau marks which appear on each frame of the film record provide a set of reference points which are used to reduce the effect of optical distortion (see section 3.3.3 and Appendix B). They also reduce the effect of film stretch due to tension in the film as it is "wound-on" in flight. Film stretch which is not constant per unit length of film may still take place between adjacent reseau marks however. An error may therefore be introduced when the position of an image on the film is determined by interpolating between reseau marks. The variation in film stretch is not known accurately but the best estimate which can be made indicates an error of ±2.5 arc minutes.

Measurement of camera attitude in the missile

The camera is fitted in the camera-bay of the missile as described in section 3.3.2 and illustrated in Figs.14 and 15. The camera-bay is then mounted on a machined surface which is horizontal to within ±0.75 arc minute and the attitudes of the principal axes of the register plate are recorded. A clinometer, travelling microscope and height gauges are used for these measurements. The error in determining the attitude of the camera register plate relative to the missile axes from these measurements is approximately ±1 arc minute.

Errors in estimating the direction of sight-lines

In order to obtain missile attitude from any pair of ground markers which appear on a given frame of the camera record, it is necessary to know the direction of the sight-lines from the ground markers to the rocket. The positions of the ground markers are known from survey data to an accuracy of ±3 ft in terms of range co-ordinates (see Appendix A). The position of the vehicle is required for a range of altitudes from 5000 ft to 75000 ft where motor burn-out occurs and acceleration ceases. This is determined from kinetheodolite and multi-station Doppler data, and it is estimated that over the greater part of this trajectory the directions of the sight-lines will be known to an accuracy of better than ±1 arc minute.

Refraction of sight-lines

Estimates of the refraction of the sight-lines, based on a standard atmosphere have been made and these indicate that the error introduced by this source is not likely to exceed ±3 arc minutes.

If the individual errors quoted above are assumed to represent 3 σ values of normal distributions, the accuracy of the photographic data from which missile attitude can be derived is better than ±4.5 arc minutes.

9 GYROSCOPE TEST EQUIPMENTS

(a) At R.A.E. (Farnborough, U.K.)

Two gyro test equipments are available in Space Department, R.A.E. The first comprises a horizontal platform which can be oscillated about vertical and horizontal axes. When a gyroscope is mounted on the platform the inner and outer gimbal bearings are exercised simultaneously so that stiction in the bearings is eliminated during a drift test. Because an oscillation having an amplitude of 5 degrees and a period of 4 seconds is applied to the gyroscopes about their outer gimbal axes in the rocket (see section 4) a similar oscillation is applied to each gyroscope during laboratory tests. At the same time an oscillation having an amplitude of 3.5 degrees and a period of 4 seconds is applied to the gyroscope about the inner gimbal axis. The drift of the spin-axis of the gyroscope is recorded during the test by reflecting a beam of light from a mirror fixed to the inner gimbal of the instrument (see section 2).

The second type of test equipment comprises a rotating wedge[2] on which the gyroscope is mounted (the wedge angle is normally set between 0 and 5 degrees to the horizontal plane). The wedge is rotated about a vertical axis in a clockwise or anti-clockwise direction at speeds of up to 6 degrees per second. This equipment shows the effect of unidirectional rotation of the gyroscope about the inner and outer gimbal axes, on the drift characteristic of the instrument. With the wedge angle set at zero degrees the equipment becomes a horizontal turn-table. The drift of the gyroscope is recorded by using the optical equipment referred to above, but since the frame and cover of the gyro rotate about the outer gimbal axis it is necessary to replace the light alloy cover with a perspex cover.

(b) At W.R.E. (Australia)

Similar test equipments, constructed at R.A.E., are used for exercising and recording the drift characteristics of the gyroscopes at Salisbury and Woomera W.R.E. Each equipment comprises a horizontal platform which forms part of a 45 degree wedge as shown diagrammatically in Fig.18. A shaft inclined at 45 degrees to the horizontal supports the wedge and is driven through gears by an electric motor so that it oscillates about its longitudinal axis. This imparts ~~an~~ oscillation to the platform about vertical and horizontal axes contained within a vertical plane through the shaft (Fig.18). The equipment was developed so that a complete interferometer-bay could be mounted on the platform after the gyroscopes have been fitted, so that the drift characteristics of the instruments can be measured under 1 g conditions after installation. Any change in the drift characteristics can therefore be determined until the bay is fitted to the rocket. If a single gyroscope is under test, drift is measured by using optical equipment as shown in Fig.18. This is not possible if a complete interferometer-bay is mounted on the test equipment and it is then necessary to use the voltage outputs from the ac pick-offs on the gyroscopes.

With the interferometer-bay in the position it occupies in the launcher (mirror of gyroscope in position 2 facing bearing 313 degrees as in Fig.3) an oscillation of ±7 degrees having a period of 4 seconds is applied to the driving shaft so that oscillations of approximately ±5 degrees are applied about the outer gimbal axes of the gyroscopes and ±3.5 degrees about their inner gimbal axes. (This condition was chosen because the programmed oscillation of ±5 degrees superimposed on the roll-stabilised head of the rocket causes a similar oscillation about the outer gimbal axes of the gyroscopes during flight.) Several gyro-drift tests, each of five minutes duration, are carried out in this manner. Additional tests are also conducted using the same oscillation period (4 seconds) but with the motion of the driving shaft reduced to an amplitude of 2 degrees. This applies oscillations of approximately ±1.4 degrees about the outer gimbal axes of the gyroscopes and ±1 degree about their inner gimbal axes when the bay is in the position described above. This modified test applies oscillations about the inner gimbal axes of the gyroscopes which approximate more closely to the flight condition.

(c) Typical test results

Gyroscopes used during the programme were tested on each of the test equipments described above. Some of the results obtained from the instruments fitted in Skylark SL 28 are given in Table 3 as examples. The data show drift rates recorded when using the test equipment described in section 9 (b) above. (An earth rate correction has been applied to the results.)

Table 3

Gyro number	Gimbal axis	Drift rate of spin axis (degrees/hour)				
		Mean value		Standard deviation σ		Direction
		±7° drive	±2° drive	±7° drive	±2° drive	
R.S. 7 (Position 1)	Inner	8.5	5.1	0.8	1.2	Mirror up
	Outer	5.4	7.7	1.7	0.8	Mirror left
R.S. 15 (Position 2)	Inner	7.2	10.2	1.1	1.1	Mirror down
	Outer	11.2	8.0	0.5	0.9	Mirror right

Table 4 shows results obtained from the same instruments when tested on the rotating wedge (see section 9 (a)) with the wedge angle set at zero degrees so that the equipment became a horizontal turn-table. Each gyroscope was mounted on the test equipment in turn with the spin-axis pointing "mirror north" and the outer gimbal axis in line with the axis of rotation of the table. The turn-table was then driven in an anti-clockwise direction when viewed from above the gyroscope and the following drift rates about the inner gimbal axis were recorded.

Table 4

Gyro number	Rotation rate of turn-table (degrees/second)	Drift rate (degrees/hour)
R.S. 7	3	59.5 (mirror down)
	6	51.2 (mirror down)
R.S. 15	3	93.5 (mirror down)
	6	77.3 (mirror down)

10 FLIGHT RESULTS

The analysis of telemetry records and the calculation of missile attitude and gyro-drift from flight data was carried out at the Applied Electronics Laboratories, G.E.C. Stanmore.

The attitude of each missile was measured in the launcher[4] with respect to the reference axes shown in Fig.19. An example of missile attitude at launch is given by the following data from Skylark SL 32

Pitch angle γ = $+88^\circ\ 20.4'$
Yaw angle β = $-5^\circ\ 25.4'$
Roll angle α = $+171^\circ\ 22.9'$ (Position of longitudinal datum line on the rocket.)

The acceleration of each rocket during powered flight was obtained from optical tracking data. A typical record which shows acceleration as a function of time (from Skylark SL 31) is given in Fig.20.

Attitude of the head of each rocket was computed at intervals of 0.2 second (approximately) from the interferometer data and from the voltage outputs of the ac pick-offs on the gyroscopes (Ref.6 Appendix H). A refraction correction was applied in addition to the skin temperature correction when attitude was derived from interferometer data. Attitude in pitch, yaw and roll obtained from the gyroscopes and interferometers fitted in Skylark SL 32 is given in Figs.21 to 24 as an example. Because ambiguities occur in the outputs from the interferometer receivers at intervals of approximately 6 degrees, mean pitch and yaw angles calculated from trajectory data are also included in Figs.21 and 22 to show that the ambiguities have been correctly resolved.

Gyro-drift was calculated by using a computer program to compare attitude obtained from the two gyroscopes with attitude obtained from interferometer data which was used as a reference. Drift was computed at times of zero outer gimbal angle on the yaw gyroscope (position 1, Fig.3) as defined by the photo-electric pick-off i.e. about every 2 seconds (see Note (v)). The voltage output of the ac pick-off was used to provide data from the outer gimbal of the pitch gyroscope (position 2, Fig.3) but an "in-flight" calibration obtained from the photo-electric pick-off was used to interpret these results; giving an estimated accuracy of within ±3 arc minutes. Information from the inner gimbals of the gyroscopes was obtained from the ac pick-offs.

Typical total drifts from launch of the spin-axes of the gyroscopes tested are shown in Figs.25 and 26 (from Skylark SL 27) and Figs.27 and 28 (from Skylark SL 32). The mean gyro drift rates during the acceleration period of Skylark SL 27 were obtained from Figs.25 and 26 and are summarised in Table 5. (An earth rate correction has been applied to the results.)

Note (v)

Although it is stated in section 3.1 that a pulse is obtained from the photo-electric pick-off on the outer gimbal of each gyroscope every 0.5 second, the time available for analysis purposes was only sufficient to allow gyro-drift to be determined each time a datum pulse was generated by the photo-electric pick-off on the yaw gyroscope (position 1, Fig.3).

Table 5

Gyro number	Gimbal axis	Mean drift rate of spin-axis (degrees/hour)
R.S. 9 (Position 1)	Inner (yaw)	85.6 (mirror down)
	Outer (roll)	31.0 (mirror right)
R.S. 16 (Position 2)	Inner (pitch)	35.6 (mirror down)
	Outer (roll)	45.5 (mirror left)

Faulty roll-stabilisation caused the head of the missile to rotate with the rocket motor in certain firings. This gave higher drift rates about the inner gimbal axes of the gyroscopes, as indicated by the results from Skylark SL 28 (Table 6), where the head rotated in an anti-clockwise direction (viewed from the nose) at approximately 1 revolution per second. An earth rate correction has been applied to the results shown in Table 6.

Table 6

Gyro number	Gimbal axis	Mean drift rate of spin-axis (degrees/hour)
R.S. 7 (Position 1)	Inner (yaw)	442.0 (mirror down)
	Outer (roll)	44.0 (mirror left)
R.S. 15 (Position 2)	Inner (pitch)	620.0 (mirror down)
	Outer (roll)	33.5 (mirror left)

In the case of rockets fitted with an aerial-survey camera, a total of approximately 110 frames were exposed by the camera after launch at intervals of about 0.5 second. The last frame was usually taken from an altitude of approximately 180 000 ft at about +75 seconds. A photograph of ground markers taken from an altitude of 7000 ft (approximately) by the camera fitted in Skylark SL 32 is given in Fig.29. The optical calibration of the camera and a correction for camera attitude in the rocket were included when attitude was derived from the film record as described in Ref.6 (Appendix J). Pitch, yaw and roll attitudes of the head of Skylark SL 32, obtained from the camera record, are shown in Figs.21 to 24.

11 DISCUSSION OF RESULTS

Because data is only obtained at intervals of 0.5 second from the camera record, the photographic method of attitude measurement was used primarily as a check on the results from the interferometer system which provides a data rate limited only by the telemetry sampling switch. Certain differences occurred between missile attitude obtained from the interferometer system and missile attitude obtained from the photographic method. Pitch attitudes of Skylark SL 31 from interferometer and camera data show a difference of about 0.6 degree although yaw and roll attitudes obtained from the two systems are in close agreement. In the case of Skylark SL 32, yaw attitudes obtained from the two systems show a difference of 0.3 degree but there is close agreement in pitch and roll. These differences could be due to normal incidence errors associated with the interferometer receivers, errors in determining camera attitudes relative to missile axes or a combination of both.

Comparisons between missile attitude obtained early in flight from the interferometers and gyroscopes suggested an error in the normal incidence calibration of the interferometers. Since incremental drift from some arbitrary datum in flight is used when drift rates are determined, this error is not important provided it remains constant. Long term stability checks on the normal incidence calibration of the interferometers, carried out at G.E.C. Stanmore, have shown changes of less than 1 arc minute in the outputs from the receivers over periods considerably longer than the complete flight time of the Skylark rocket, and so the effect of this error on the determination of gyro-drift rates is considered to be negligible.

A continuous record of drift rate and a close correlation with acceleration could not be obtained from the gyro-drift data because it was not possible to obtain an accurate estimate of drift rate during periods which were short compared with the complete experiment (about 30 seconds). This was due to the scatter associated with the data and because a measurement of gyro-drift was obtained only every 2 seconds. The mean drift rate recorded and the mean acceleration of each rocket was therefore used when "in-flight" drift rates were compared with results recorded during pre-flight tests.

Faulty operation of the roll control units fitted to two rockets (Skylarks SL 28 and SL 30) caused the instrumented heads to rotate during flight. The very high drift rates of the gyroscopes about their inner gimbal axes (e.g. see Table 6) is thought to be due to the combined effects of missile acceleration and unidirectional friction torques in the outer gimbal bearings. Gyro-drift results recorded under these conditions were compared with ground test results obtained from the gyroscopes by using the turn-table equipment described in section 9 (a). (Wedge angle zero degrees and rate of rotation approximately 6 degrees per second in the appropriate direction.)

12 CONCLUSIONS

The results obtained from twelve gyroscopes tested during the flight programme (Skylarks SL 27 to SL 32)[12,13,14] show that drift rates about their inner gimbal axes could be predicted from ground tests. Results from eight gyroscopes show that the mean value of the outer gimbal friction torque increased to between 1.13 and 1.64 times the value that would be reached if the friction torque remained directly proportional to acceleration. These values compare favourably with factors of from 1.25 to 1.31 derived from a typical bearing characteristic and the mean acceleration of each vehicle. Flight records of short duration (15 seconds and 12.5 seconds) obtained in two other cases (gyroscopes R.S. 14 and R.S. 25 respectively) each gave a factor of 2.1 but these are not considered to be as representative as those quoted above. Only in the remaining two experiments was it found impossible to correlate ground test results with drift rates which occurred in flight and this was because the "in-flight" drift took place in the opposite direction to that expected from ground test results. When drift rates about the inner gimbal axes of the gyroscopes obtained from ground test results were compared with flight results, it was noted that the $\pm 5^{\circ}$ oscillation about the outer gimbal axis of each gyro gave a closer correlation with the "in-flight" drift rate than the smaller amplitude of oscillation ($\pm 1.4^{\circ}$). This was expected because the effect of outer gimbal friction torques in flight is more accurately simulated, in the 1 g condition, by an oscillation which has the same amplitude (and frequency) as the programme superimposed on the roll-stabilised head ($\pm 5^{\circ}$ about the roll axis).

Eight of the twelve gyroscopes tested drifted about their outer gimbal axes during flight in the direction predicted. In five of these eight experiments the drift rate was within ±31% of the drift rate predicted from ground test results and in one of the remaining three cases no results were available from the test equipment at W.R.E. which simulates the flight condition. Four of the twelve gyroscopes tested drifted about their outer gimbal axes in the opposite direction to that expected, so no correlation was possible. A drift characteristic which remains stable during transportation is a prerequisite in obtaining correlation between ground test results and "in-flight" drift rate and towards the end of the programme there was some unavoidable deterioration in this respect. The effects of mass unbalance and inner gimbal friction torques on drift rates about the outer gimbal axes could not be separated because the difference between each effect acting alone did not exceed about 12% of the total drift rate[12]. Drift rates appeared to be controlled mainly by friction torques however because correlation with ground tests depended on selecting results obtained by using the more representative "in-flight" oscillation about the inner gimbal axes of the gyroscopes ($\pm 1^{\circ}$).

Anisoelastic effects which cause drift rates proportional to the square of the acceleration did not appear to affect the gyroscopes tested during this programme because there was no obvious increase in drift rates during periods of peak acceleration. This is possibly because the direction of the applied acceleration was practically parallel to one of the principal axes of the gyroscopes (outer gimbal axis).

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Appendix A

POSITIONS OF THE SKYLARK LAUNCHER, TRANSMITTERS 1 AND 2 AND GROUND MARKERS IN RANGE CO-ORDINATES

The position of an object is defined with reference to the range origin in terms of the following co-ordinates:-

Y = Distance measured down-range (bearing 304° 42.7').

X = Distance measured across-range (positive direction, bearing 34° 42.7').

Z = Height above the range origin.

(a) Positions of the Skylark launcher and Transmitters 1 and 2.

Site	Y (ft)	X (ft)	Z (ft)
Skylark launcher	7	-173	-8
Transmitter 1 (Coondambo site)	141 904	-151 599	49
Transmitter 2 (P.5 site)	136 144	111 408	-122

(b) Positions of ground markers for the Photographic Method of measuring missile attitude.

Site No.	Y (yd)	X (yd)	Z (ft)
1	1660	-58	-14.4
2	2550	-58	-17.7
3	3670	-58	-24.9
4	5330	-58	-30.4
4A	5330	+110	
5	7670	-58	+15.8
6	11000	-58	-55.3
7	16000	-58	-80.6
7A	16300	-58	-75.3
8	23330	-58	-147.6
9	33660	-58	-274.5
10	50000	-58	-544.7

Appendix B

CALIBRATION OF THE OPTICAL SYSTEM OF THE CAMERA

Calibration of the camera is carried out by the manufacturer. The position of the lens is set at its focal length from the plane of the film adjacent to the register plate and the filter is fitted. The camera is then mounted on a goniometer with the optical axis horizontal so that rotation of the camera about a vertical axis can be measured on a horizontal scale. (The camera can also be rotated about the optical axis.) The reseau marks on the register plate are viewed through the optical system of the camera by means of a fixed collimator fitted with cross-wires.

To check the radial distortion of the sight-line from a particular reseau mark the camera is set up with the radius between the reseau mark and the intersection of the two principal axes, U_o and V_o (Fig.13) horizontal. The angle through which the camera must be turned to align first the selected reseau mark and then the optical axis of the camera with the cross-wires in the collimator is then measured and compared with the value calculated from a knowledge of the focal length of the lens and the known position of the reseau mark on the register plate. Any departure from the calculated value is a measure of radial distortion. A measurement of distortion in a direction perpendicular to radial distortion (tangential distortion) is also obtained from the goniometer at each reseau mark and the two measurements are resolved to give distortion parallel to the two principal axes of the register plate (U_o and V_o, Fig.13). These values are plotted so that the effect of distortion between the reseau marks can be obtained by interpolation.

The reseau marks provide a nominal 1 cm grid of reference points. Errors in the position of each reseau mark are known from measurements taken off the master grid from which the register plate was engraved. The maximum error in the position of each reseau mark is 8 microns. Since the error in the nominal position of each reseau mark is known the actual position of the reseau mark is used when optical distortion is calculated from measurements obtained from the goniometer.

Appendix C

SUMMARY OF THE SOURCES OF ERROR ASSOCIATED WITH THE INTERFEROMETER SYSTEM

Source of error	Remarks	Estimated errors (arc minutes)
Variations in transmitter frequency	A change of 25 Mc/s is required to give an error of 1 arc minute. Transmitter frequency is cavity controlled with temperature compensation[9].	Negligible
Multiple reflections from fins	Checked[11]. No significant effect.	Negligible
Error in knowledge of sight-lines from transmitter to missile at any instant	Missile position determined by kinetheodolites and multi-station Doppler. Accurate data only required to an altitude of 75000 ft.	±1
Refraction of sight-line at X-band frequencies	Refraction correction applied is based on standard atmosphere. (Unpublished work by G.E.C. Stanmore.)	±3
Mechanical distortion of the interferometer base	Expansion of the interferometer base length in flight due to a change in temperature will affect the calibration. The aerials are mounted rigidly and the temperature of each interferometer base is telemetered so that a correction can be applied[6,9].	Negligible
Phase modulation of R.F. from transmitters by motor flame	Should not occur since sight-lines are almost normal to the missile axis.	Negligible
Variations in the amplitude of the signals from the reference generator (Fig.12)	The voltage outputs from the phase sensitive rectifiers will change if the amplitude of the reference signals vary. This effect is checked during flight by frequent calibrations (see section 3.2.2 and Ref.9).	±1
Random variations in receiver performance	Noise etc. Not attributed to any particular part of the receiver[9].	±2

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6	T.C. Lea	Estimation of missile attitude from gyroscopic, interferometric and photographic data. G.E.C. Stanmore Technical Note 7622/TCL/165C, July 1961
7	B. Secker	Estimation of missile attitude by photographic methods. G.E.C. Stanmore Technical Note GW(TS)/BS/KPR/21 (1957)
8	Sqn. Ldr. D. Biltcliffe Flt. Lt. A. Thirkettle Flt. Lt. A. Kedar	A roll control system for Skylark. R.A.E. Technical Note Space 12 (1962)
9	K.P. Robinson	Notes on the design of a radio interferometer for a missile attitude indicator. G.E.C. Stanmore Technical Note GW(H)/KPR/8, January 1957

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No.	Author	Title, etc.
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11	B.D. Campbell R. Wallace	Skylark missile attitude interferometer receiver - the effect of the fins on the signal received by a simple aerial of the interferometer receiver. G.E.C. Stanmore Technical Note GW/BDC/RW/468, August 1957
12	F.R. Knott	The drift of an auto-pilot gyroscope due to prolonged acceleration in the Skylark rocket. Part 2: Results from Skylarks SL 27/28. R.A.E. Technical Report 65162 (A.R.C. 27489) (1965)
13	F.R. Knott	The drift of an auto-pilot gyroscope due to prolonged acceleration in the Skylark rocket. Part 3: Results from Skylarks SL 29/30. R.A.E. Technical Report 66132 (A.R.C. 28612) (1966)
14	F.R. Knott	The drift of an auto-pilot gyroscope due to prolonged acceleration in the Skylark rocket. Part 4: Results from Skylarks SL 31/32 and general conclusions from programme. R.A.E. Technical Report 66133 (A.R.C. 28613) (1966)

Fig.1. The G.W.5 Mk.3 two-axis gyroscope

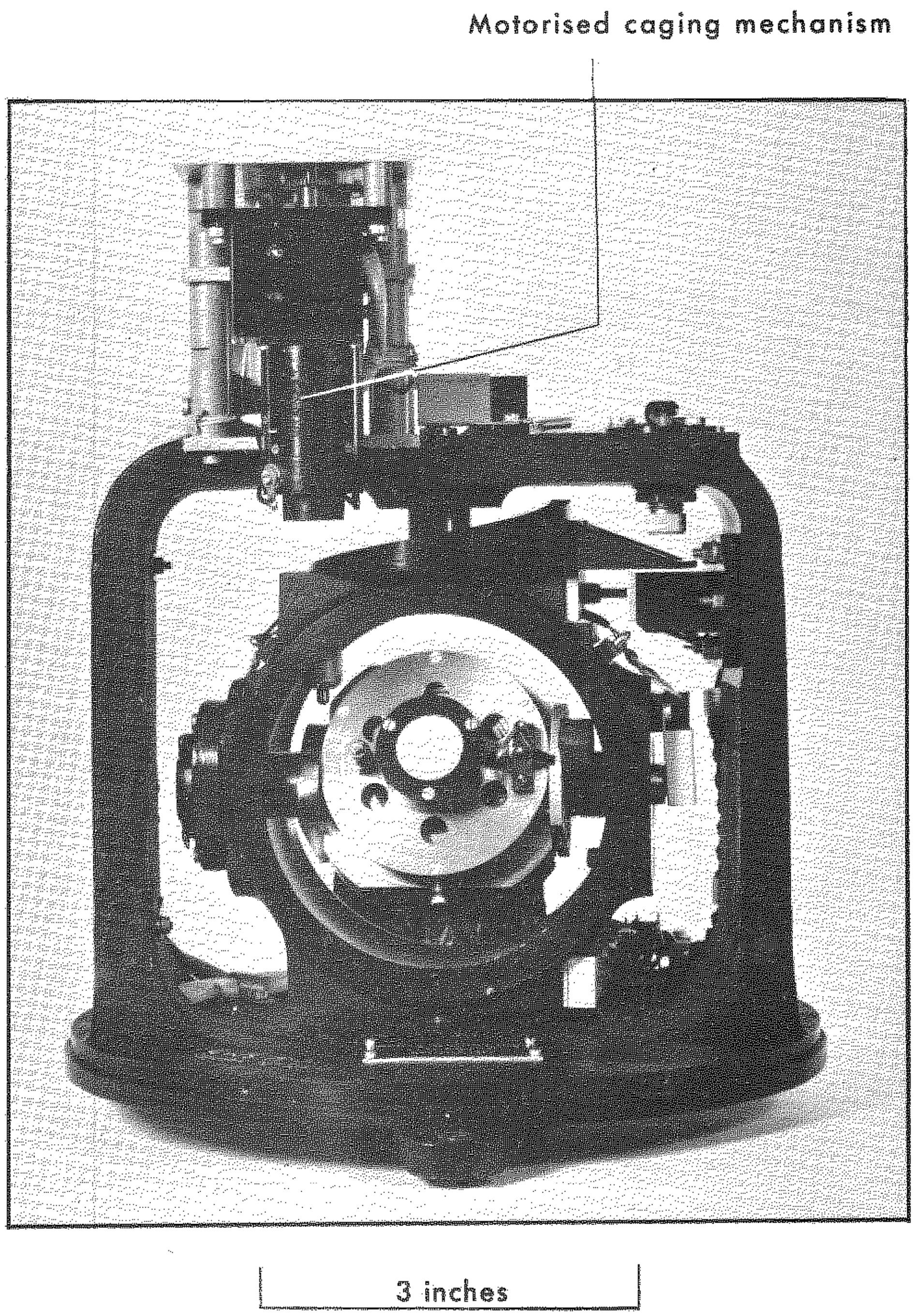
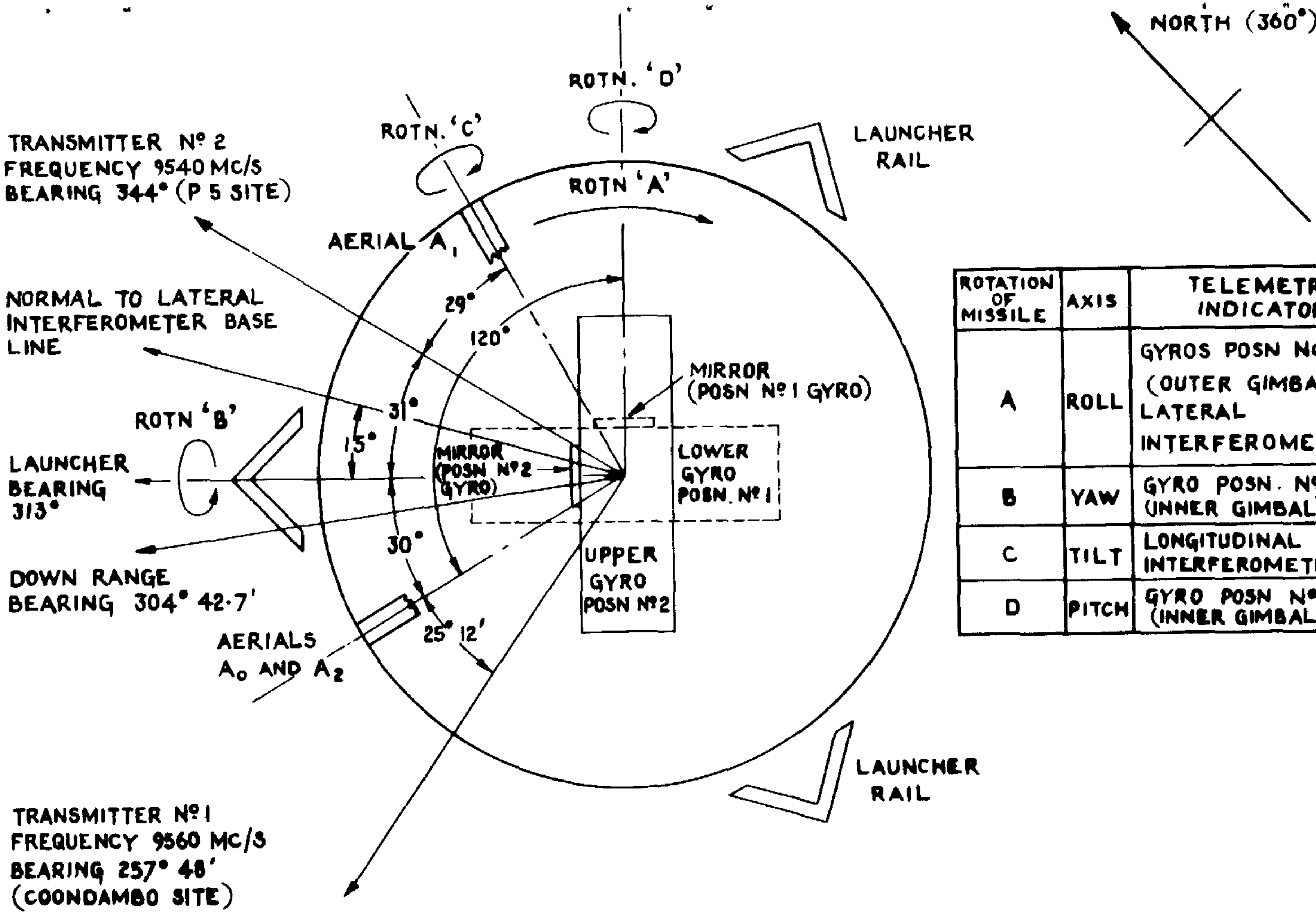


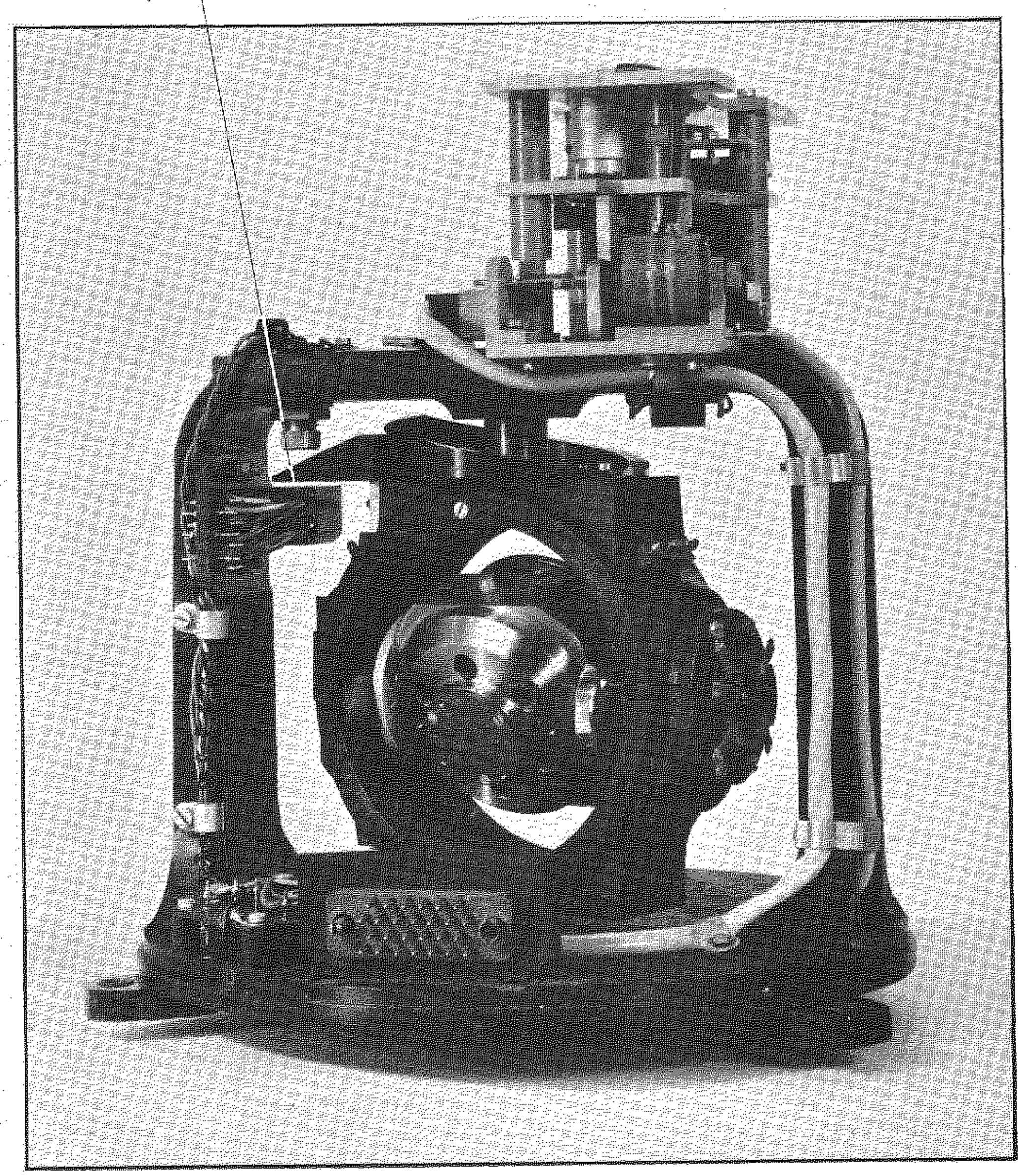
Fig.2. The GW.5 Mk.3 gyroscope (cover removed showing gimbals, mirror, caging mechanism etc.)



ROTATION OF MISSILE	AXIS	TELEMETRY INDICATOR
A	ROLL	GYROS POSN Nos 1 AND 2 (OUTER GIMBALS) LATERAL INTERFEROMETER
B	YAW	GYRO POSN. Nº 1 (INNER GIMBAL)
C	TILT	LONGITUDINAL INTERFEROMETERS
D	PITCH	GYRO POSN Nº 2 (INNER GIMBAL)

FIG. 3 SECTION THROUGH INTERFEROMETER—BAY WITH THE MISSILE IN THE LAUNCHER (VIEW LOOKING TOWARDS FIN ASSEMBLY)

Vane of photo-electric pick-off



3 inches

Fig.4. The G.W.5 Mk.3 gyroscope (cover removed showing photo-electric pick-off on outer gimbal)

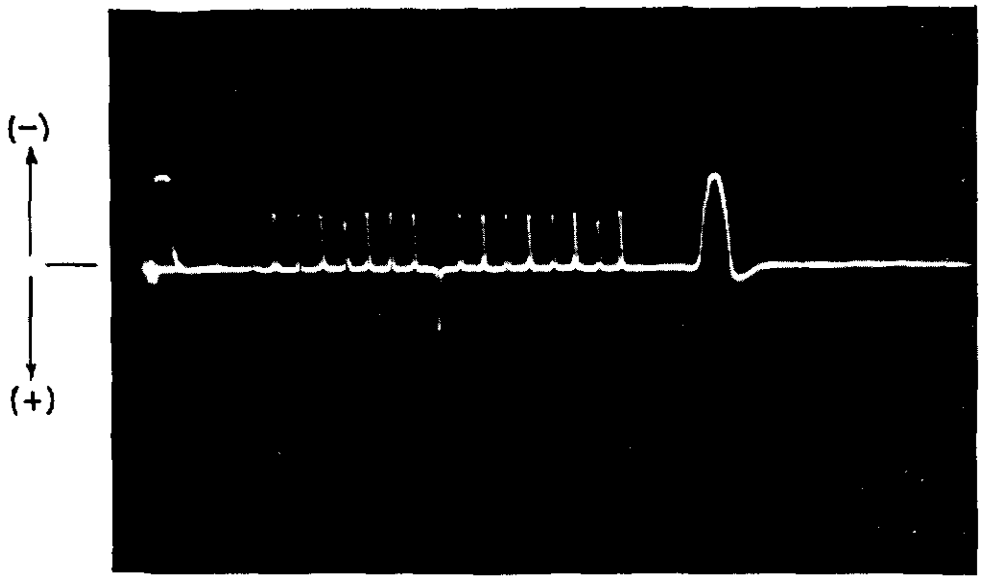


Fig.5. Voltage output from photo-electric pick-off

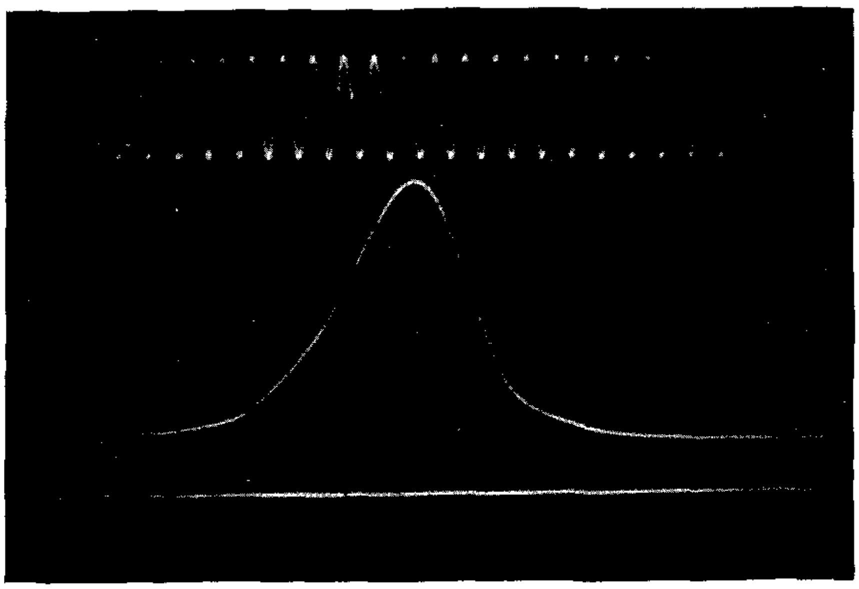


Fig.6. Voltage output from photo-electric pick-off showing *shape of individual pulse and a reference waveform* of 2 minutes of arc per cycle

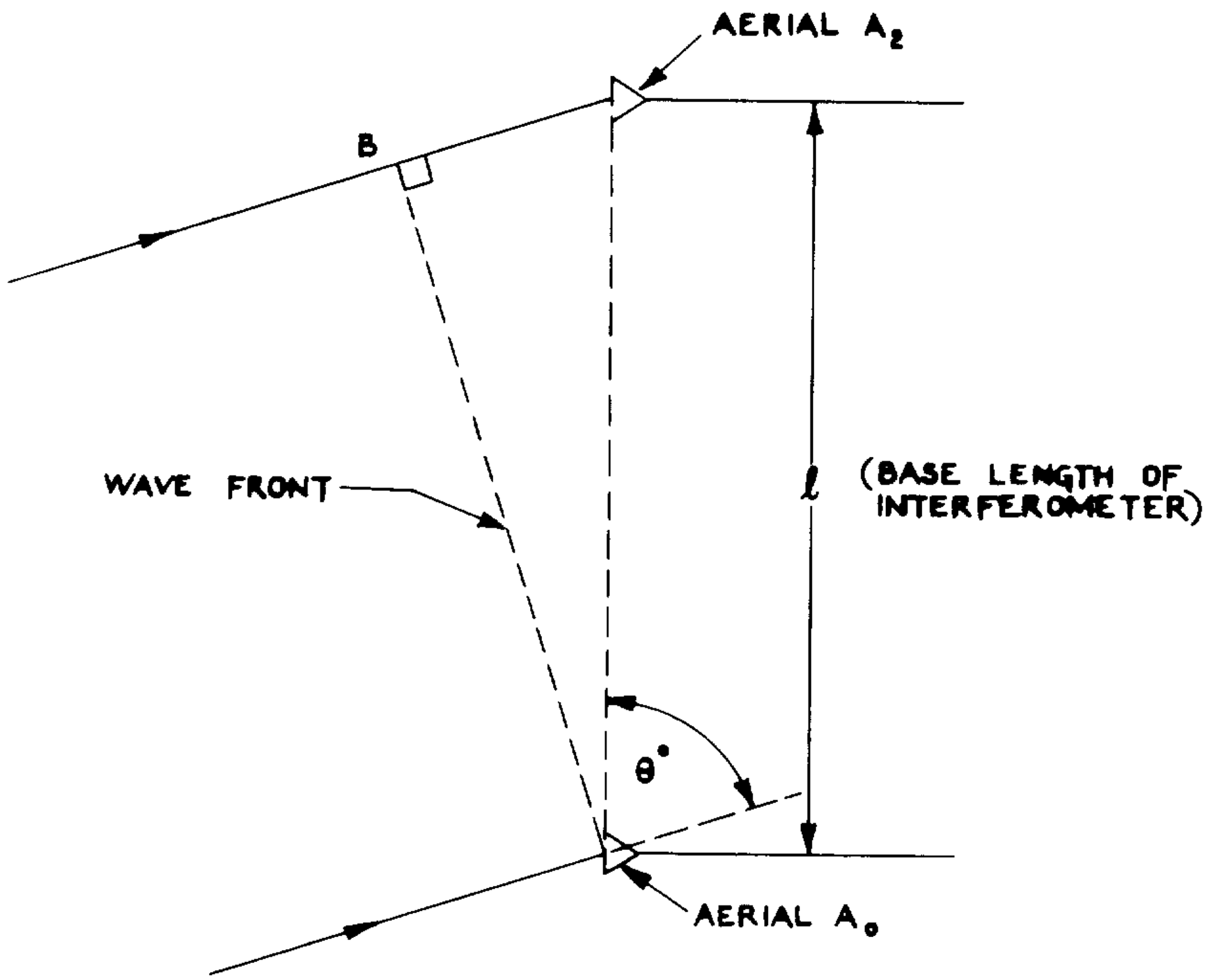


FIG. 7 THE BASIC INTERFEROMETER

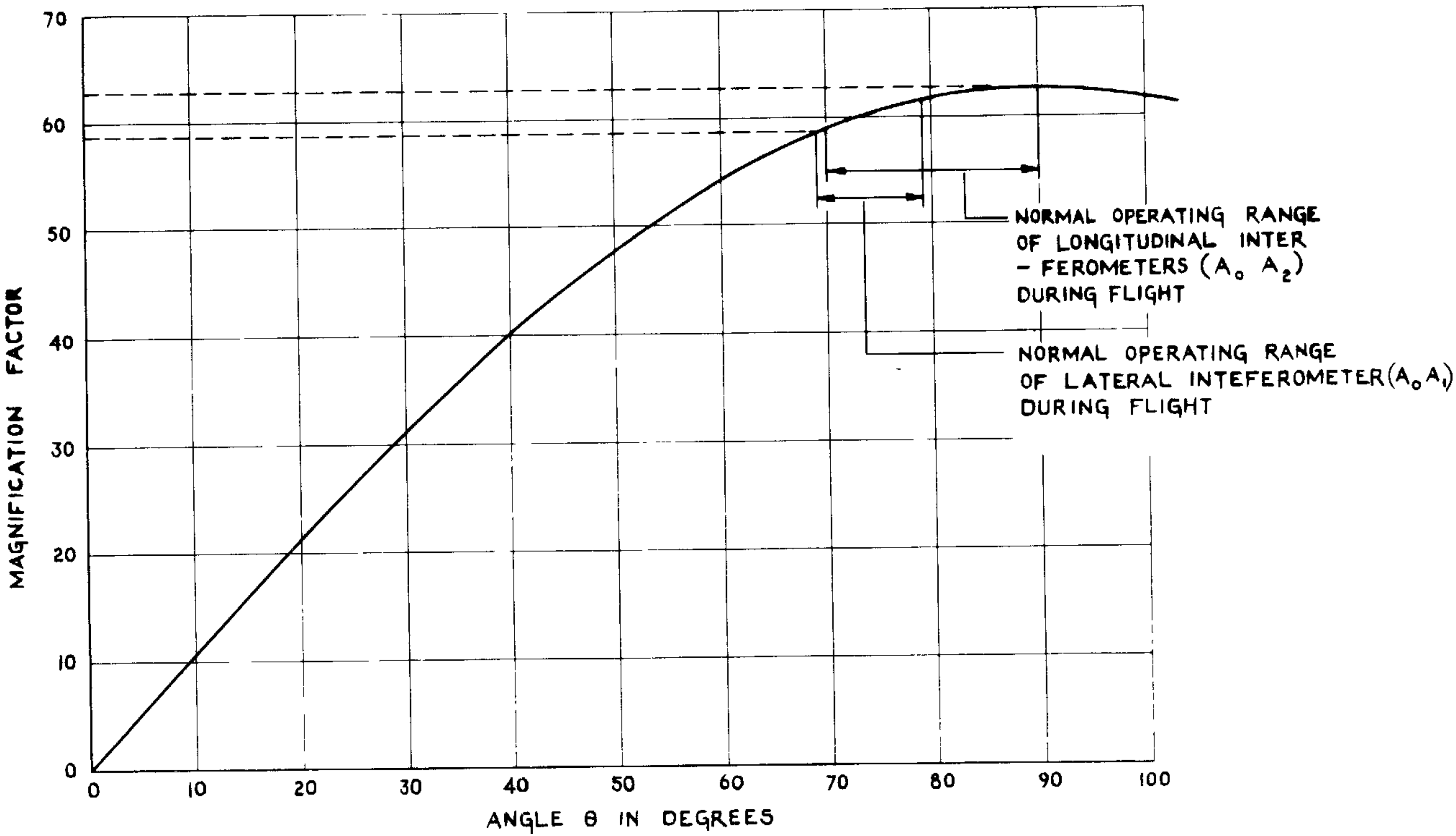


FIG. 8 VARIATION IN MAGNIFICATION FACTOR OF AN INTERFEROMETER WHEN $\ell = 10\lambda$

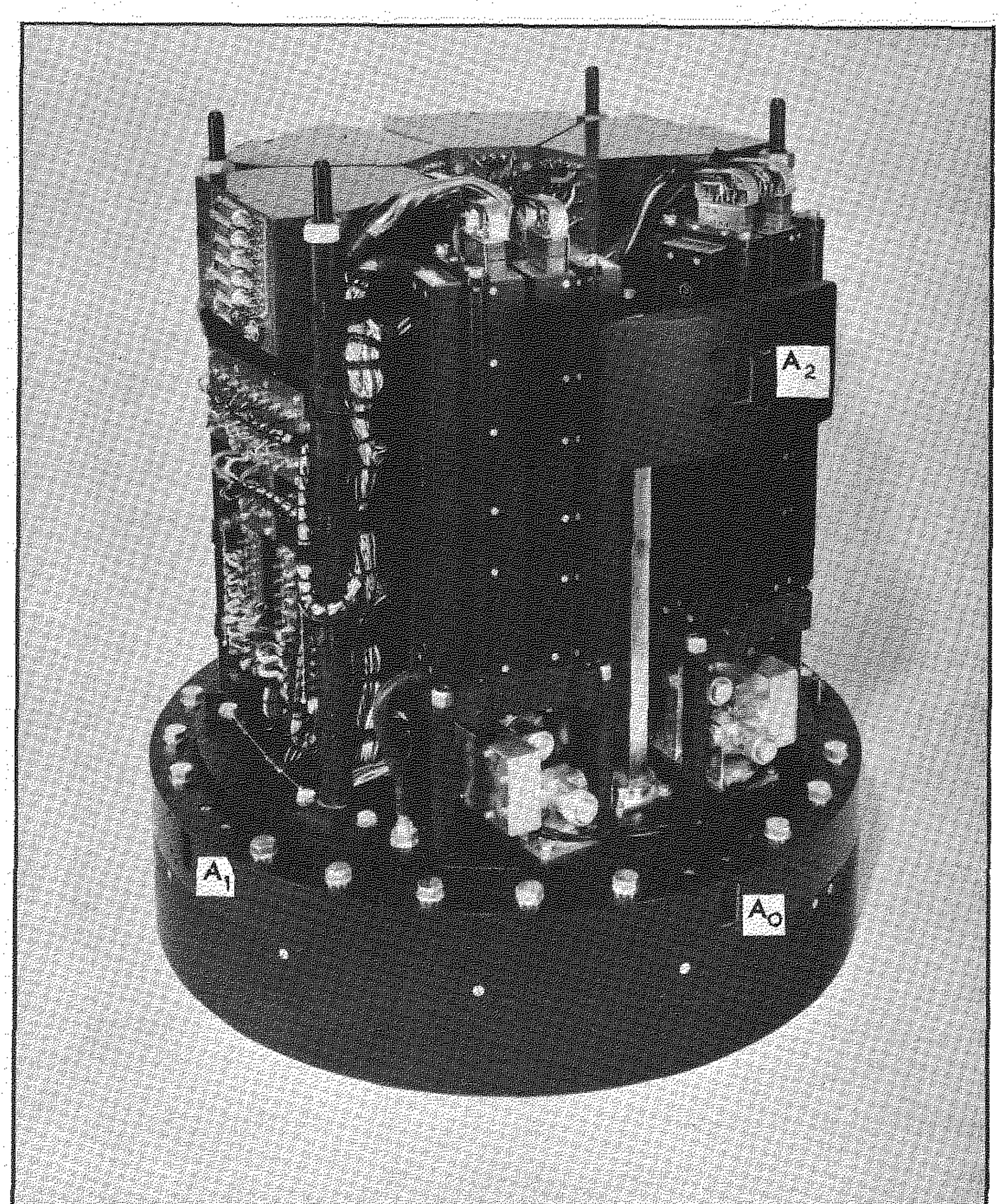


Fig.9. The interferometer receivers and aerials (A_0, A_1 and A_2)

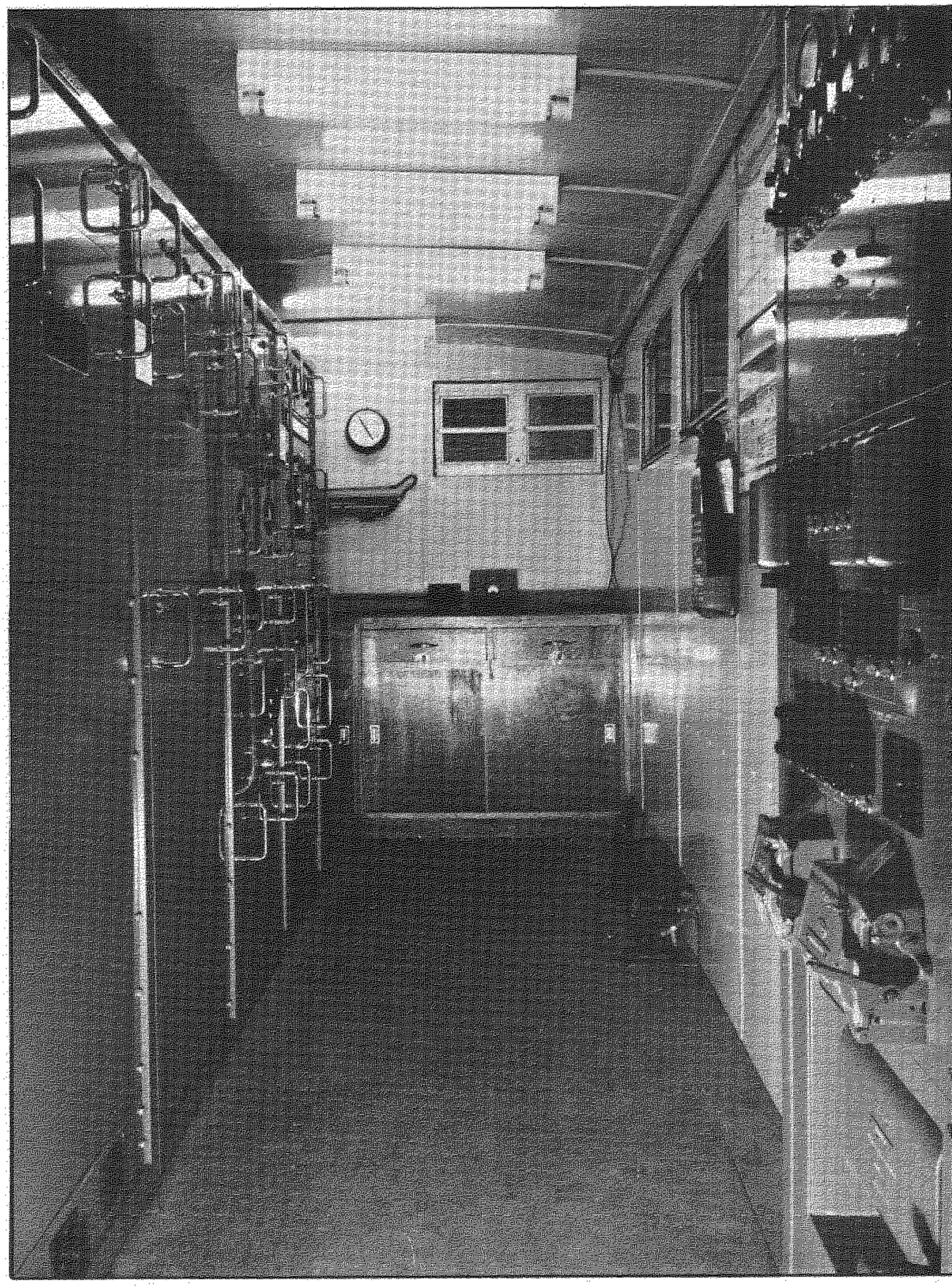


Fig.10. The interior of one of the interferometer transmitter vans

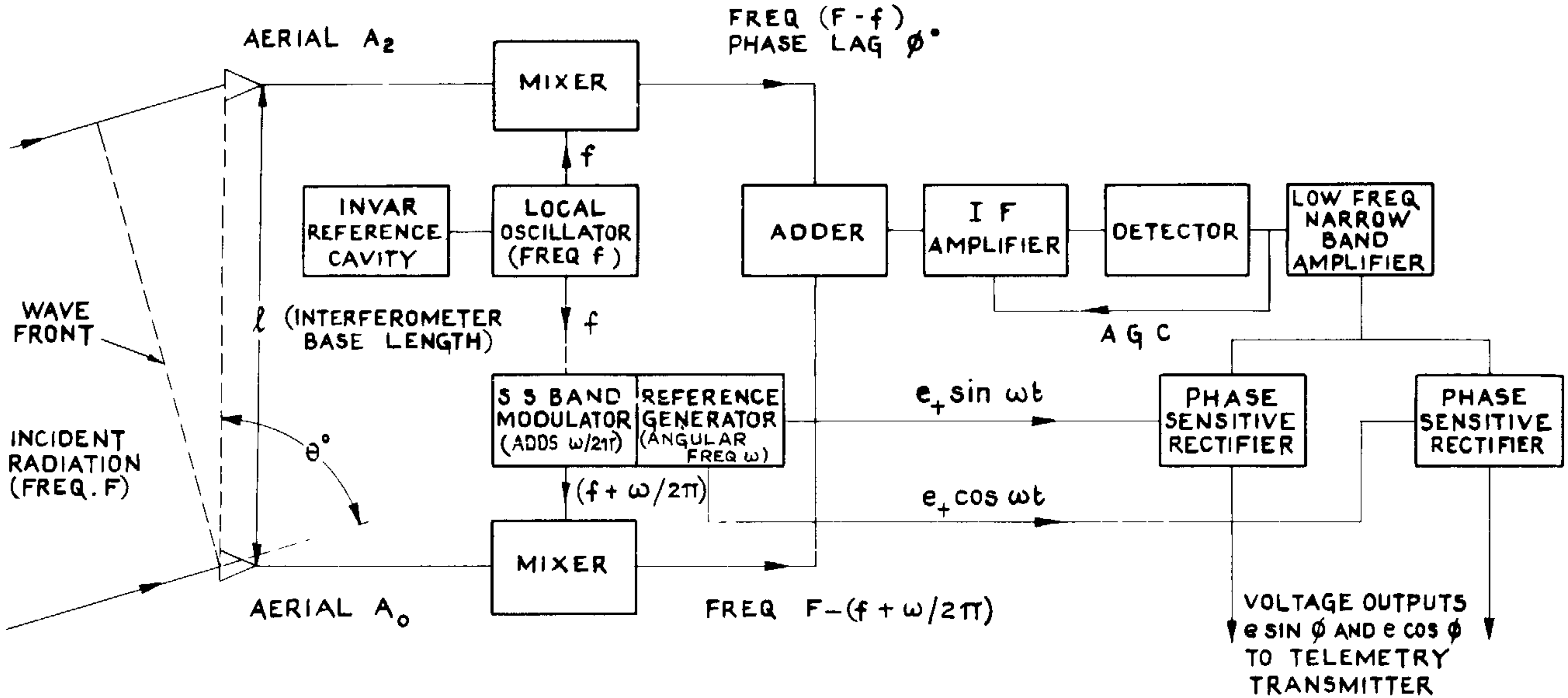


FIG 11 BLOCK DIAGRAM OF A SINGLE CHANNEL INTERFEROMETER RECEIVER

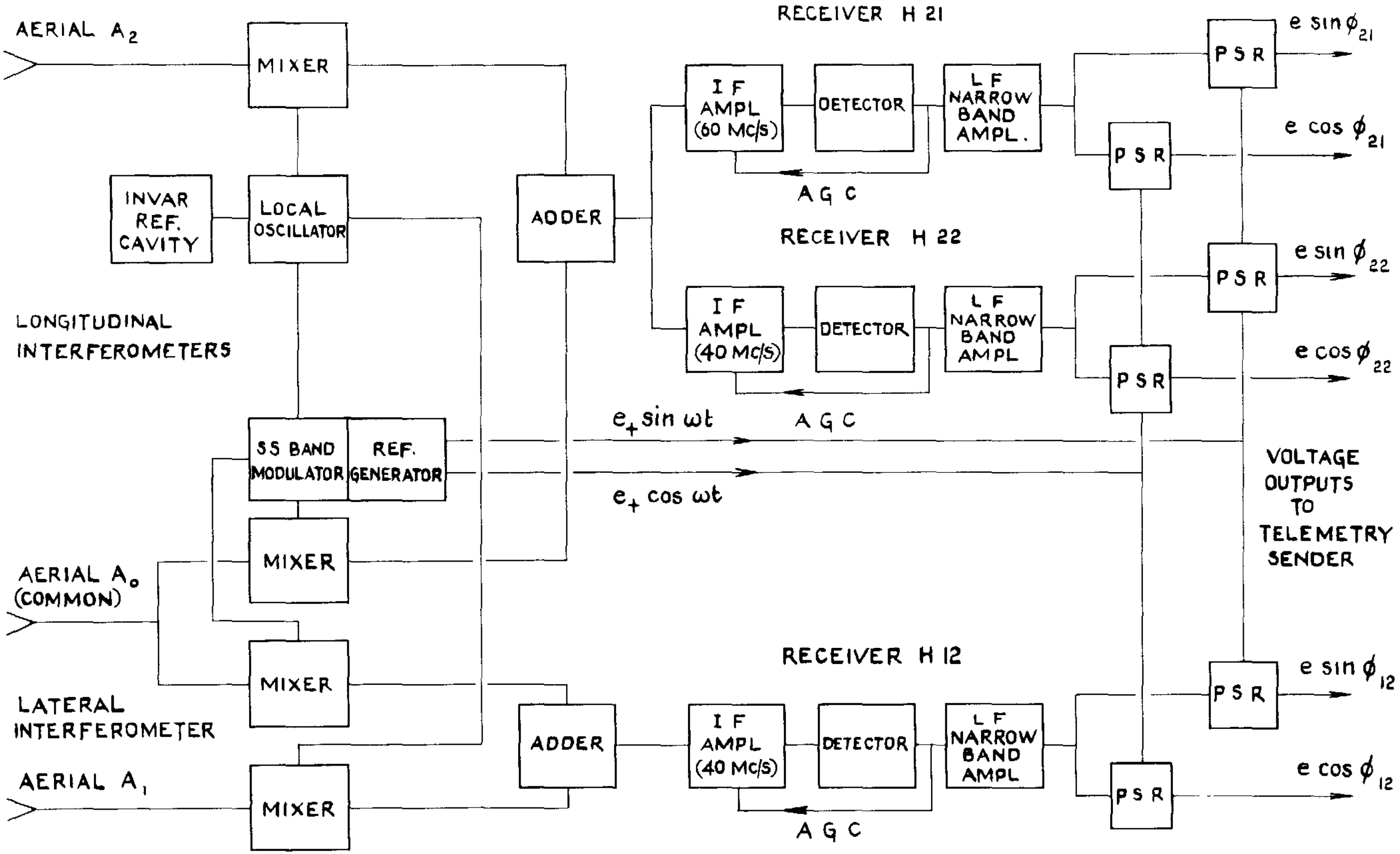


FIG. 12 BLOCK DIAGRAM OF THE THREE CHANNEL INTERFEROMETER RECEIVER

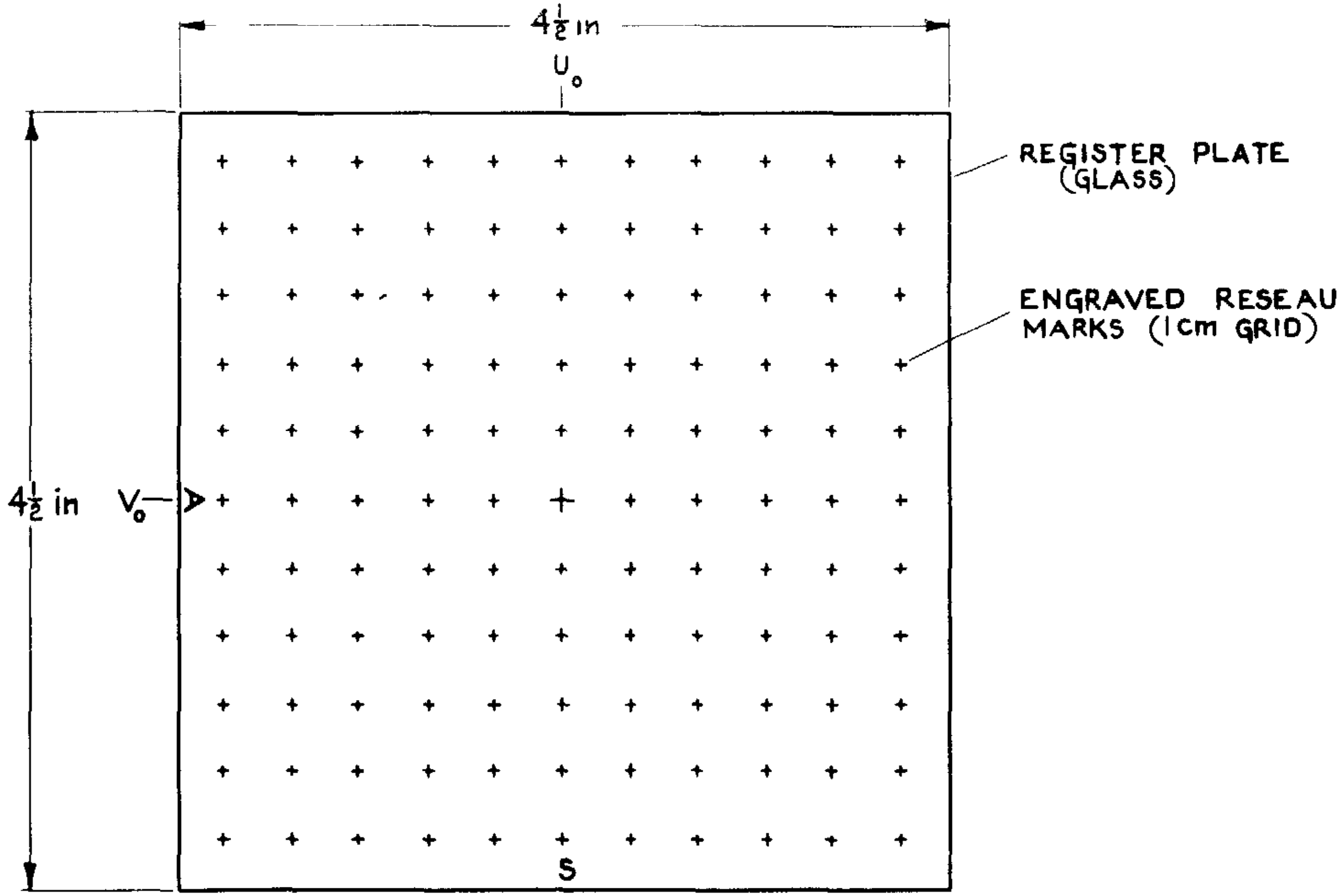


FIG 13 REGISTER PLATE OF CAMERA

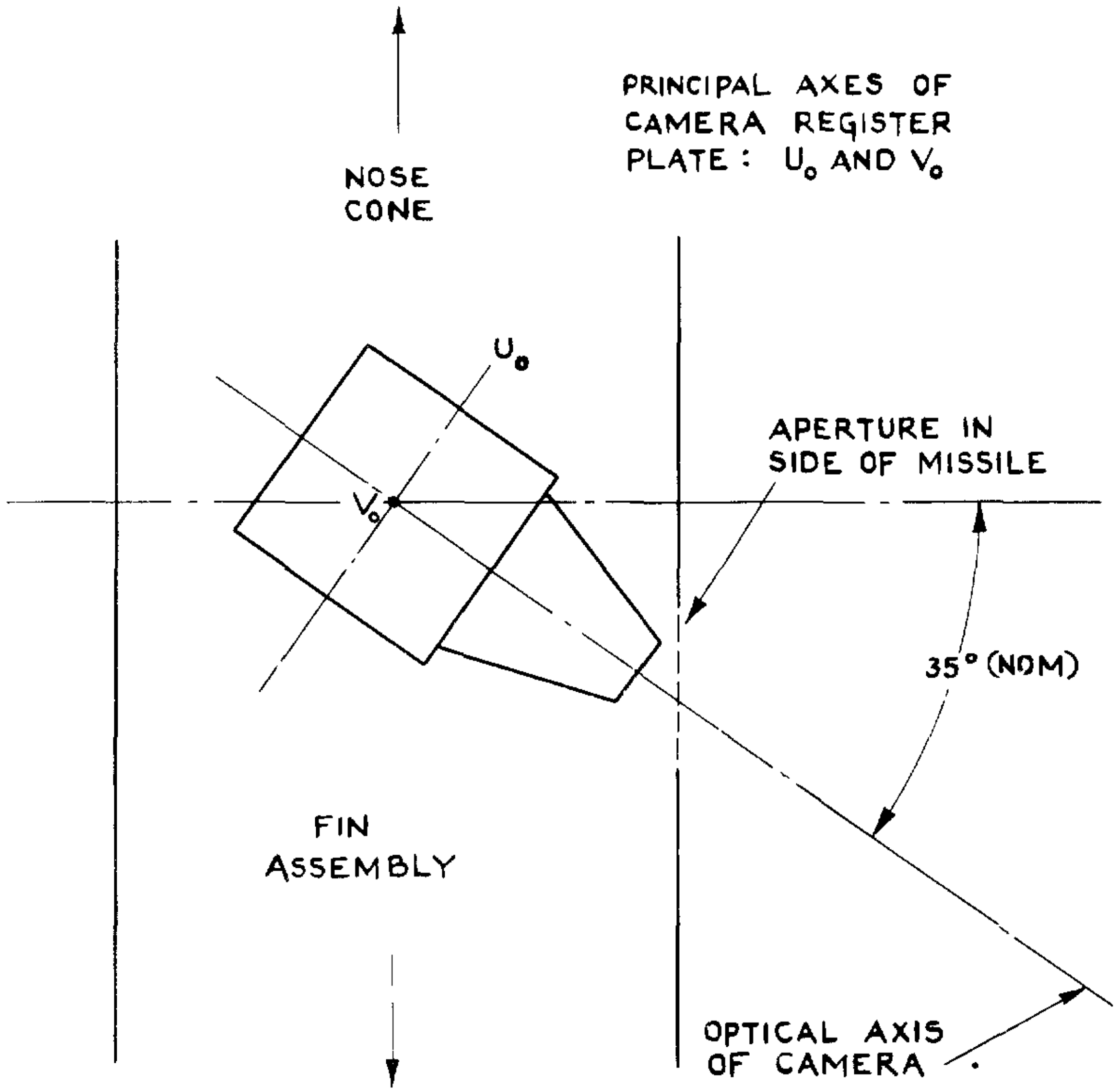
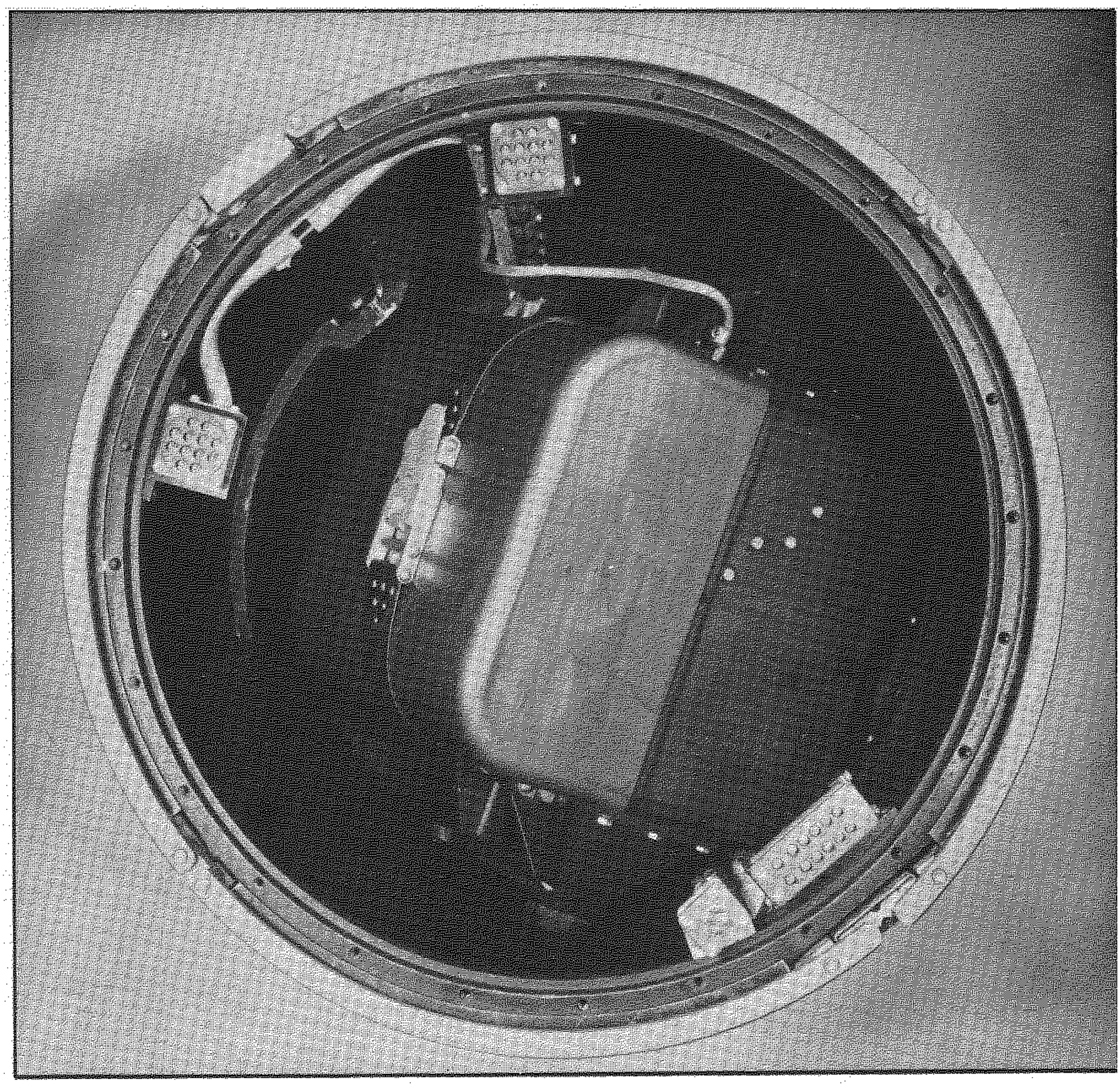


FIG 14 ATTITUDE OF CAMERA IN MISSILE



6 inches

Fig.15. Camera fitted in missile

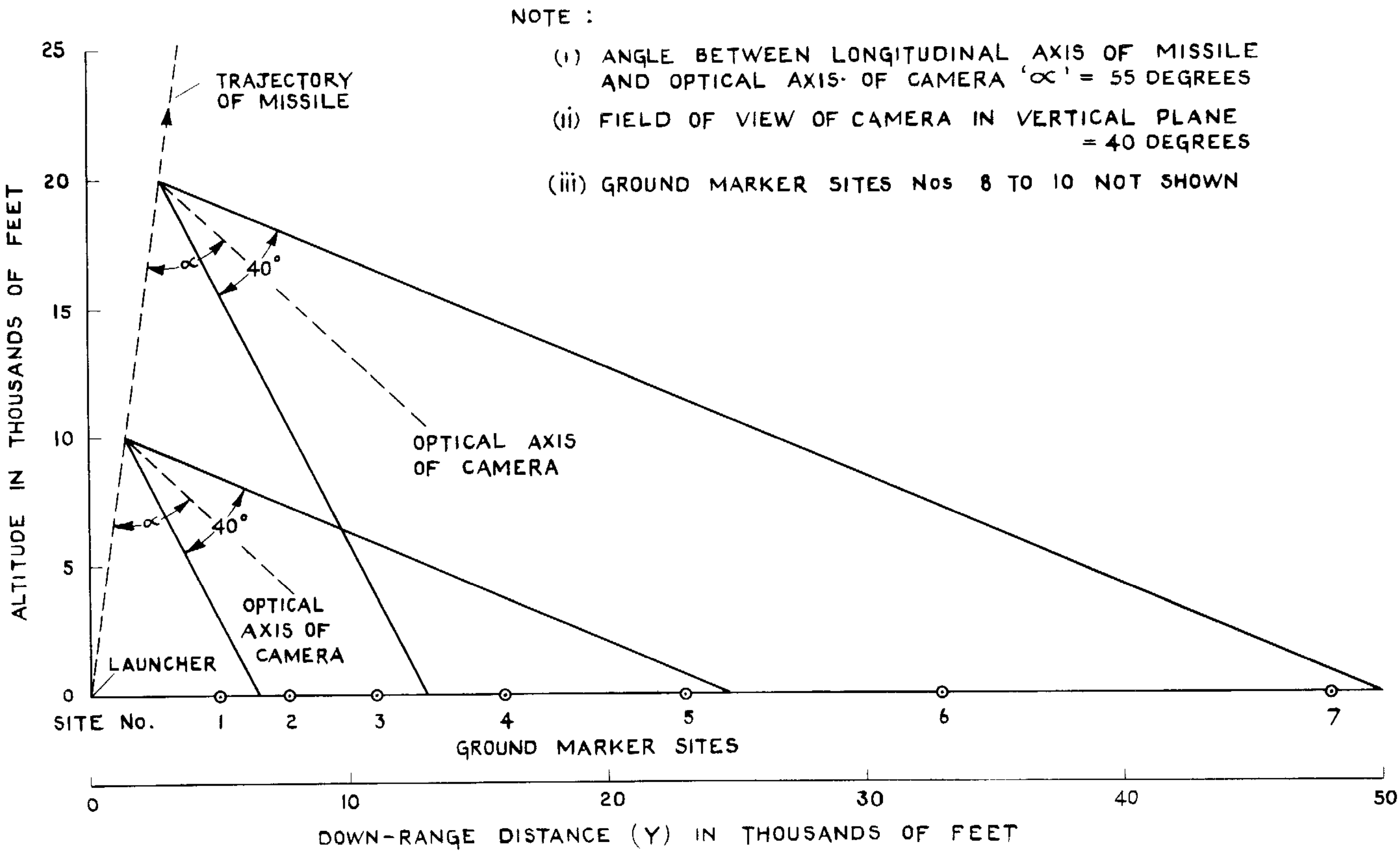


FIG 16 METHOD OF PHOTOGRAPHING GROUND MARKERS FROM THE MISSILE

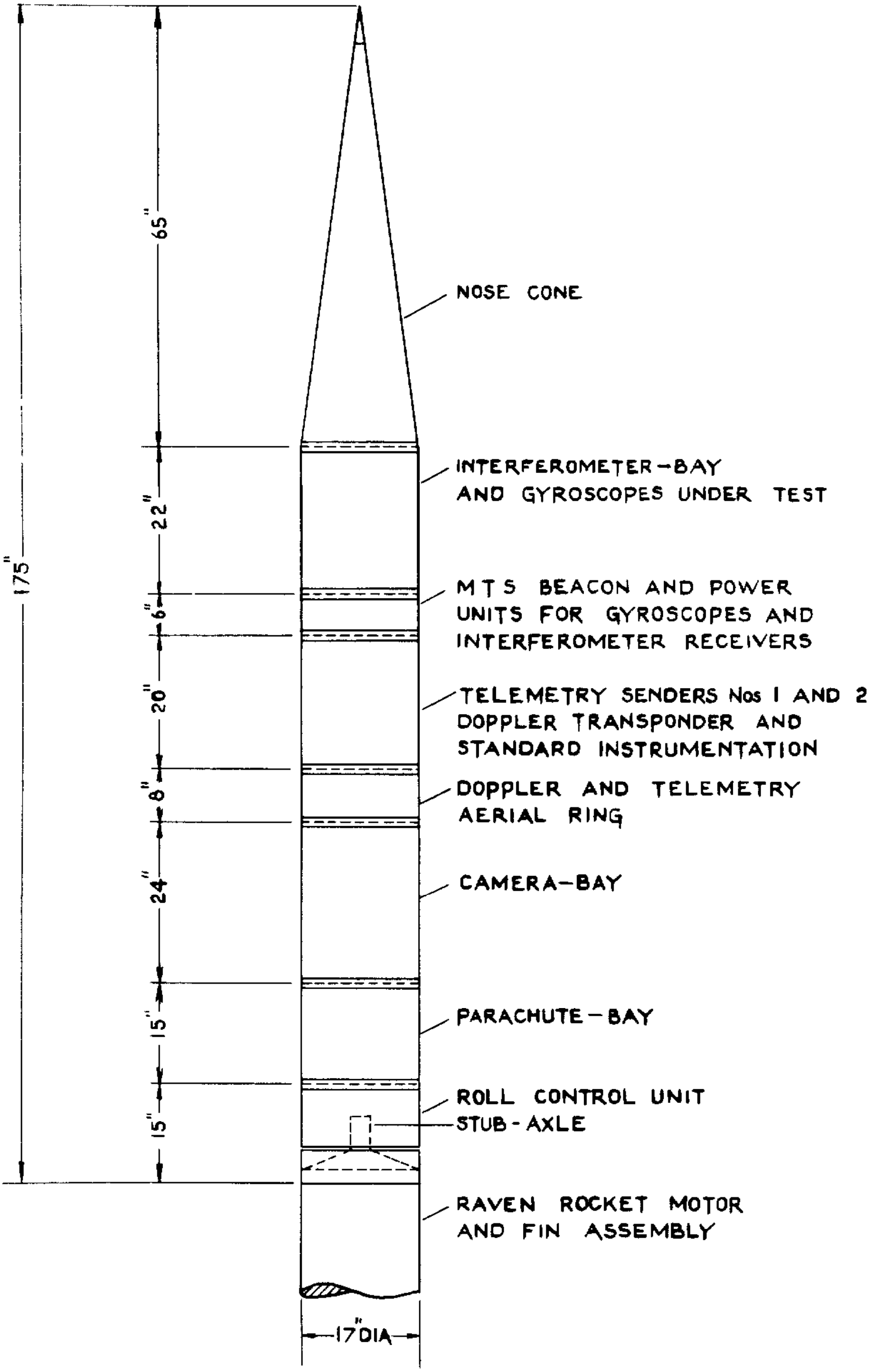


FIG. 17 COMPLETE HEAD ASSEMBLY OF THE GYRO-DRIFT TEST VEHICLE

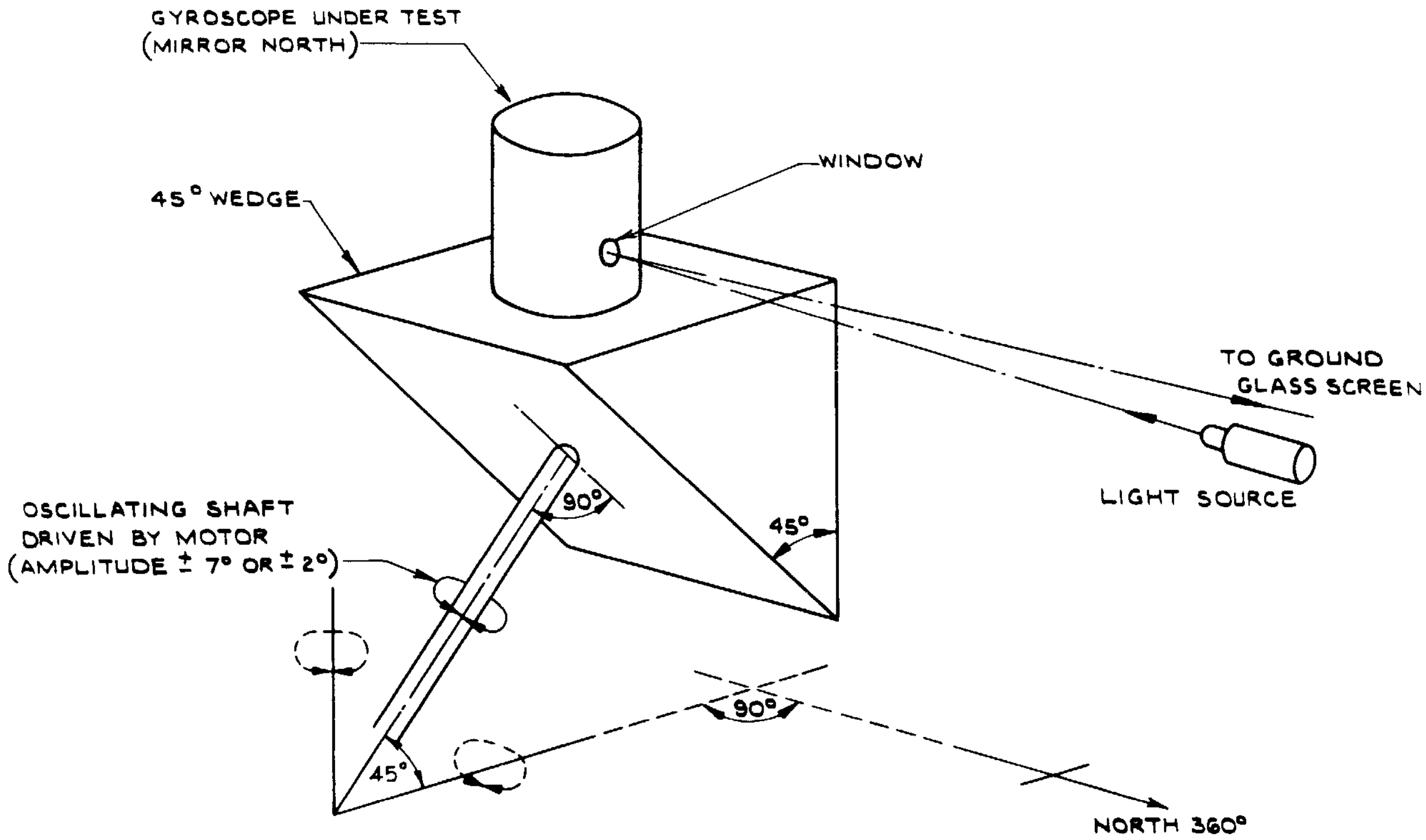
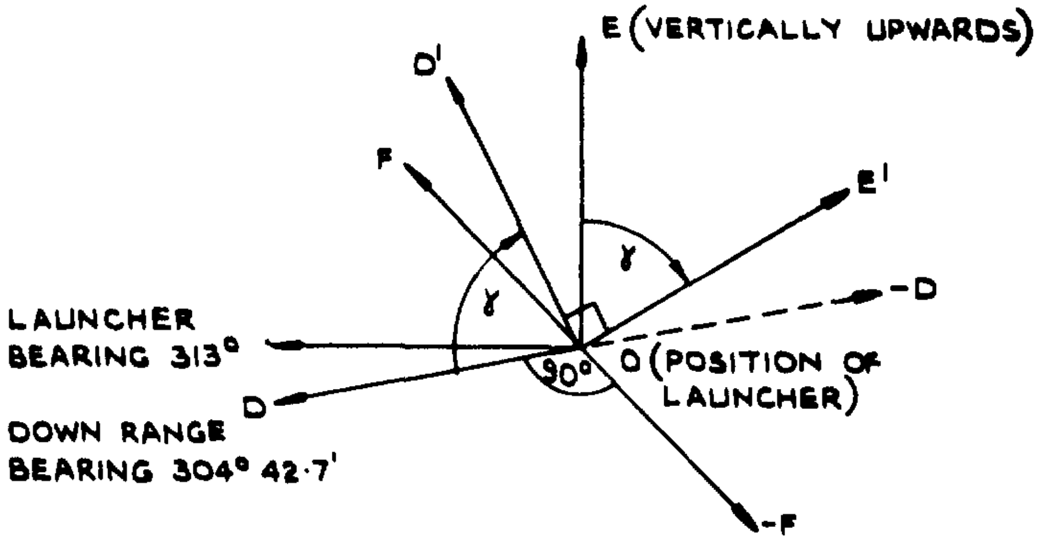


FIG 18 SCHEMATIC DIAGRAM OF GYRO DRIFT TEST EQUIPMENT FOR A SINGLE GYROSCOPE OR A COMPLETE INTERFEROMETER-BAY

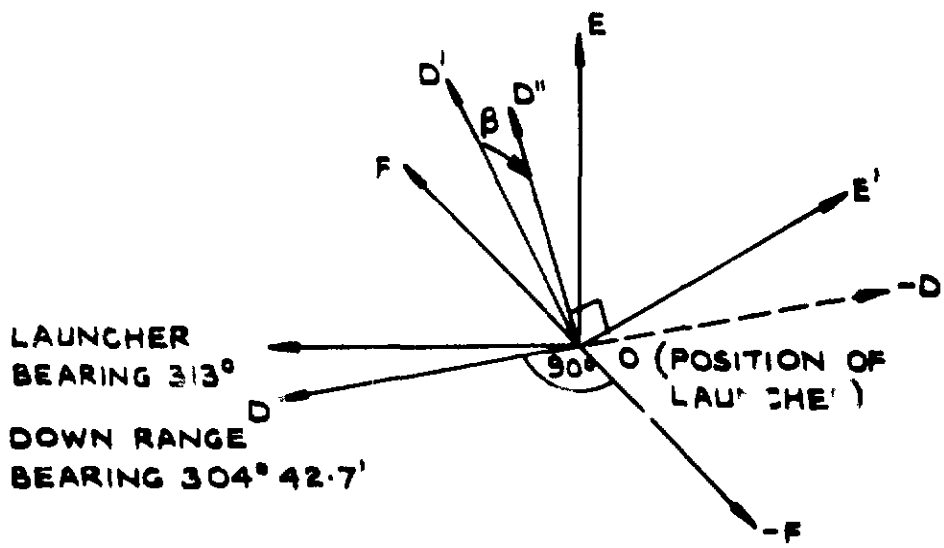
(i) PITCH ANGLE γ

ROTATE OD ABOUT AXIS OF TO POSITION OD'



(ii) YAW ANGLE β

ROTATE OD' ABOUT AXIS OE' TO POSITION OD" (I.E TOWARDS -OF)



(iii) ROLL ANGLE α

ROTATE MISSILE ABOUT OD" (LONGITUDINAL AXIS)

THE ROLL DATUM IS GIVEN BY THE DIRECTION OF OE'

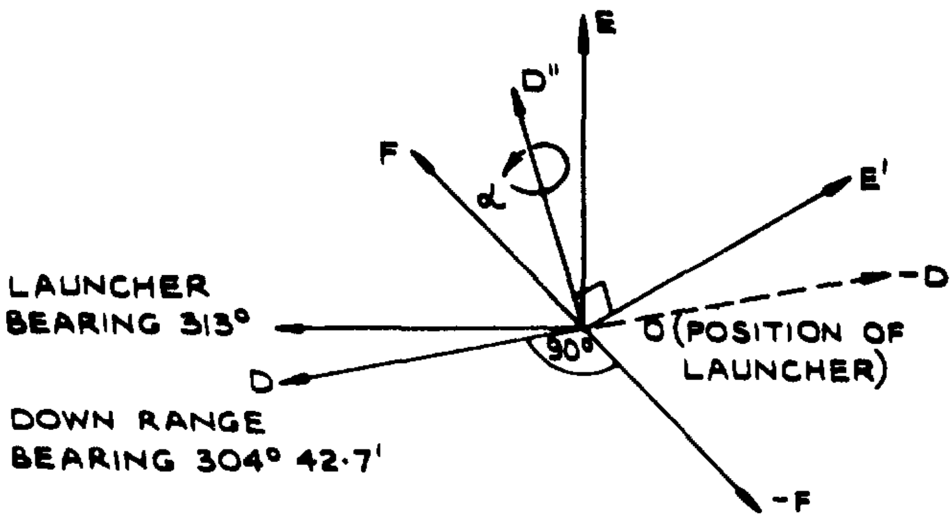


FIG. 19 PITCH, YAW AND ROLL ANGLES

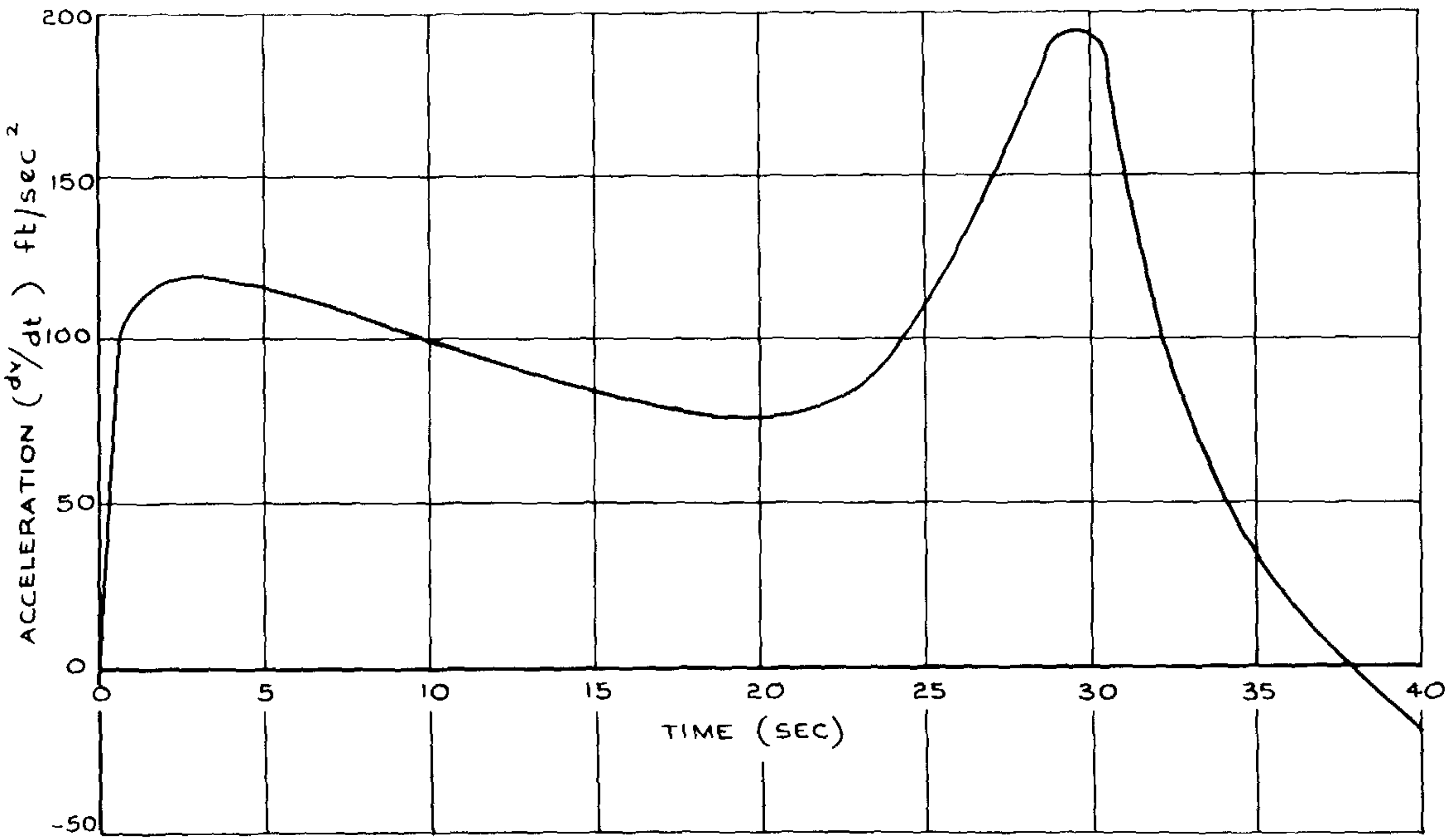


FIG.20 ACCELERATION OF SKYLARK SL 31

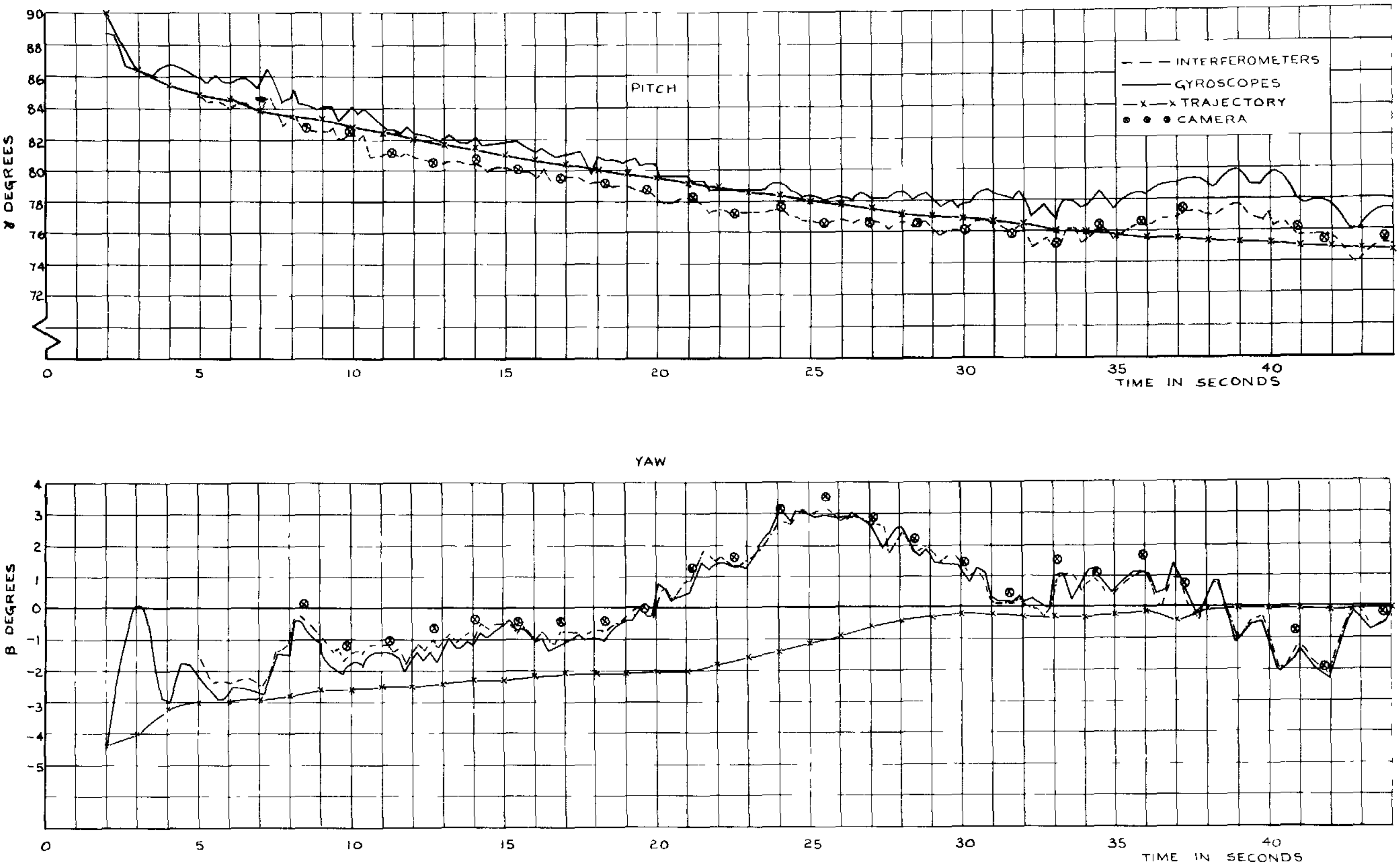


FIG 21 SL 32 PITCH AND YAW ATTITUDE OF MISSILE (O TO 44 SEC)

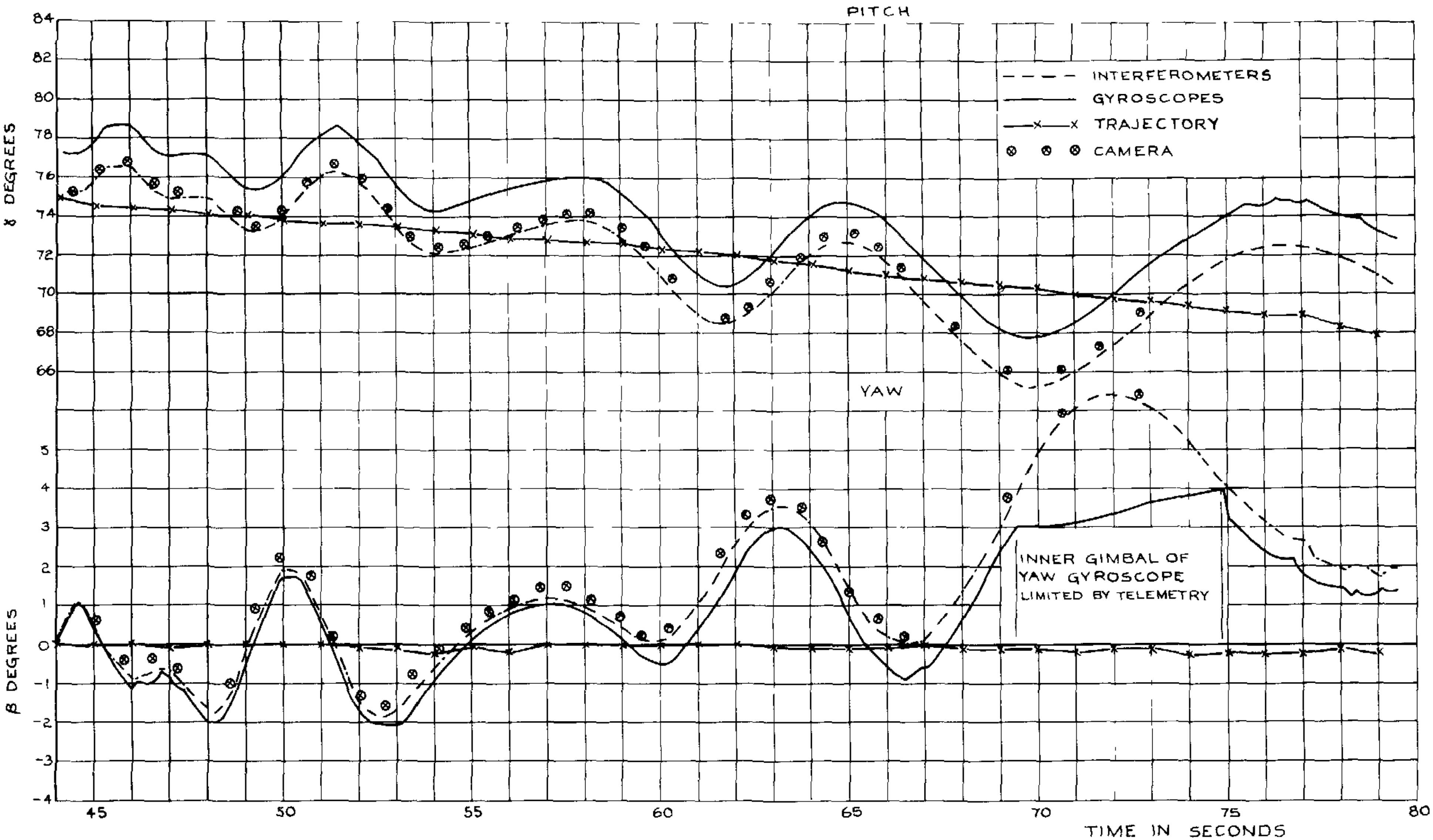


FIG.22 SL 32 PITCH AND YAW ATTITUDE OF MISSILE (44 TO 80 SEC)

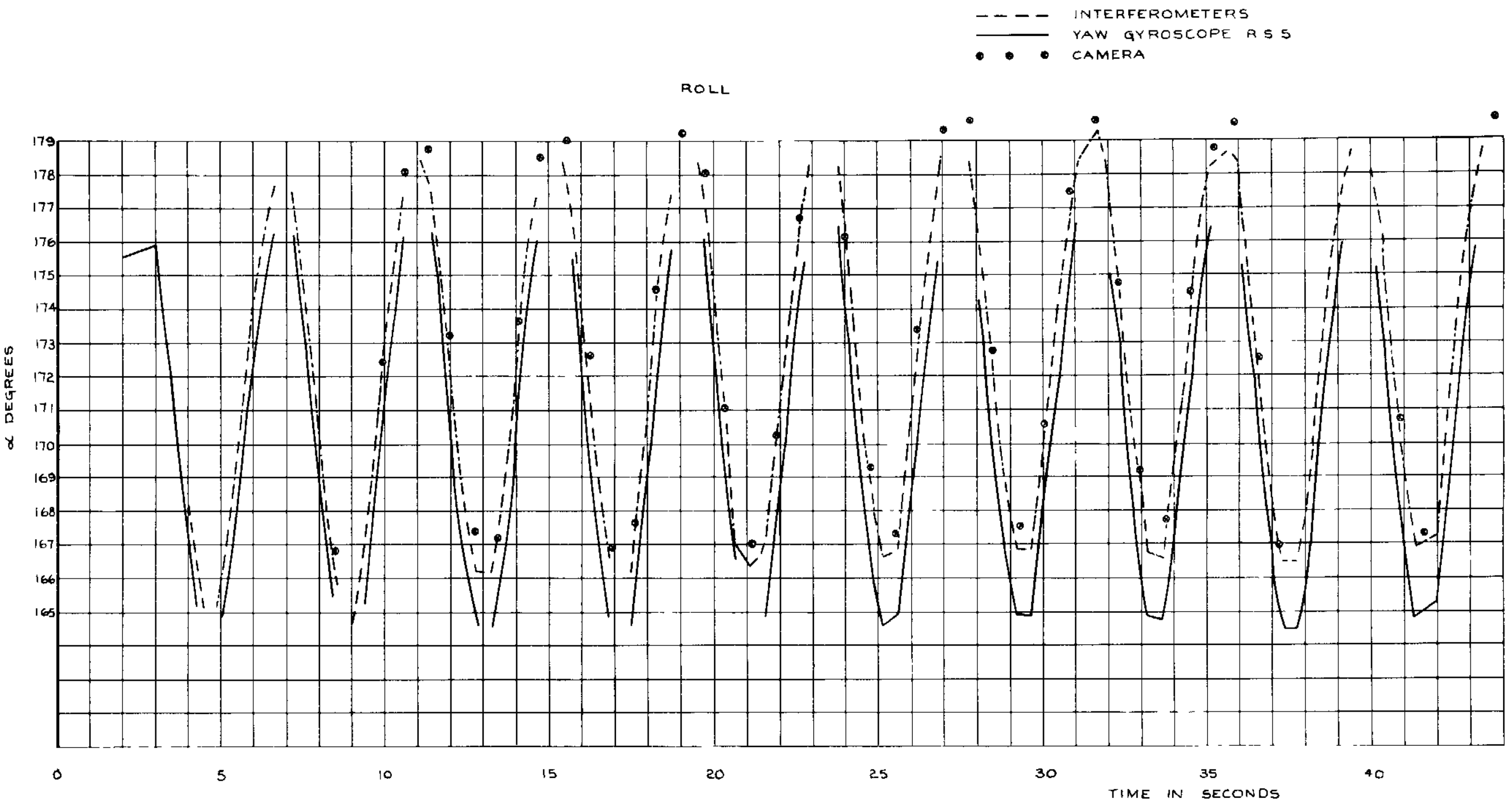


FIG.23 SL.32 ROLL ATTITUDE OF THE HEAD (0 TO 44 SEC)

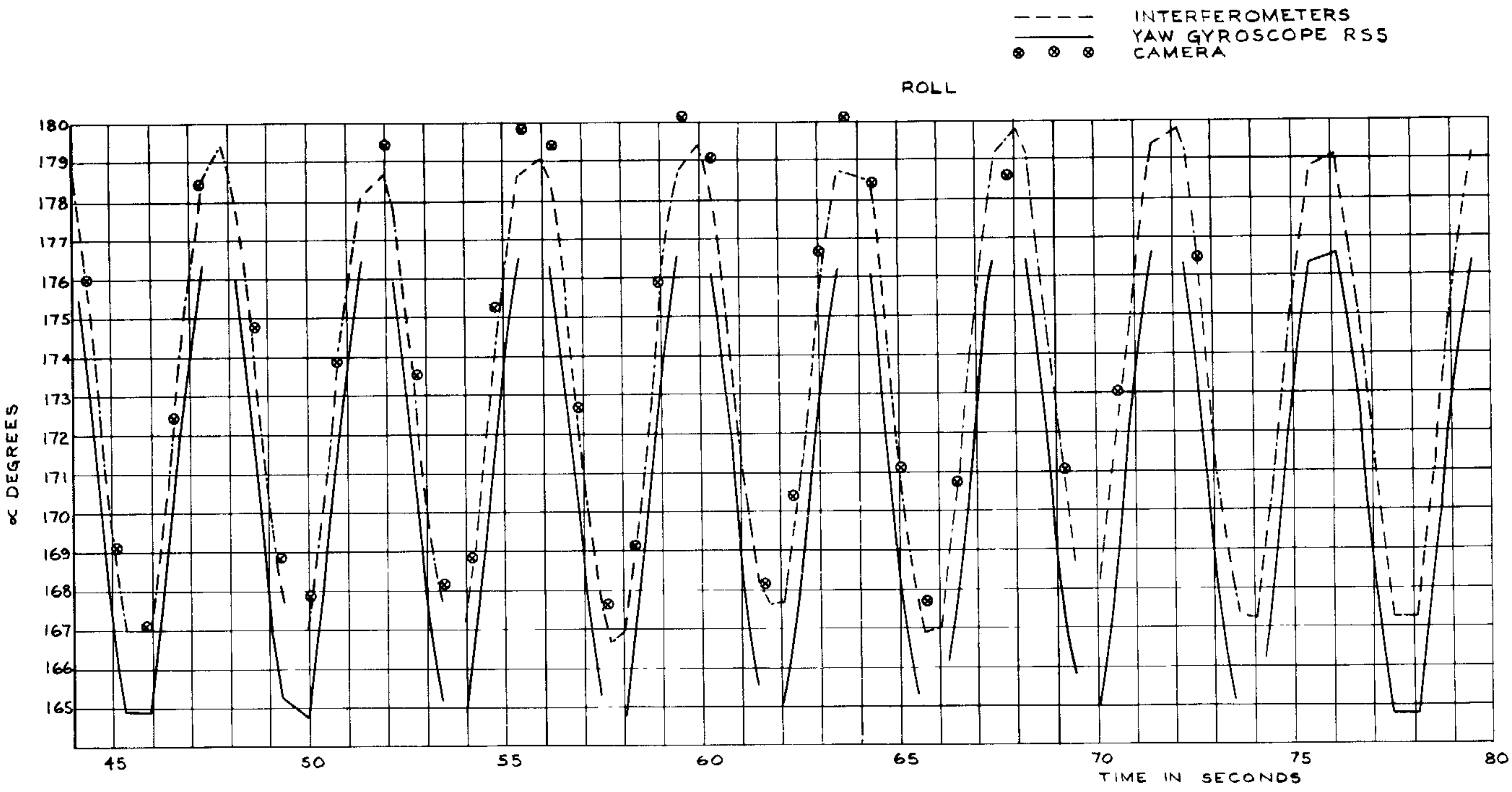


FIG.24 SL 32 ROLL ATTITUDE OF THE HEAD (44 TO 80 SEC)

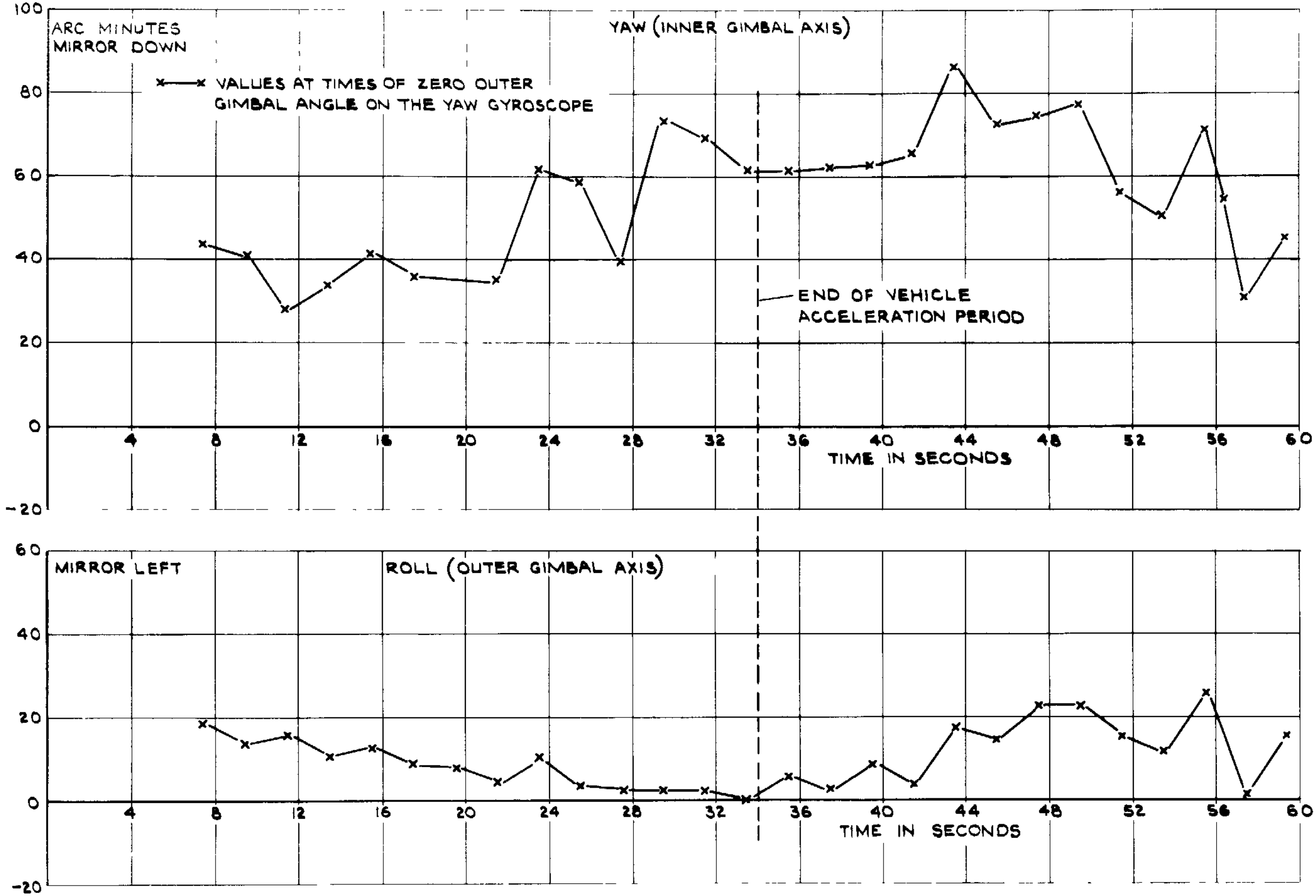


FIG.25 SL27 TOTAL DRIFTS FROM LAUNCH OF YAW GYROSCOPE RS 9 (POSN No 1)

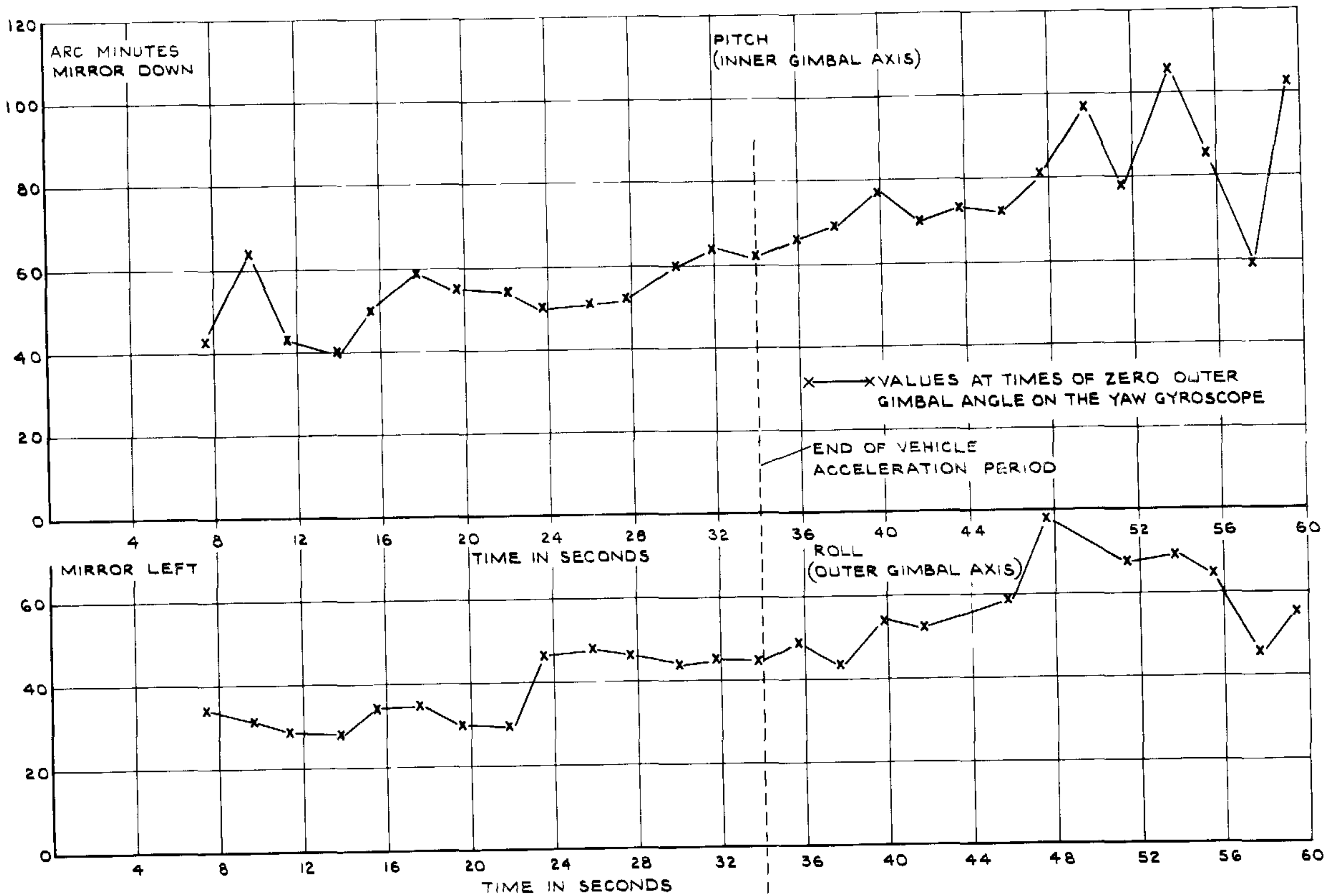


FIG. 26 SL27 TOTAL DRIFTS FROM LAUNCH OF PITCH GYROSCOPE R.S.16 (POSN No 2.)

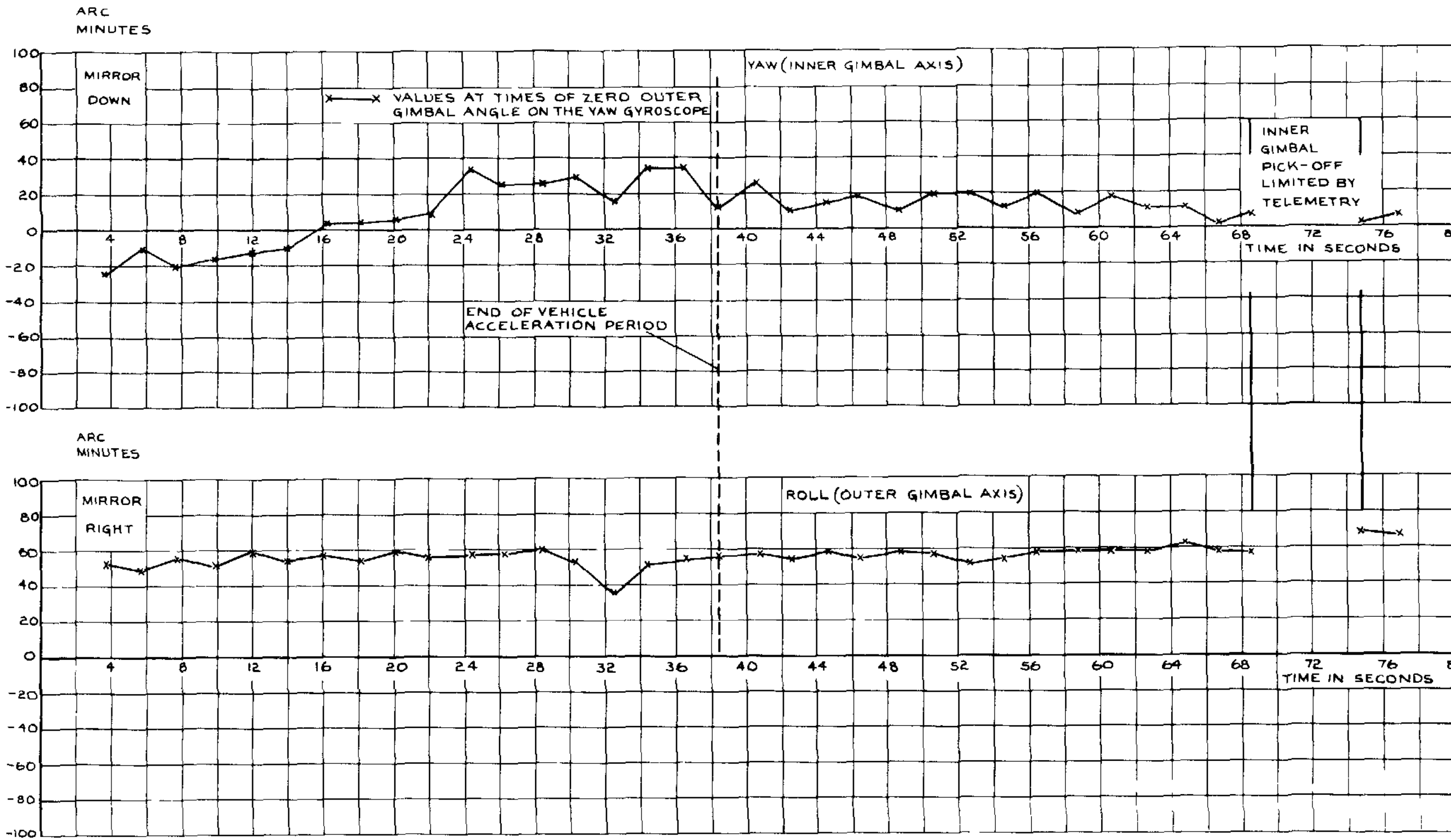


FIG.27 SL32 TOTAL DRIFTS FROM LAUNCH OF YAW GYROSCOPE RS 5 (POS № 1)

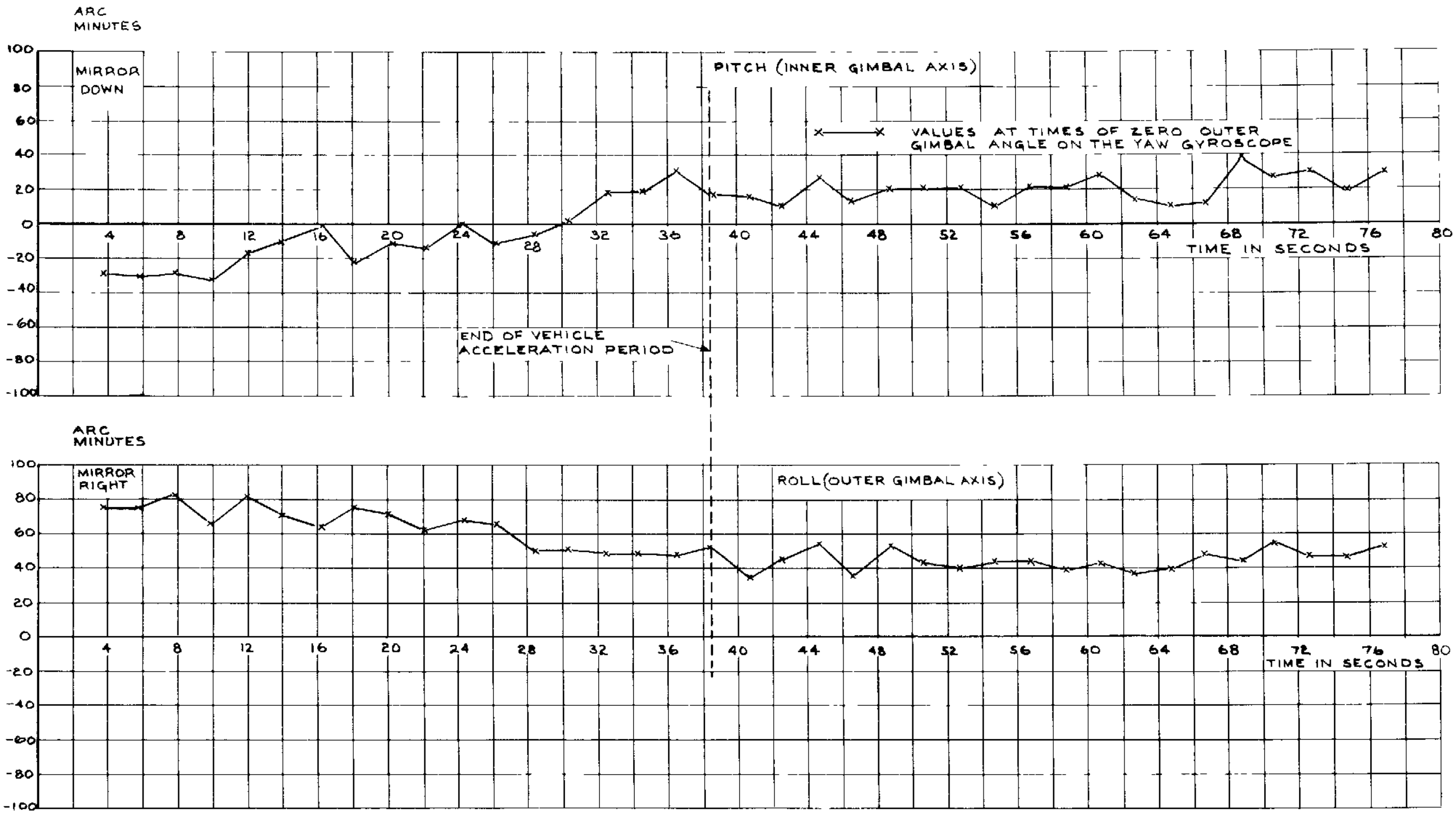


FIG. 28 SL32 TOTAL DRIFTS FROM LAUNCH OF PITCH GYROSCOPE RS 22 (POS № 2)

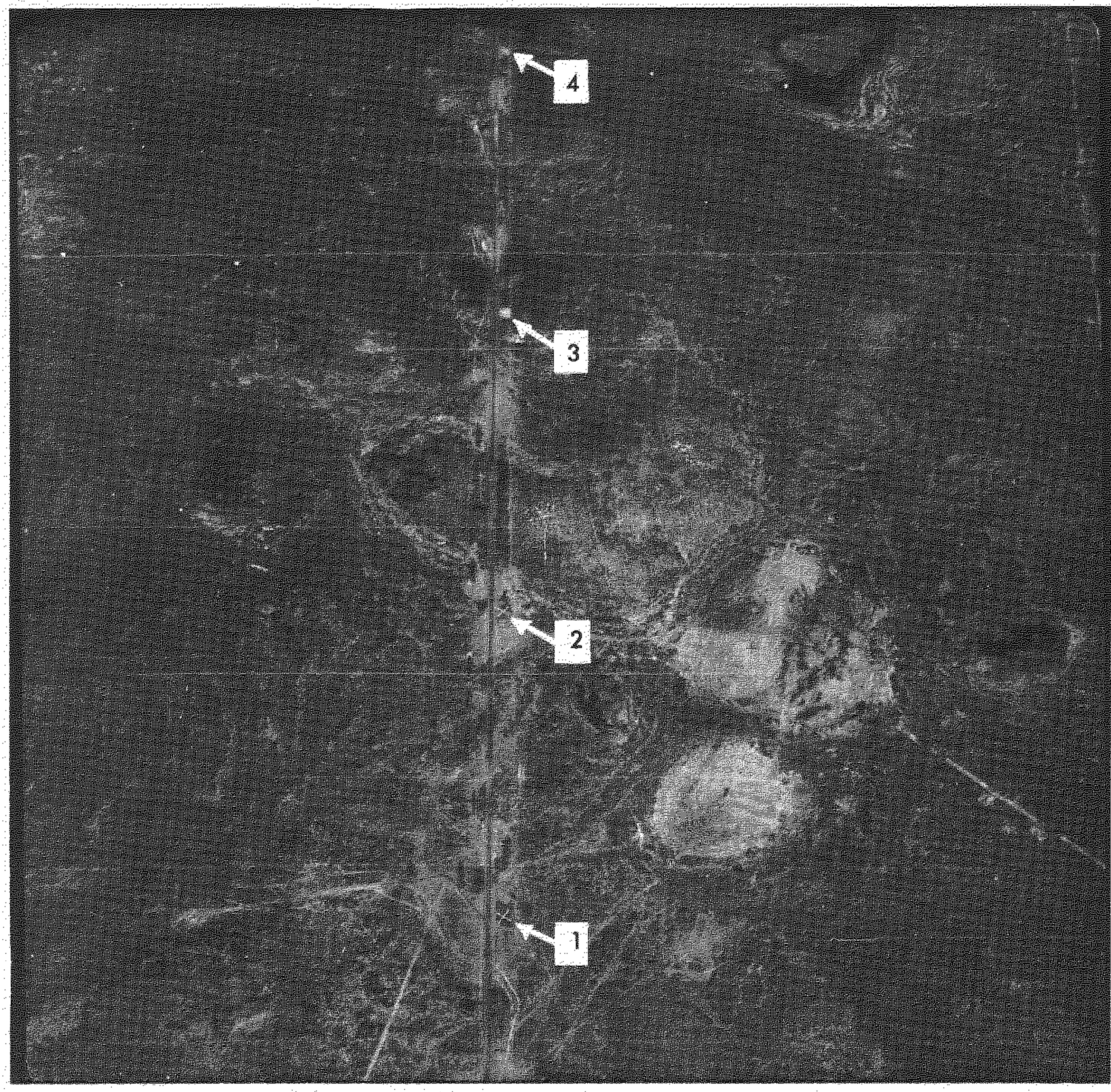


Fig.29. Photograph of ground markers taken from Skylark SL32 at an altitude of 7,000ft (approx)

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ARC CP No.1147
September 1968

629.13.014.1 :
629.13.052.333/335 :
531.113 :
629.19

Knott, F.R.

THE DRIFT OF AN AUTO-PILOT GYROSCOPE DUE TO PROLONGED ACCELERATION IN THE SKYLARK ROCKET (Abridged version)

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Rocket attitude obtained from the gyroscopes under test is compared with rocket attitude determined from other instrumentation systems to obtain gyro-drift. The principal reference system used to determine missile attitude is provided by three radio-interferometers fitted to the rocket. A secondary reference system comprises an aerial-survey camera which is installed in the vehicle and used to photograph ground markers.

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(Over)

DETACHABLE ABSTRACT CARDS

The instrumentation referred to has been developed and flown in a series of Skylark rockets. The programme of firings is summarised and the accuracy of the instrumentation systems assessed. Ground equipments used to carry out pre-flight drift measurements on the gyroscopes under test are also briefly described.

Typical gyro-drift data obtained from flight results and measurements made before flight are given, together with conclusions based on data from twelve gyroscopes tested during the programme.

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