History of Key Technologies

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Origins of Inertial Navigation

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Introduction

A VIGNETTE[1] of the stages of development of inertial navigation at the M.I.T. Instrumentation Laboratory has been provided by the author. An earlier work[2] provides a tutorial reference on the basic principles of the instruments and systems that were developed. Because this technology grew largely from classified programs, publication of the description of the technology for specific systems has often lagged by a decade or more, and sometimes has lacked coverage altogether. Also because of the author's responsibility for these classified programs, he was often constrained from publishing until after someone less intimately involved had already disclosed information on the programs of inertial navigation.

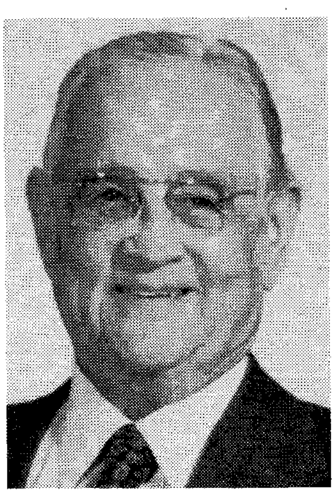
Even at this late date, a description of the Mark 14 "Draper Sight" has not appeared in the open literature. It is the purpose of this treatise to describe some of the major systems that contributed to the evolution of inertial navigation, and to provide a record of the performance of some of these historically significant programs.

An underlying theme is the continuity of gyroscopic sensor technology from the fire control systems of the Mark 14 vintage to the first successful inertial navigators. It is shown how the single degree of freedom gyroscopic element was adapted to perform simultaneously both the sensing and computation requirements for solving the fire control problem associated with close-in attacking aircraft. Refinements made to this gyroscopic element, including precise control of torques applied to the output gimbal, and eventual flotation led to the sensor technology that was eventually adapted to solving the inertial navigation problem.

Naval Antiaircraft Fire Control with Mechanized Stabilized Coordinates

Antiaircraft fire control became important for navies of the allied powers as the days of World War II lengthened with engagements in the Mediterranean, the Atlantic, and the Pacific. Close-in simultaneous dive bombing and Kamikaze multiple attacks by rapidly maneuvering aircraft turned against surface ships. This situation developed during a time when the writer, as a Professor of Aeronautical Engineering at the Massachusetts Institute of Technology was teaching instrumentation and control to groups of graduate students that included Naval and Air Force officers. The Instrumentation Laboratory of the Department, which was devoted to projects of real-world useful technology, was an ideal place for students to conceive and carry ideas into experimental hardware and operational tests. A number of people accepted antiaircraft fire control as an interesting and most important teaching laboratory project.

A review of existing Naval fire control equipment showed that it had been designed primarily for shooting at slowly moving ships or for shore bombardment. Ranges were long, and targets generally so rugged that relatively heavy guns were used. System errors required manual spotting after initial salvos. The generally used fire-control system included a tracking unit and a gun mount, both arranged with directional components to rotate in train and in elevation with respect to the deck. Essential parts of the overall system included a stable element and a gyrocompass. A deck tilt corrector was arranged to convert indications given in deck corrdinates into stabilized coordinates for computation of lead, gravity drop, etc. A second unit for geometry, the trunnion tilt corrector, was arranged to convert stabilized coordinates into deck coordinates for the gun mount drives. Course and speed of the host ship were entered automatically from ship's navigation equipment, while course and speed of the target were initially estimated and manually set into the computer. Initial inaccuracies of these manual inputs could cause unacceptably long solution times. Operation was largely by mechanical components such as ball and disk integrators, gear differentials, and shafting with synchros for transmission of signals over complicated connections. The dynamic range, accuracy, and reliability cause severe limitations in using these systems. Even with these difficulties,



Charles Stark Draper was born in 1901 in Windsor, Missouri. He attended high school at Windsor, spent two years at the University of Missouri, and then went to Palo Alto, California where he received a B.S. in psychology from Stanford University in 1922. He entered the Massachusetts Institute of Technology in September of 1922, graduating in 1926 with an S.B. in electrochemical engineering. During 1927 he attended the Army Air Corps Flight School at Brooks Field, Texas, and returned to M.I.T. for an M.S. (without specification) in 1928. There he became interested in aeronautical powerplants and, in 1930, began to devote his time to developing instruments for recording cylinder process measurements and to teaching courses in aircraft instruments. At the same time he worked toward a Ph.D. in physics with a minor in mathematics; he received this degree in 1938. During the 1930's Draper's interest shifted largely to measurement and control and the instrumentation technology by which these functions are accomplished. He helped develop antiaircraft fire control equipment for the U.S. Navy and the Army Air Corps that was widely used during World War II. After 1945, he and the Instrumentation Laboratory, which he started at M.I.T. (divested from M.I.T. in 1973 and now known as the Charles Stark Draper Laboratory, Inc.), conceived and created inertial guidance systems for aircraft, submarines, and missiles. These subsequently became standard equipment for U.S. armed forces and civilian airlines. In the 1960's his laboratory conceived, designed, and assisted into operation the guidance, control, and navigation equipment for the Apollo moon landings. Dr. Draper has written several books and many articles on control and instrumentation, with an emphasis on comprehensive systems using sensors of higher and higher performance. He believes that this field will have increasing importance as the years pass, and that many excellent opportunities for creative work lie in the years ahead.

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EDITOR'S NOTE: This manuscript was invited as a History of Key Technologies paper as part of AIAA's 50th Anniversary celebration. It is not meant to be a comprehensive survey of the field. It represents solely the author's own recollection of events at the time and is based upon his own experience.

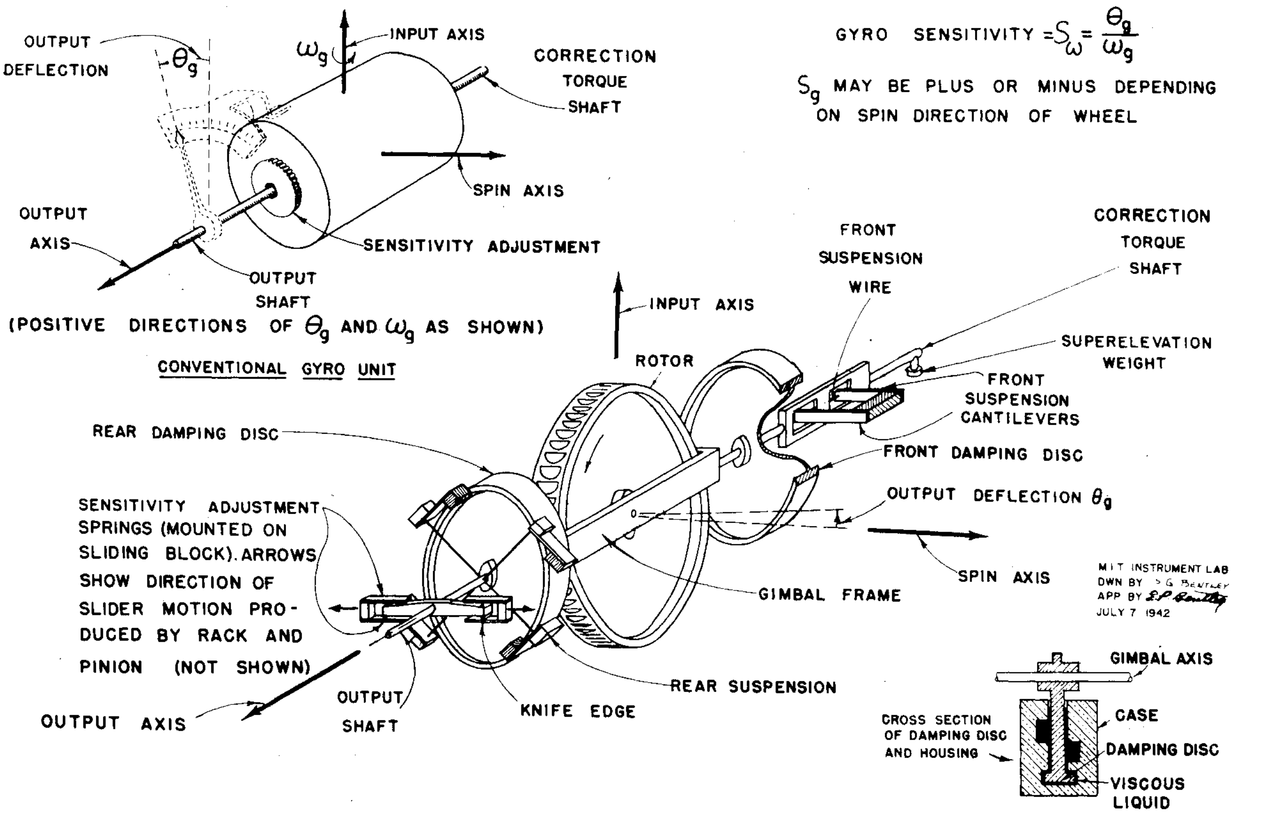


Fig. 1 Schematic diagram showing the essential elements of the gyroscopic rate-of-turn indicators used in gunsight Mark 14.

the 5 in. 38 caliber gun with its stabilized coordinate system was a formidable weapon for one-on-one situations. Under multiple close-in bombing and Kamikaze attacks, the number of stabilized coordinate systems was much too small, they were too large, weighed too much, and were too expensive. In particular, stabilized coordinate systems certainly were not useful for controlling fire of the large number of 20 mm and 40 mm guns that were available and to a considerable extent installed on the decks of many naval ships.

Unstabilized Coordinate Fire Control System with Tracking Stability and Solution Time Controlled by Viscous Damping

News media and discussions with my students during the times of World War II brought strongly to my attention the serious nature of multiple dive bombing and Kamikaze attacks on Navy ships. Because the important ranges for operation were short and targets often maneuvering rapidly, it was realized that the principal component of prediction angle must be associated with target motion during times of flight for projectiles, while gravity drop, windage, drift, jump, and other trajectory refinements were probably not very important. Accepting these ideas, I realized that the essential component of angle for the gun to be pointed ahead of the line of sight to the target depended on time of projectile flight to the target, which could be related to present range and the angular velocity of the tracking line. From the standpoint of instrumentation, it was understood that the gyroscopic instruments, such as the aircraft rate-of-turn indicator, could be used to indicate angular velocity with output for a given input dependent on stiffness of the spring restraint and the amount of damping used.

Gyroscopic Unit for Unstabilized Fire Control Systems

With attention directed toward elastically restrained, damped, indications of angular velocity for antiaircraft fire control purposes, the writer remembered that in his laboratory were two experimental rate-of-turn indicators with gimbal support ball bearings replaced by steel wires and heavy viscous damping to prevent damage by shock and vibration. It was realized that these experimental aircraft instruments might provide the essential computer elements for unstabilized short-range antiaircraft fire control. This suggestion was made to the Sperry Gyroscope Company, the organization that had originally supported the aircraft instrument work with a grant of about $500. The company was kind enough to allow $1500 for designing and building the first instruments.

Figure 1 suggests the arrangement of features in the gyroscopic turn indicators after the addition of a sliding block to carry sensitivity adjustment springs. The gyroscopic rotor was spun by an air jet because time and money were too short to permit the design and building of electrically driven wheels. The rotor-carrying gimbal frame had a damping disk on each end with a matching housing having no solid mechanical contacts to the disks. Each disk was partially immersed in temperature-controlled heavy viscous fluid with amounts chosen so that there were always clearances, but leakage did not occur for any position of the overall unit.

On one end, the gimbal frame was supported by a wire stretched between leaf springs. The other end was carried through crossed pieces of piano wire attached to the instrument case (not shown in Fig. 1) by means of flat springs. Sensitivity adjustment springs acting on knife edges attached to the output shaft were used to vary elastic restraint on the gimbal by means not shown in the diagram.

Unstabilized Fire Control Systems

Figure 2 shows the elements of the fire control problem attacked by the writer and his associates. The case of the computer was mounted so that it moved in train and elevation with the gun, either under direct drive by human operators or by action of automatic power drives. In early models, reticle patterns controlled by computer mechanisms, indicated the present target line for the operator who moved the gun so that the controlled line, aligned to the gun, pointed in the proper direction for projectiles to strike the target. The resultant

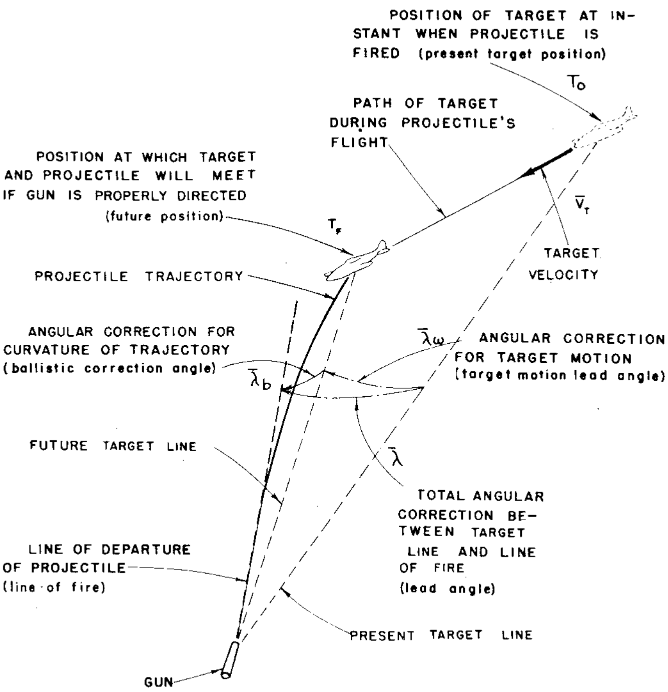


Fig. 2 Diagram illustrating the essential elements of the fire control problem.

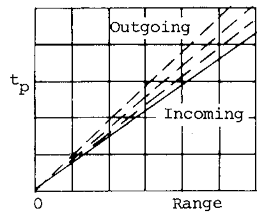
angular correction between the present target line and the future target line is shown as made up of a target motion lead angle and an angular correction for curvature of the trajectory that was commonly much smaller than the lead angle. Correction angles became smaller as the target came closer to the gun so that proper adjustments for range to the target had to be determined and rapidly set into the computer mechanism.

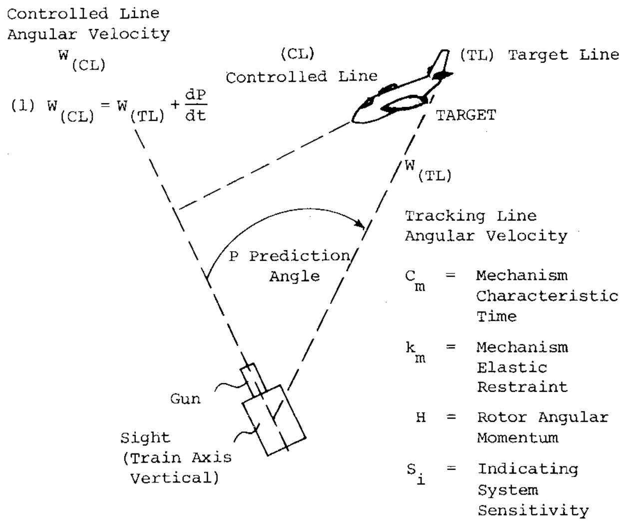
The first unstabilized coordinate fire control arrangement to be designed, produced and used operationally in considerable numbers for Oerlicken 20 mm machine guns was gunsight Mark 14 with capabilities for dealing with projectile times of flight up to about five seconds.

The basic mechanism of this sight consisted of two "rate gyros" with range-variable spring restraints having the general features of the unit shown in Fig. 1. The input axis of one gyro unit was aligned to the elevation axis of the gun, while the other input axis was at right angles along the traverse axis of the gun. The gyro unit output axes were both parallel to the barrel of the gun.

Weights fixed to arms attached to the gyro output axes were aligned to receive the proper gravity forces for providing superelevation correction torque components to the elevation axis and the traverse axis. The resultant prediction angle was determined by the gyroscopic torques acting against the elastic restraints of springs, and the viscous damping drag torques due to angular velocities of the gimbals with respect to fluid filled housings. The most important balancing effort was that associated with the ratio of angular momentum to the elastic coefficient which, together with the sensitivity of the reticle indicating system, had the dimension of time and largely determined the output angle for angular velocity input. In practice, this sensitivity was related to prediction time, the theoretically calculated value for sensitivity with all the available factors taken into account. By choosing range scale calibrations for the sight to give optimum results for likely situations, maximum effectiveness could be obtained under combat situations. When Mark 14 gunsights were first applied in significant numbers, accurate information on present range was not generally available, so that the averaging process of determining calibrations for range probably did not have very important effects.

In practice, S_m is set to t_p, defined as Prediction Time by analysis of assumed operations, as a function of present range.





$$(1)\quad W_{(CL)} = W_{(TL)} + \frac{dP}{dt}$$

MECHANISM PERFORMANCE EQUATION

$$(2)\quad C_m \frac{dP}{dt} + k_m P = HS_i W_{(CL)}$$

$$(3)\quad C_m \frac{dP}{dt} + k_m P = HS_i W_{(TL)} + \frac{dP}{dt}$$

Rearranging Terms

$$(4)\quad \left(\frac{C_m}{k_m} - \frac{HS_i}{k_m}\right)\frac{dP}{dt} + P = \frac{HS_i}{k_m} W_{(TL)}$$

Define $S_m = \frac{HS_i}{k_m}$ = Mechanism Sensitivity

$\sigma T_m = T_s$ = Sight Characteristic Time

so that $\sigma = \left(\frac{C_m}{HS_i} - 1\right)$ = Stability Number

$$(5)\quad \sigma S_m \frac{dP}{dt} + P = S_m W_{(TL)}$$

Sight Performance Equation

Derivation Summary: Geometrical, physical, and mathematical relationships that determine basic performance equation for unstabilized antiaircraft sight.

Gunsight Mark 14 covered ranges up to 2000 yards and about 5 s times of flight. Weight on the gun was about 25 lb, and cost near the end of production for about 90,000 units was in the region of $300 each. With many Mark 14's on Oerlicken guns ringing the edges of large ships, reports noted that substantially every one of the aircraft forming simultaneous attack groups were shot down or driven away.

Dynamics of Unstabilized Coordinate Sights

General principles available for realizing antiaircraft fire control systems without the use of stabilized coordinates have been reviewed, and gunsight Mark 14 described as a particular example to provide elements of defense against aircraft targets at short ranges. The general pattern of actions needed to generate prediction angles between lines of sight to targets and the necessary direction of gun fire has been treated from the standpoint of quasistatic, that is, not rapidly varying events. This point of view may be reasonable enough once a nearly correct prediction angle has been brought into existence and a satisfactory state of tracking is being maintained. However, the dynamical problems of tracking and solution time must both be solved effectively before any equipment for fire control can be considered as satisfactory. The geometrical and mechanism operation relationship associated with results from gunsights based on unstabilized coordinates are reviewed in the Derivation Summary.

In this summary, Eq. (1) suggests simplified circumstances in which the train axis is at right angles to the plane of the paper, and no rotation about the controlled line exists. For this situation, Eq. (1) shows that the angular velocity of the controlled line is equal to the angular velocity of the tracking line plus the time rate of change of the prediction angle.

Equation (2) is the operating equation for the sight mechanism when the prediction angle responds to the angular velocity of the controlled line as the forcing function. The mechanism parameters are the mechanism viscous damping coefficient C_m, the mechanism elastic restraint k_m, the rotor angular momentum H, and S_i the angle sensitivity of the optical system. Equations (3) and (4) are forms of Eq. (2) with tracking line angular velocity replacing controlled line angular velocity. Equation (5) introduces the stability number σ and the mechanism sensitivity S_m. It is to be noted that in operations against aircraft targets, S_m is set into the mechanism by means of a calibration worked out on the basis of carrying through assumed attack situations to determine the prediction time, t_p, as an averaged function of range to the target for the circumstances of a reasonable variety of attacks.

For any given attack, present range may be estimated or taken from the output of some range-indicating device. However, even with perfect range inputs, satisfactory results can be achieved only if movements of the controlled line as responses to indications of reticle deviations from the tracking line actually reduce magnitudes of these errors and keep them within satisfactorily small magnitudes. Equation (5) of the Derivation Summary represents, in approximate terms, the angular changes associated with tracking. Inspection of Eq. (5) indicates that the stability number, σ, must be finite and positive for stable operation.

Much experimental work showed that a stability number value of about 0.2 generated good results under operating conditions.

With tracking stability assured by the selection of a proper stability number for an unstabilized coordinate antiaircraft fire control system, the next important problem is that of solution time; the length of time required to go from some convenient initial situation, such as the control line in coincidence with the tracking line and range set at some reasonable estimated or indicated distance, to the situation for effective firing. If the tracking line angular velocity may be considered substantially constant in comparison with the change occurring as the prediction angle develops, the general nature of the dynamic effects associated with solution time can be understood by considering the exponential transient term associated with solutions of Eq. (5). This term is an exponential with σS_m, sight characteristic time, as the multiplier for time in the exponent. If a reduction of the transient error to about 0.02 of the initial value is satisfactory for defining solution time, four sight characteristic times are required. For gunsight Mark 14, this solution time corresponds to about one-half of the projectile time of flight for the range existing when the solution is accomplished. That is, for a range of 2000 yards corresponding to a prediction time of about 5 s, a reasonably satisfactory fire control solution would be developed in a time of 2-3 s.

One other effect associated with damping action included in the mechanism of gunsight Mark 14 appears when the case of the sight is rotated about the controlled line with reasonably large prediction angles. Situations of this kind appear when the ship carrying the sight and a gun, or a director supporting structure, rolls and pitches because of waves on the sea. The resulting rotations of the elevation axis and the traverse axis about the controlled line carry the corresponding elevation and traverse components of prediction angle with their motions. The input components that generated these components change, but mechanism damping introduces lags in adjustments for the elevation and traverse angles. In effect, the prediction angle is "dragged around" with rotation about the controlled line, which is called cross roll.

Cross roll also appears when the target follows a relatively close-in crossing course that causes the prediction angle to change from predominantly about the traverse axis to predominantly about the elevation axis near the minimum range point. For the short ranges used in gunsight Mark 14 no features were included to correct for cross roll effects. Later models of unstabilized coordinate antiaircraft fire control systems did incorporate provisions for cross roll corrections in their mechanisms.

Further Developments of Unstabilized Coordinate Antiaircraft Systems—Gunsight Mark 15

Mark 14 gunsights had demonstrated so quickly and so well their usefulness in protecting Naval vessels from close-in simultaneous attacks by many airplanes, that the Instrumentation Laboratory at M.I.T. was given an opportunity to conceive and design a model with extended capabilities to work with the 40 mm Bofors, the 5 in. 38 caliber, and the 3 in. 50 caliber guns. Projectile times of flight and ranges were changed with goals some three to five times those built into the Mark 14. Director stands to support the sights in locations away from the smoke-obscured visibility regions near the firing barrels were used to point the guns by remote control

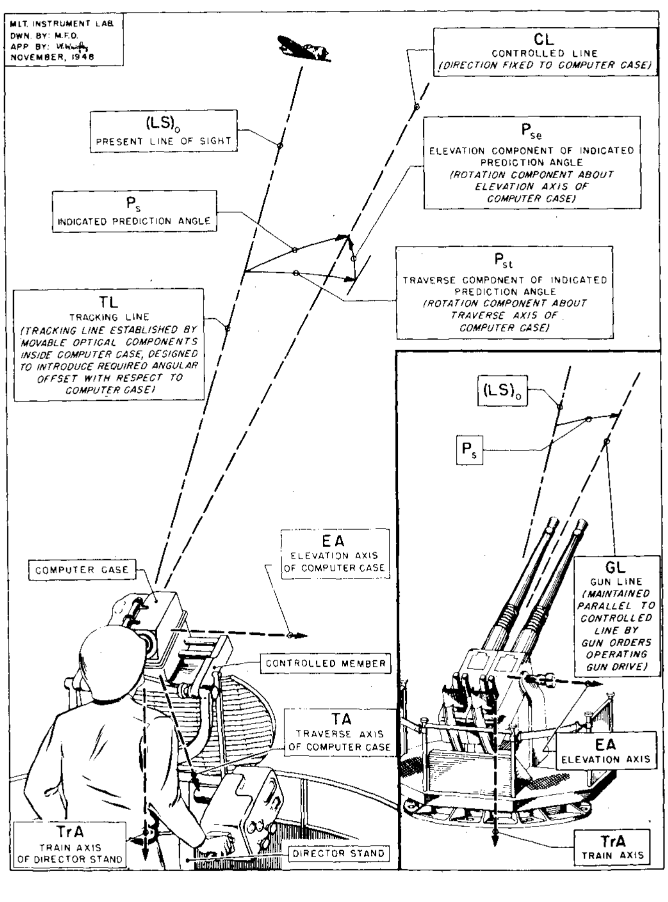


Fig. 3 Essential features of a fire control system with computer operating in unstabilized computer coordinates, using optical tracking with manual director drive.

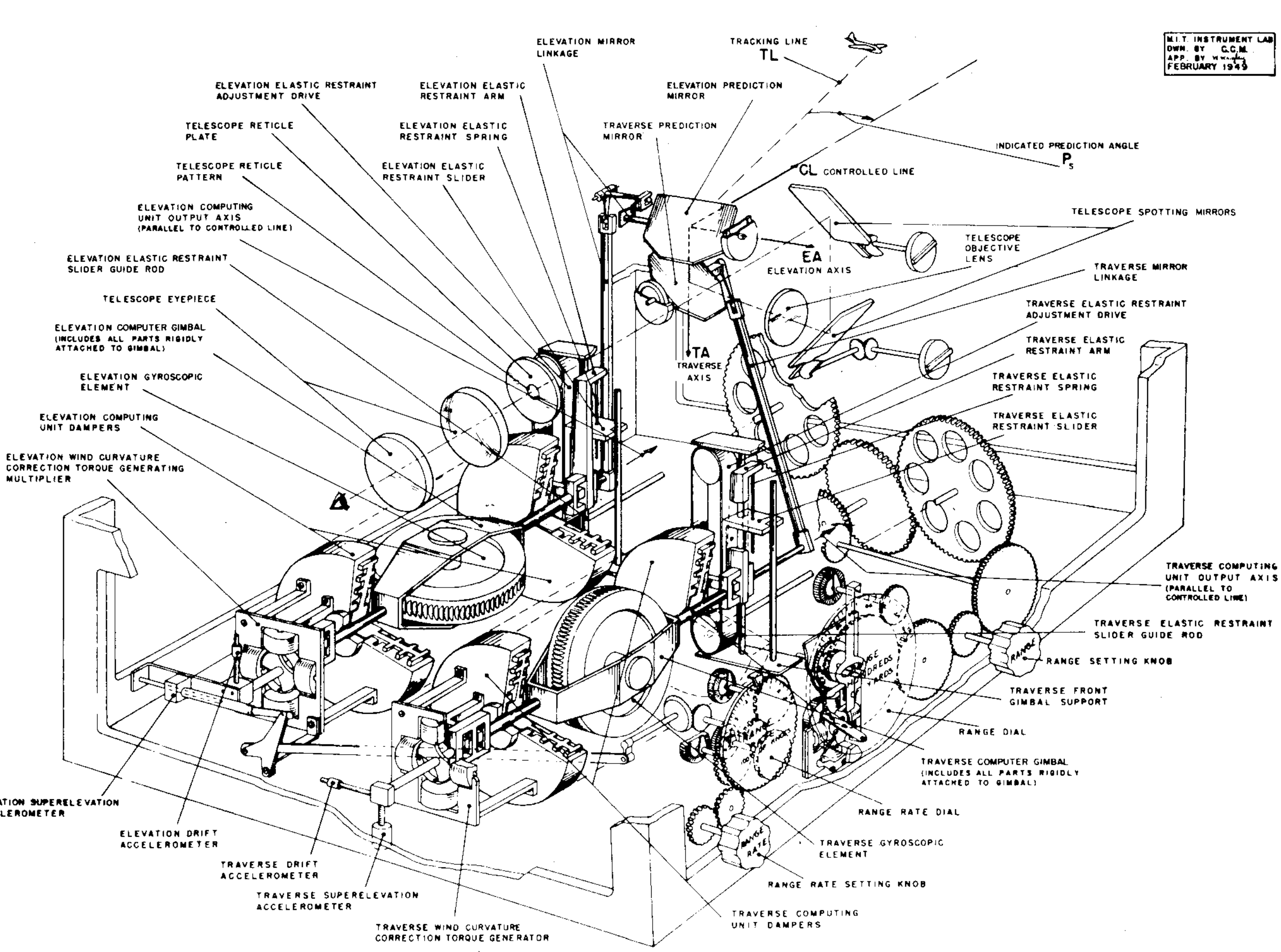


Fig. 4 Pictorial schematic diagram of gunsight Mark 15 showing the essential mechanism features used for optical tracking.

from servo-signals. The first of these off-mount stands was the Mark 51, which carried the Mark 14 by hand power operation and could be used only in association with 40 mm, 3 in. 50 caliber and 5 in. 38 caliber guns because they incorporated standard power-driven mounts.

With improvements in the sight it became important to make sure of accurate and reliable indications of present range. The first step in this direction was the Mark 52 director which carried gunsight Mark 15 on a hand-powered stand that was also provided with a range-only radar gimbaled to point in the direction of the target. Indications of range were automatically set into the sight as the reticle was placed on the target and a solution developed by moving the sight case so that the reticle remained on the target. The servodrives for the gun maintained the gun line ahead of the tracking line established by the sight, so that effective fire was delivered when the gunner operated his switch.

Wind and strong ship motions made it difficult for the operator to move the Mark 52 director smoothly, so an improvement was introduced by fixing a radar antenna dish support with angular freedom on top of the gun mount with servodrives to keep the radar tracking line close to the target line of sight, and provide signals for the operator to follow in controlling tracking motions of the sight.

Mechanism Features of Mark 52 Director and Gunsight Mark 15

Figure 3 shows the general features of the hand-powered Mark 52 director provided with range-only radar tracking and carrying a Mark 15 gunsight. The figure suggests control of a 40 mm Bofors twin gun mount by electrical signals from the director. This arrangement would have been that of the Mark 63 gun fire control system if a tracking radar antenna had been shown as mounted on the gun.

The basic principles and instrumentation used in gunsight Mark 14 were applied in gunsight Mark 15 as suggested by Fig. 4, with prediction time and range inputs extended by more than twice, as indicated by the curves of Fig. 5. Superelevation corrections were continued with refinements beyond those of Mark 14. Wind corrections, drift corrections, wind-curvature corrections, velocity jump, and cross roll corrections were provided. The computed elevation and traverse angles were transferred by mechanical leverages from the gyro output axes with about a 1 to 10 ratio, to mirrors mounted outside the objective lens of the tracking telescope. This arrangement gives a reticle image visible to the operator who looks through the telescope eyepiece. The reticle acts as the tracking index when visibility can be used; a radar spot is provided for the same purpose in the Mark 63 system when visibility is obscured. Range and range rate are set in by hand or by an automatic drive.

Electromagnetic torque generators receiving signals from direction and velocity inputs to determine effects on the output angles of the gyro gimbals were designed to compensate for the action of wind on projectile flight. In some models of the Mark 15, torque generators of the same kind were used for the purposes of realizing cross roll corrections. These corrections were based on signals representing cross roll angular velocities of the sight about the two computing gyro output axes, and were used as torque generator inputs to compensate for prediction angle rotation effects associated with viscous damping. The cross roll angular velocity inputs were sensed by a third gyro unit with its input axis fixed parallel to the computing gyro output axis direction.

Note in the upper right hand corner of Fig. 4, that there are hand knobs for elevation and traverse spotting corrections. These are vestiges from ship-to-ship and ship-to-shore fire control tactics, which remained as requirements for an-

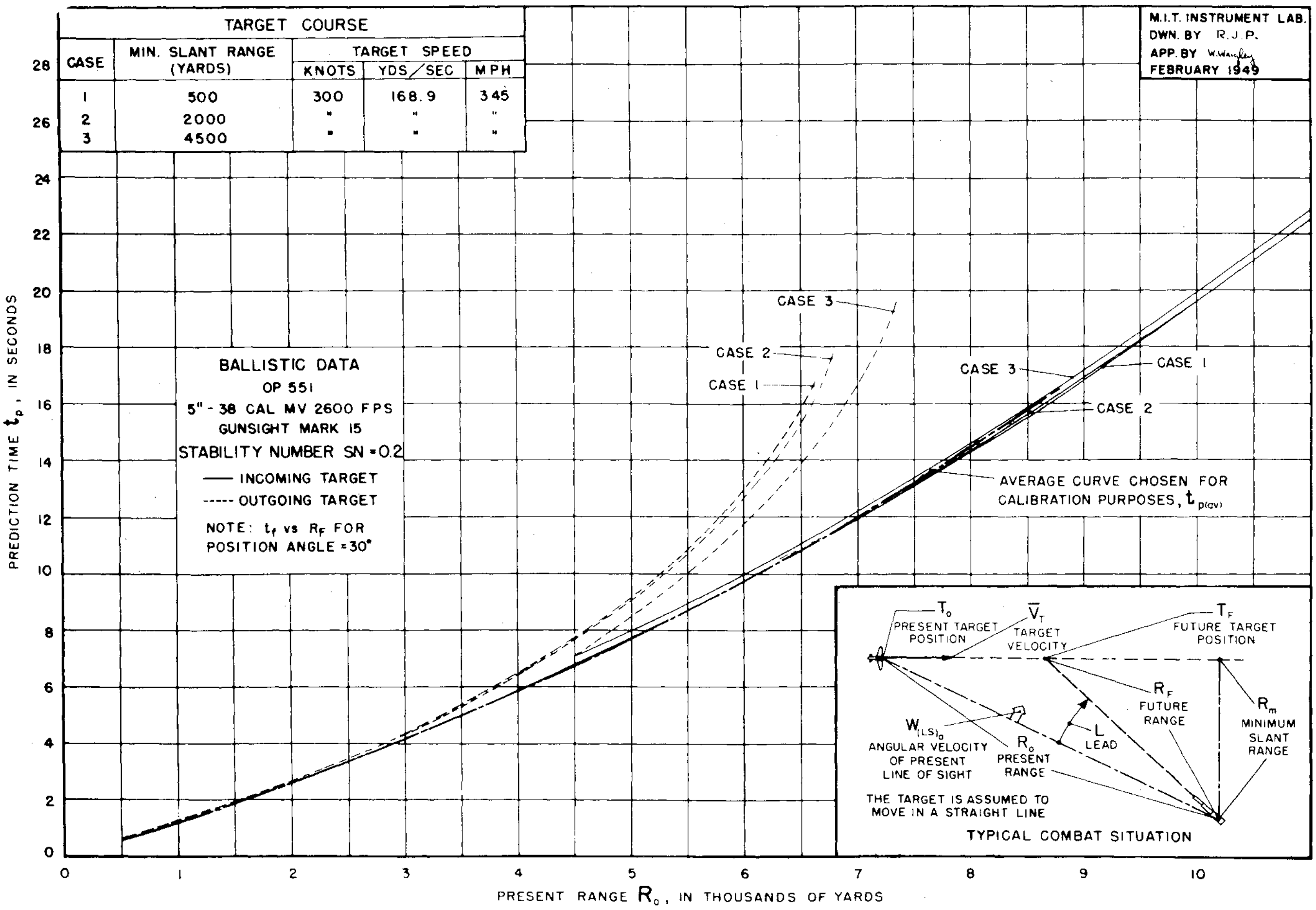


Fig. 5 Variation of prediction time with present range for typical combat situations.

tiaircraft fire control in World War II. During evaluation testing, these knobs were surreptitiously disconnected—many test crews were delighted to watch the positive results of their spotting corrections as the tracer bullets converged on the target seemingly resulting from their efforts!

An inspection of Fig. 5 shows the result of converting gun ballistic tables to useful instrumentation functions for the Mark 15. The curves are corrected for the lag caused by the mechanism damping. It is clear that the prediction time is not a strong function of range rate for targets that will come close to the defended ship. Also, it is clear that for the Mark 14, the range could be set at a constant for which an incoming aircraft crosses the line of fire.

Results from firing trials made with Mark 15 gunsights combined with Mark 63 radar automatic ranging and tracking systems were favorable enough for further developments of unstabilized coordinate fire control systems to be considered. Already, discussions of physical principles, and applications of design engineering to working models of fire control systems for use in fighter aircraft had been carried into the stage of working models with Air Force students of my classes. Details of the applications differed somewhat from those applied in antiaircraft systems, but the basic principles were identical for all cases. Aircraft installations required that computing units be separate from optical units, which displayed reticles by reflection for windshields, with signal-controlled servodrives to transfer required motions. Angles were larger, and prediction times involved much greater ranges of values. These new requirements meant modified computer parts with "electrical springs" replacing mechanical elements. Orientations of units with respect to acceleration and gravity changed over increased ranges in operation so that dampers had to be redesigned. Space and weight dictated that the airborne sights were limited to yaw and pitch computers with no cross roll gyro. However, it was found that the pilot did need at least a partial cross roll correction during tight lead pursuit attacks to maintain tracking. This correction was provided by tilting the yaw gyro input axis 10 deg about the pitch axis to couple part of the roll rate into the yaw computer. The prediction time for the yaw and pitch computer were then calibrated for all residual effects of damping and cross roll.

Various models of the airborne gunsight were introduced to match the needs of fighters and also of bombers. For example, the so-called A-4 system was used in the F-86 fighters that operated over Korea. Space is lacking here to review the details of particular systems, so it will only be noted that nonstabilized fire control systems found a number of applications for aircraft.

Unstabilized Fire Control Systems with Radar Automatic Tracking and Ranging—The Gunar System

The Gunar Antiaircraft System was intended to apply the technology learned from production and use of Mark 14 and Mark 15 systems, to realize an arrangement with automatic radar tracking and ranging in which the operator did not need exposure on deck so that he could be protected by gun mount structure. Refinements of computer theory and design were directed toward long ranges and more accurate corrections using electromagnetic elastic restraint and electrical output signal generators. These and other possibilities for advancement were being engineered, built, and tested during the middle years of the 1940's decade when the U.S. Navy, on the basis of new developments, decided to substantially abandon guns for antiaircraft defense of ships in favor of rockets. This decision effectively ended projects of the Instrumentation Laboratory in the field of defense against aircraft although a few 3 in. 70 caliber and 5 in. 54 caliber gun systems were built.

Motivation for Self-Contained Navigation Systems

Completion of systems for aircraft and ending of World War II apparently left the Laboratory without viable projects for the Air Force. It was in this situation that the writer started discussions with then Colonel Leighton I. Davis who, as commandant of the Armament Laboratory at Wright Field, considered the general subject of control, navigation, and guidance systems for aircraft.

Long-range transport and bombing missions that became basic operations during the war introduced strong requirements for self-contained control, navigation, and guidance equipment of substantially perfect reliability and great accuracy. The key characteristic was operation for complete missions without continuous links of either natural or artificial radiation to external points. By using mechanization that does not require radiation contacts, difficulties from weather, darkness, terrain, the absence of complete patterns of cooperating stations, enemy interference, and, in fact, all effects except actual destruction of equipment may be eliminated.

As a matter of fact, the writer, his students, and members of the Instrumentation Laboratory had been thinking of self-contained systems for control, navigation, and guidance for fifteen years and, in a small way, carrying out construction of devices, with some flight tests in the writer's OX5 Robin airplane for thesis work, theoretical and experimental in the general field of guidance. The basic notion was that of extending the philosophy of time-keeping escapements as practical self-contained devices for indicating time, to the realization of self-contained references for geometrical orientations, and for translation into practical results by accurately carrying out commanded physical steps in angle and in linear displacement from the references.

Escapement for time keeping may be based on oscillations of a member that by properly balancing inertial reaction force with elastic restraint force, each cycle of an indefinitely long sequence may be made to occupy effectively the same period of time. Any cycle may be taken as the reference cycle, and the process of measuring time to any given instant is just that of counting the number of oscillations that have occurred since the initial point.

Geometrical indications follow the same pattern, with the complications introduced by the fact that our physical world includes three dimensions, each having a degree of rotational freedom and a degree of linear displacement. The realization of references by self-contained mechanizations must start with Newton's Law of Motion under which the application of force to a mass produces acceleration with respect to inertial space. Useful results for rotation may be realized by forming mass into symmetrical rotors for gyroscopic elements.

To serve the purposes of fire control, gyroscopic elements were elastically restrained and viscously damped to rapidly generate output angles with orders of magnitude in the 1- or 2-deg range. Navigation and guidance had to deal with flights of 5-10 h, with circles of erratic inaccuracy of no more than a few hundred feet and 1 mile or less terminal inaccuracies. Taking 1 min of arc between local verticals on the Earth's surface as representing 1 n. mi. and 900 min of arc approximating the 15 deg/h of Earth's rate of rotation on its axis, the buildup of one mile error in 10 h would mean a drift rate of about 0.1 of one thousandth of Earth's rate as an approximation required from gyroscopic units for providing satisfactory angular references within self-contained navigation systems. A circular inaccuracy of some 5 s of arc would be needed for defining terminal points.

It was obvious that the instruments applied to fire-control purposes had nothing to offer for navigation and guidance. It was known that available aircraft instruments were brought to specifications of about one Earth's rate (symbol Eru), while the needed performance was 0.1 of one thousandth of Earth's rate (symbol Meru). A survey of the situation for specific force receivers (commonly called accelerometers) showed that indications of resultants of gravity and acceleration required instruments with performance improvements of about 10,000 times beyond the behavior of commonly available devices. Few of the basic technologies required were available. The angular deviation receivers and specific force receivers did not exist, so they had to be developed. The writer and his colleagues felt very strongly that self-contained systems were possible and that with the existing motivation, useful results could be brought to realization in a few years.

The Febe System

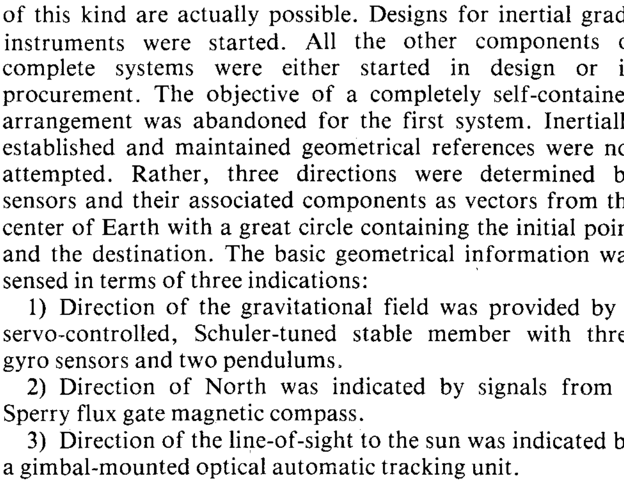
The theoretical and technological circumstances were extensively reviewed at the Laboratory, discussed with Colonel Davis, and considered by a group in the Armament Laboratory under Dr. J.E. Clemens. It was decided that the overall objective of achieving self-contained systems for navigation and guidance was definitely worthwhile, at least from the standpoint of determining whether or not equipment of this kind are actually possible. Designs for inertial grade instruments were started. All the other components of complete systems were either started in design or in procurement. The objective of a completely self-contained arrangement was abandoned for the first system. Inertially established and maintained geometrical references were not attempted. Rather, three directions were determined by sensors and their associated components as vectors from the center of Earth with a great circle containing the initial point and the destination. The basic geometrical information was sensed in terms of three indications:

1) Direction of the gravitational field was provided by a servo-controlled, Schuler-tuned stable member with three gyro sensors and two pendulums.

2) Direction of North was indicated by signals from a Sperry flux gate magnetic compass.

3) Direction of the line-of-sight to the sun was indicated by a gimbal-mounted optical automatic tracking unit.

These inputs were processed by computer components, including integration and the results applied to determination of position and control of aircraft movements toward desired targets over great circle paths programmed with respect to Earth. Figure 6 is a diagram showing relationships among the various geometrical concepts involved when great circle



TRIGONOMETRIC RELATIONSHIPS:

$$\cos(A_R)_D = \sin(\mathrm{lat})_D \sin(\mathrm{lat})_T + \cos(\mathrm{lat})_D \cos(\mathrm{lat})_T \cos(\mathrm{long})_{(D-T)}$$

$$\sin\left[(A_R)_D + (A_{AGC})_D\right] = \frac{\sin(\mathrm{lat})_T}{\cos(\mathrm{lat})_{(PA)TR}}$$

$$\sin(A_{AGC})_D = \frac{\sin(\mathrm{lat})_D}{\cos(\mathrm{lat})_{(PA)TR}}$$

Fig. 6 Relationships between programmed geocentric coordinate system and astronomical latitudes and longitudes of departure point and target.

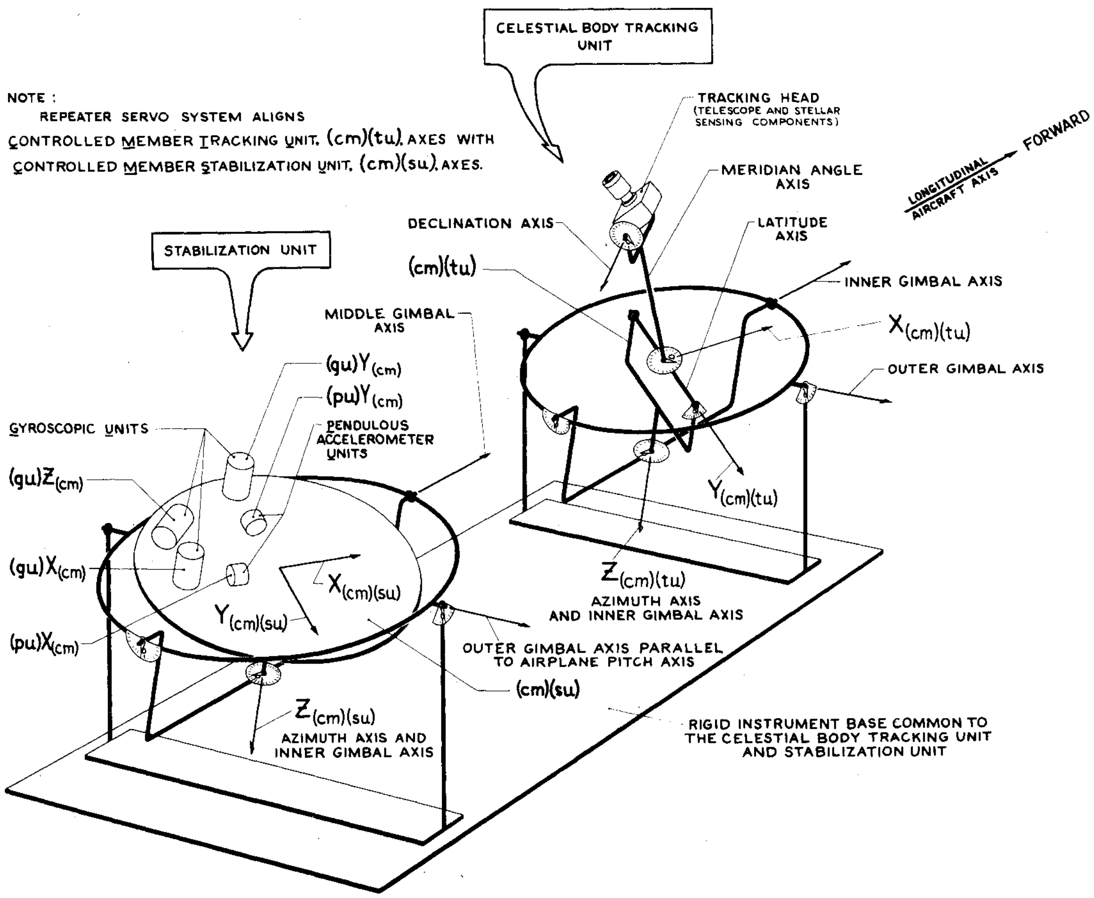


Fig. 7 Febe system—schematic diagram showing stabilization unit and celestial body tracking unit.

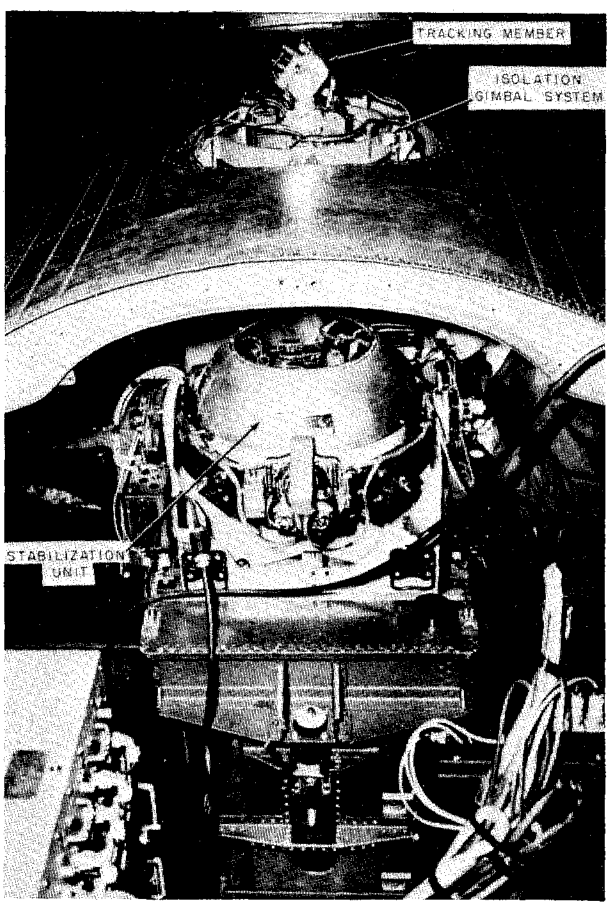
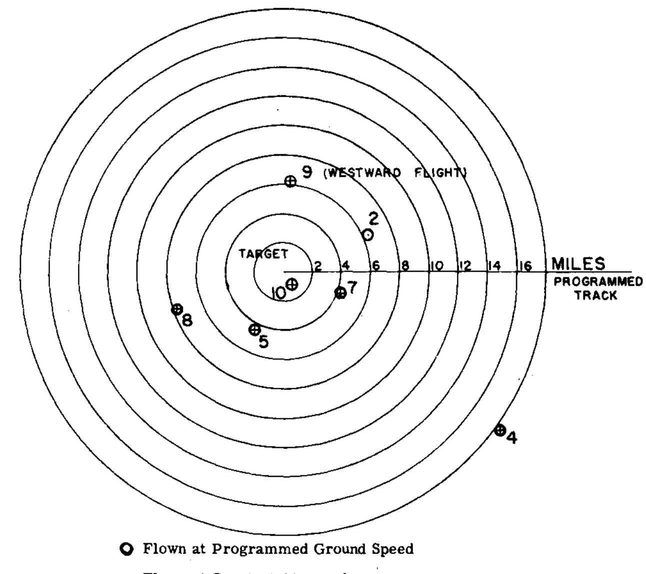


Fig. 8 Febe system—installation showing stabilization unit and celestial body tracking member.



Test nos. 2, 4, 5 and 7: Bedford, Mass. to Wright Field, Ohio (601 n.mi.)

Test no. 8: Bedford, Mass. to Alamagordo, N.Mex. (1737 n.mi.)

Test no. 9: Bedford, Mass. to Mansfield, Ohio and return (514 n.mi.)

Test no. 10: Bedford, Mass. to Canton, Ok. (904 n.mi.)

NOTE: The mean radial error at the destination for test nos. 2, 5, 7, 8, 9 and 10 as shown is five miles. Test no. 1 was not considered for extensive engineering evaluation. On this "shakedown" test sonic equipment malfunction was experienced and the actual track as determined by photographs from the aircraft was not recorded for the last two-thirds of the test distance due to an extensive cloud undercast. Test nos. 3 and 6 were not evaluated due to equipment failures. Test no. 4 was not included in the algebraic averaging because the terminal error departed abnormally from the mean error value.

Fig. 9 Febe system—summary of guidance errors at the destination for significant flight tests.

courses are used to program navigation between a point of departure and a selected target. The actual subsystems were made up of the electromagnetic mechanical computer components that were in common use during the 1940's.

Figure 7 is a schematic diagram showing the disposition of components within the stabilization unit and the celestial body tracking unit. The two units were gimballed to have angular freedom with respect to the base, by which they were both carried in the aircraft. The stabilization unit with three single-degree-of-freedom gyro units with input axes mutually at right angles, and two pendulum units with input axes perpendicular to each other in the plane of the unit effectively acted as a Schuler-tuned indicator of the vertical. The celestial body tracking unit automatically followed the line of sight to a single celestial body, and transferred the sensed direction through computer processing and signal transfers in a proper combination with the indicated direction of gravity generated by operation of the stabilization unit. The flux gate compass sensed the direction of indicated North, and sent the resulting signals through a computer to add the necessary third direction to the complex, and provide indications of navigational positions and guidance corrections for achieving desired results for aircraft missions.

Figure 8 is a photograph showing the installation of the Febe System on the structure of a B-29 aft pressurized compartment. The stabilization unit and the celestial body tracking unit are shown as separate components.

Flight Tests of the Febe System

The Febe System was designed, built, and installed in an Air Force B-29 aircraft so test flights could be systematically started and completed during 1949. Two programmed great circle courses were used for most of the tests. One, from Bedford, Mass. to Wright-Patterson Air Force Base, Dayton, Ohio, a distance of 601 n.mi., and a second from Bedford, Mass. to Holloman Air Force Base, Alamogordo, N.Mex., a distance of 1737 n.mi.

In all, 10 test flights were planned and carried out to achieve experimentally valid results from six of the missions. Results are summarized by the points plotted on Fig. 9. Of the tests reported, four were not considered for the reasons noted on the figure. The remaining six missions had a mean error of 5 n.mi. It was noted that a loss of sun tracking information, due to cloud cover occurring for periods of some tens of minutes after system operation had been started, did not disturb results. This performance came from the actions of integration included in computer operations.

The flight test results, in general, showed erratic oscillations of indications superposed on slower swings of inaccuracy as missions continued. There were a considerable number of intervals with errors of one or two miles. It was not possible to predict when these times of good results would occur. Febe was certainly not a self-contained inertial system because it depended on tracking a celestial body, the Earth's magnetic field, and the Earth's gravitational field. On the other hand, the mechanization used suggested that if sensors of the required performance could be achieved, automatic systems for navigation and guidance could probably be realized.

Sensor Performance

Sensors for angular deviations and receivers for specific force that were available for use in Febe did not have the required performance. This meant that creative thought, design efforts, and technological work had to be expended if the goal of completely self-contained inertial control, navigation, and guidance systems was to be accepted as realistic. Febe system tests had demonstrated conclusively that integration is an essential computer operation in dealing with sensed information, so that strong consideration was needed for incorporating high-quality integration as an operating function of sensors for angular deviations and for specific force to be used as basic components of inertial systems.

The angular deviation receivers that were applied as sensors in the Febe system were based on Marine gyro compasses and on the gyroscopic elements of unstabilized antiaircraft fire-control systems. The spinning rotors were carried by ball bearings, and the supporting gimbals were suspended either on springs or by means of ball bearings. It was realized that sensors should be able to resolve fractional arc seconds (corresponding to a few feet distance on the Earth's surface). Ball bearings for gimbals with rotors of few million $g \cdot cm^2/s$ angular momentum, certainly did not allow performance of this kind. Working toward resolution of this difficulty, the rotor and its gimbal were enclosed in an hermetically sealed cylinder and floated in a very viscous fluid. The viscous shear drag on rotation of the cylinder about an axis at right angles to the spin axis matched the output torque of the rotor as it responded with angular velocity to an input torque. By careful temperature control, to achieve flotation of well-balanced floats, considerable improvements in angular deviation reception were realized, but further advances had to be accepted as goals for future endeavors.

Electromagnetic torque generators to apply their outputs at right angles to angular momentum vectors were carefully designed to provide means for commanding the angular velocity inputs required for the operations of integration.

The specific force receivers presented problems generally similar to those associated with angular deviation sensors, and were attacked with means that had been useful for angular deviation receivers. Spinning rotors were, of course, replaced by unbalanced structures contained within floated cylinders carrying signal generators and torque generators. During the first years of instruments following this pattern, integration was carried out beyond the sensor. Later, gyroscopic integration was incorporated within the instrument itself—the pendulous, integrating gyro accelerometer (PIGA).

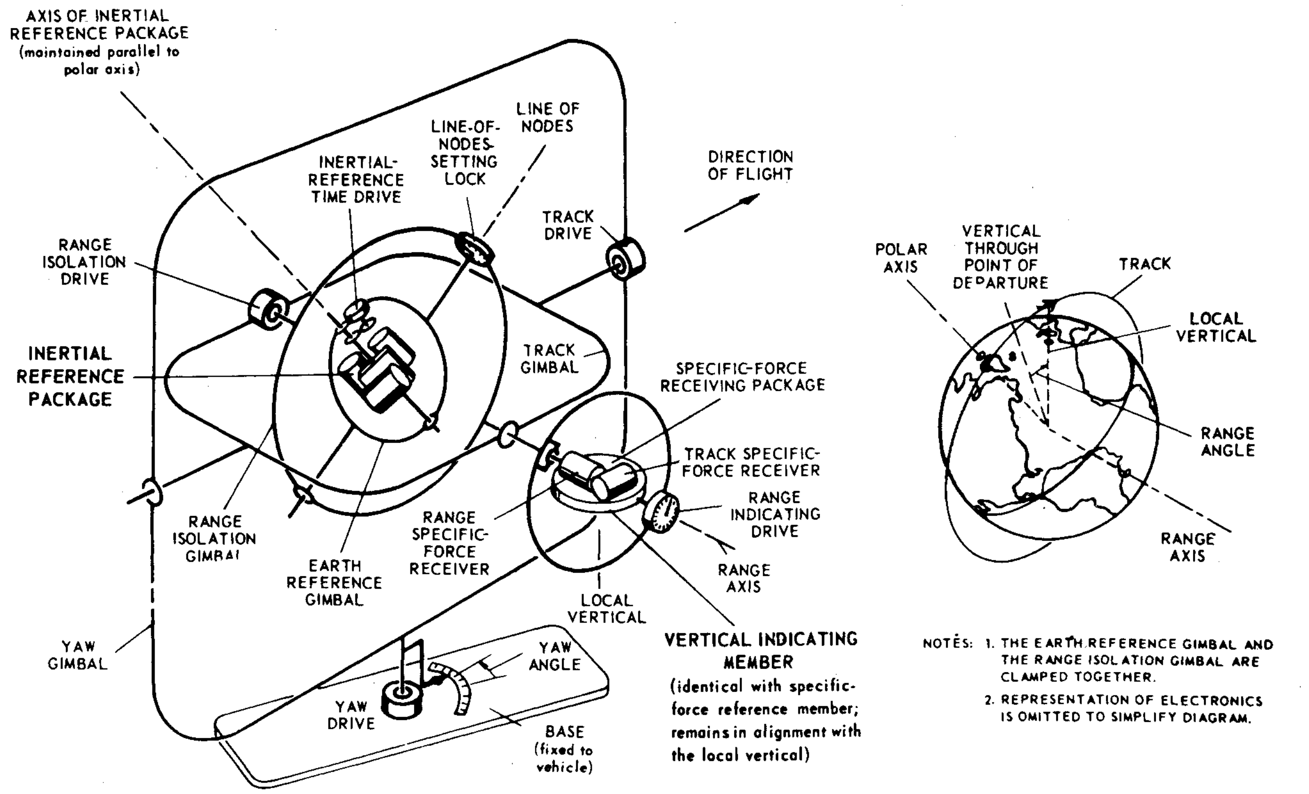
All Inertial System—Spire

Results from Febe left unanswered the basic question of the usefulness that a completely inertial system could provide, and, in fact, did not clearly demonstrate the feasibility of self-contained systems. For the purpose of removing this uncertainty, the Air Force assigned to the Instrumentation Laboratory a continuation project to Febe whose object was designing, building, and testing a control, navigation and guidance system that would depend only upon inertial and gravitational inputs except, perhaps, for the purpose of initial settings.

New sensors for angular deviation and for specific force were combined into an arrangement of servo-driven gimbals and instrument-supporting members. A triad of angular deviation receivers were mounted, as suggested by Fig. 10, on an inertial space reference member with their input axes mutually at right angles and connected to servodrives of the reference member so that one of its axes was maintained in alignment with the Earth's axis of rotation. The two other perpendicular-to-Earth's-axis servodrives were initialized and connected so the inertial reference member remained nonrotating with respect to inertial space. The geometrical gimbal arrangement involved in supporting the inertial space reference packages is shown in the diagram of Fig. 10. The inertial space reference is carried by the Earth reference gimbal through the indicated polar axis. The electrically driven (from controlled frequency) inertial reference time motor causes the Earth reference gimbal to rotate with the Earth, while the inertial reference package holds its orientation with respect to inertial space.

The great circle track over which the system operates is determined by turning the range isolation gimbal about the line of nodes and locking so it always has the orientation of the great circle chosen for the track of the airplane involved.

One pendulum was mounted with its axis-of-freedom in the plane of the programmed great circle course. A second



Inertial guidance system that is based on an earth reference gimbal with a time drive from an inertial reference package and on a Schuler-tuned specific-force receiving package.

Fig. 10 Mechanofunctional diagram for the Spire system (about 1950).

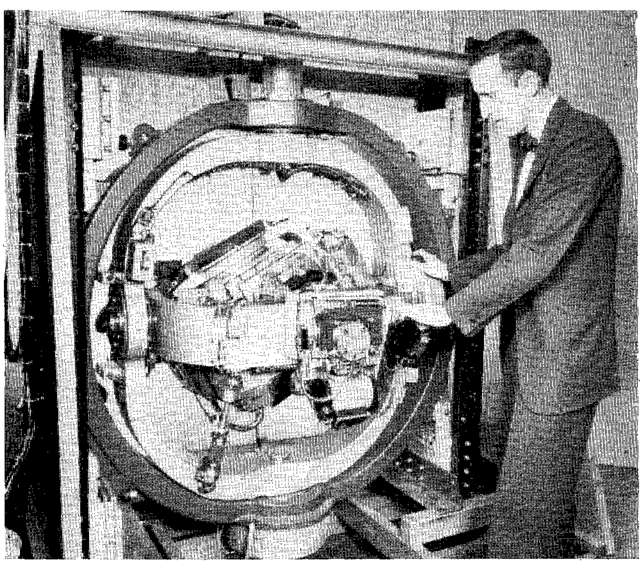


Fig. 11 Spire system gimbals.

pendulum was mounted with its axis at right angles to the programmed great circle. Computer elements were included, with each pendulum Schuler-tuned so that signals for the distance cross track, and for position along the programmed track, were generated as signal outputs. These outputs gave navigational information on cross programmed track inaccuracies and on progress toward the target.

Figure 11 shows the appearance of the Spire system with the pendulum units forward in the horizontal plane. The Earth's rotation rate drive motor appears at the end of the tipped down inertial member just behind the pendulum units.

During the years of 1953 through 1955 a number of flights to Holloman Air Force Base and to Los Angeles from Bedford, Mass. were made with Spire. Figure 12 summarizes results from the trials made during 1955. The approximately 9

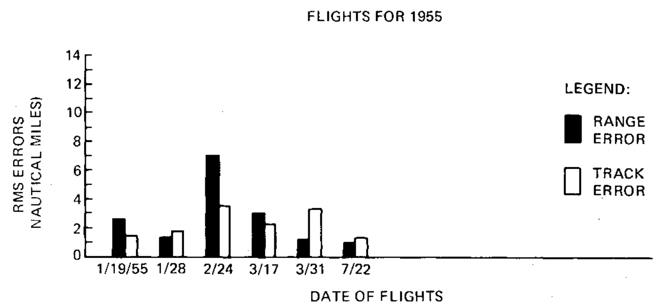


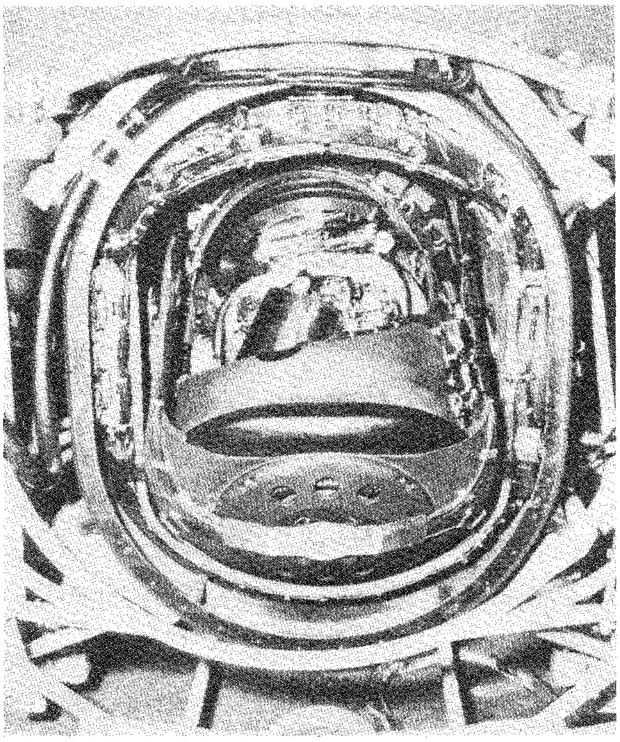
Fig. 12 Summary of results of Spire flights (1955).

h flights were accompanied by inaccuracy components in the range of two miles to less than four miles during trips of about 1800 miles. Thus, error buildups at rates of between one and two miles per thousand miles of travel were demonstrated by a system that was, beyond doubt, completely self-contained because it included only sensors depending on inertial principles for their operation.

Results from system tests were limited principally by the performance of sensors. For this reason continuous efforts were devoted toward improvements in angular deviation receivers and their ability to accept and process command signals. Similar efforts were applied to specific force receivers, with particular attention to incorporation of integration within the instruments themselves. Improvements in performance were always taken as primary objectives, with smaller size, reduced weight, and smaller costs also strongly considered.

A Local-Level System—Spire Jr.

Spire Jr. was a self-contained all-inertial system undertaken to continue the developments started in Febe and Spire with weight and size reduced by more than a factor of two, better components, and improved performance. The so-called local-level design principle was applied with angular deviation receivers and specific force receivers mounted rigidly on the



same servo-controlled platform. For the first time, specific force integrating receivers using gyro rotors as internal components were applied as self-contained integrators.

The angular deviation receivers with input axes mutually at right angles provided deviation signals that, supplied to the platform servos, accepted initial settings along North, along East, and along the vertical to stabilize the specific force receiver input axes along these directions.

Settings before the start of movement are initial latitude, initial longitude, and North, with the system itself establishing the indicated vertical and indicated North after a period of initialization by being allowed to remain with the base stationary on the Earth's surface.

Figure 13 is a photograph of the Spire Jr. system on a hoist for lifting it into place on a C97 airplane, and, at the left, the working components with the temperature shield removed.

Figure 14 is a photograph of Eric Severaid and Stark Draper observing the actions of Spire Jr. gimbals during a flight from Bedford, Mass. to Los Angeles, Calif. on March 7, 1958.

Flight test results with Spire Jr. remained erratic. Some tracks showed inaccuracies of two miles or less in range and lateral direction while with other flights gave results at considerably poorer levels. Figure 15 is an example of an

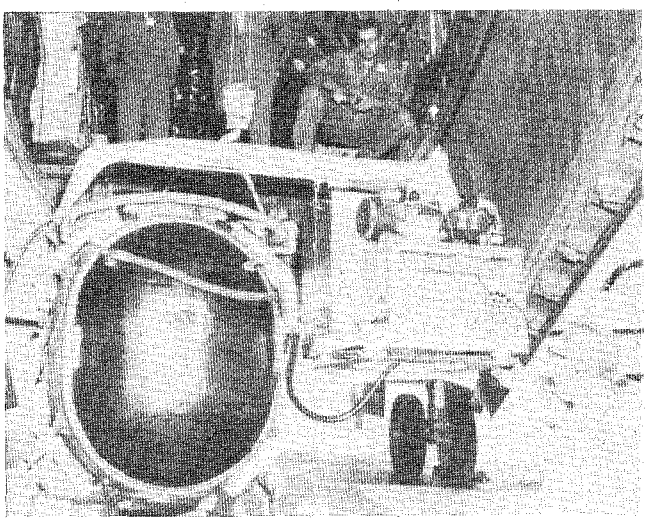


Fig. 13 Project Spire Jr.—gimbals with electronics.

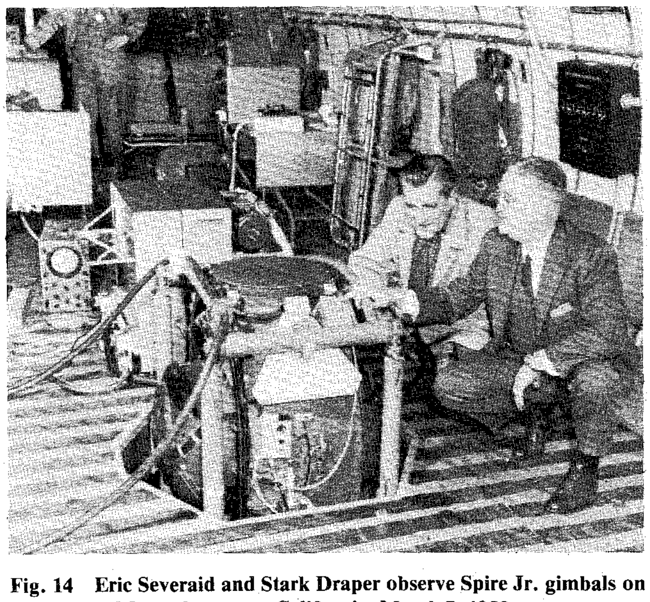


Fig. 14 Eric Severaid and Stark Draper observe Spire Jr. gimbals on flight from Massachusetts to California, March 7, 1958.

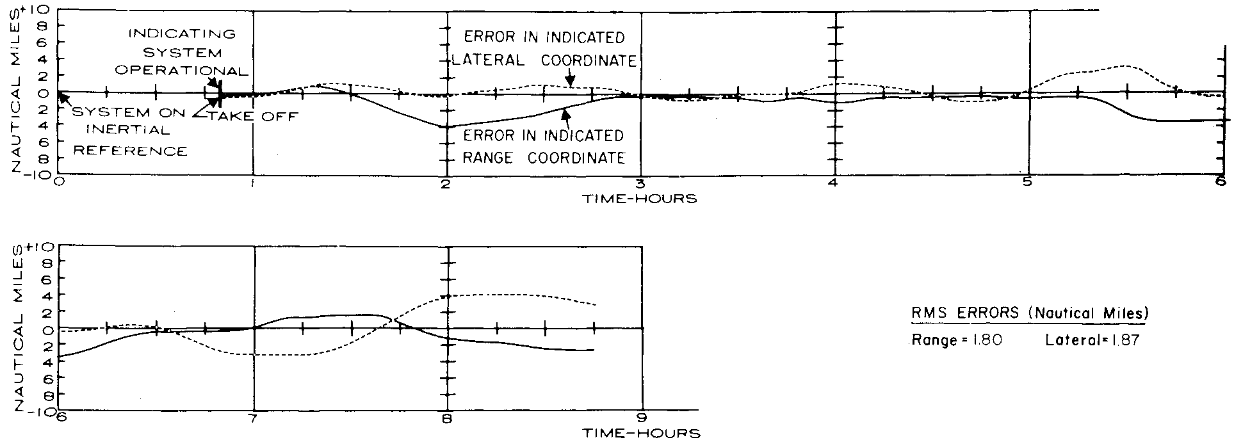


Fig. 15 Errors in purely inertial flight of Spire Jr. system.

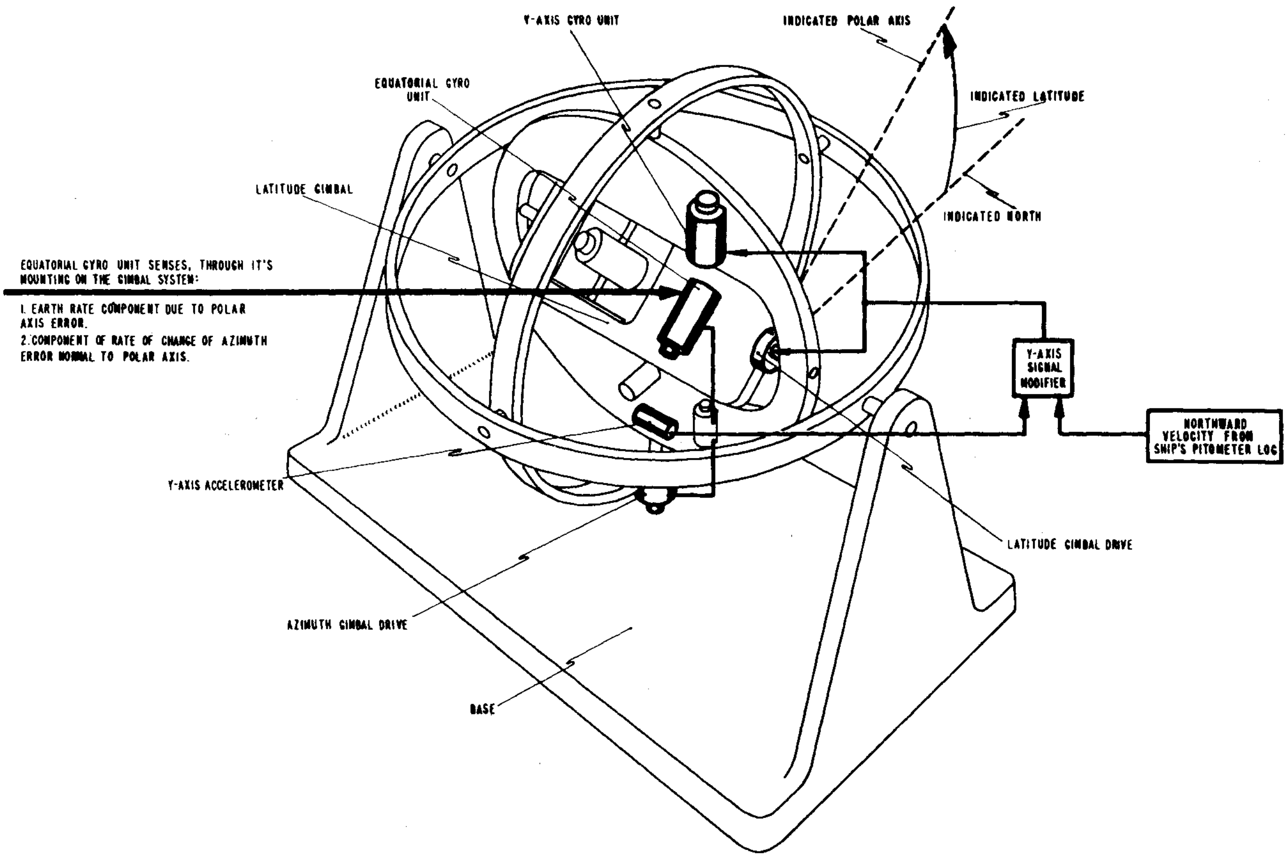


Fig. 16 Simplified drawing of SINS latitude and azimuth indication loop.

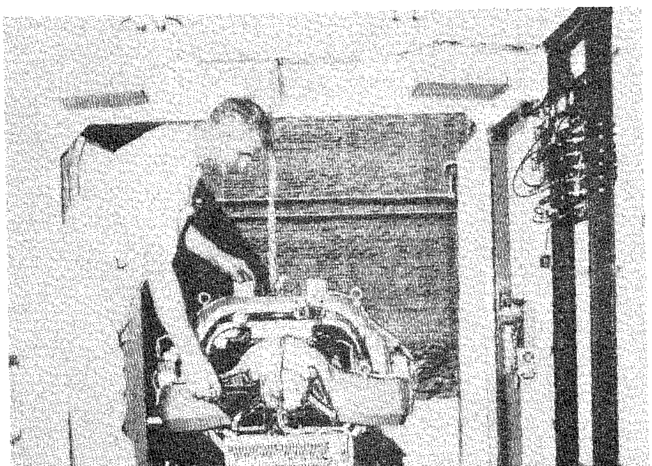


Fig. 17 Front view of gimbal system—SINS.

approximately 9 h flight (about 1740 n.mi.) with root mean square errors of about 1.8 n.mi. in both directions.

New sensors for angle and for specific force, which had been under design for some years, were showing promise, but had not yet reached stages of readiness for use in operational systems. Some new instruments had been replaced between the Spire and Spire Jr. system embodiments, but enough of the old pattern remained to prevent the desired order of magnitude improvements beyond those achieved in tests of Spire. In addition, requirements for guidance of long-range missiles introduced new performance characteristics to be provided during short times of operation. Because of this change in emphasis, the attention of the Laboratory was shifted toward inertial systems for missiles and away from aircraft equipment.

Submarine Inertial Navigation Systems

Results from inertial systems for guidance of aircraft stimulated interest toward self-contained navigation equipment for use by ships and on submarines. Proposals were generated by the Laboratory to make use of the concepts and instruments developed by the Laboratory for Febe and Spire to realize units combining vertical indications and azimuth indications to give the results ordinarily derived from stable element-gyroscopic compass combinations and to be called MAST.

MAST was tested at sea in 1954 against a quadrilateral horizon tracker and successfully proved a superior capability over standard equipment for ship-to-shore and ship-to-ship bombardment reference. By addition of navigational functions, such arrangements would become the submarine inertial navigation system (SINS) that was, in its first embodiments, built up from instruments that were originally designed and constructed for the Air Force to be applied in the Spire inertial navigation system for aircraft. The concepts and requirements involved in submarine inertial navigation were discussed with officers in charge at Wright Field who gave permission for applying units belonging to the Air Force, which had not been required for Spire, to the purposes of SINS development for the Navy.

The first arrangement for SINS used angular deviation receiving gyro units, accelerometers, servo-driven gimbals, and dynamic interconnections among indicated vertical, indicated polar axis, indicated longitude change, and indicated azimuth to generate outputs of indicated ground speed, indicated distance traveled, indicated roll, and indicated pitch. Figure 16 suggests the arrangements of sensors and gimbals that were used for the first working models of SINS. Figure 17 shows a front view of the SINS gimbal system mounted in the mobile test van.

Operating tests of SINS were carried out with the system stationary in the laboratory, as well as moving in a test van. A number of runs were made over roads of the eastern part of the U.S., and the van was carried over various courses by a

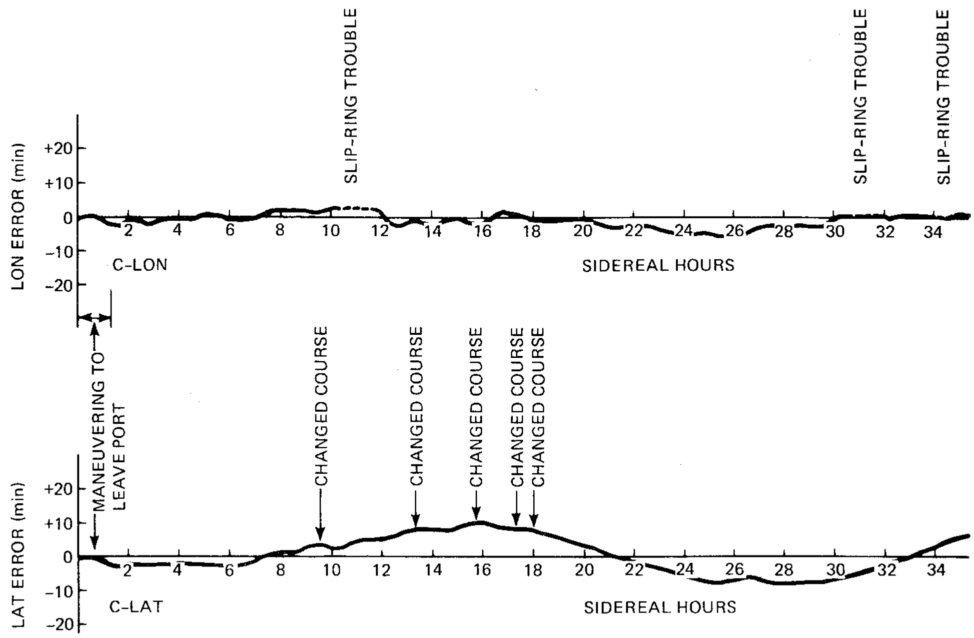


Fig. 18 Error plot for SINS system (Havana, Cuba to Norfolk, Va., June 27-28, 1955).

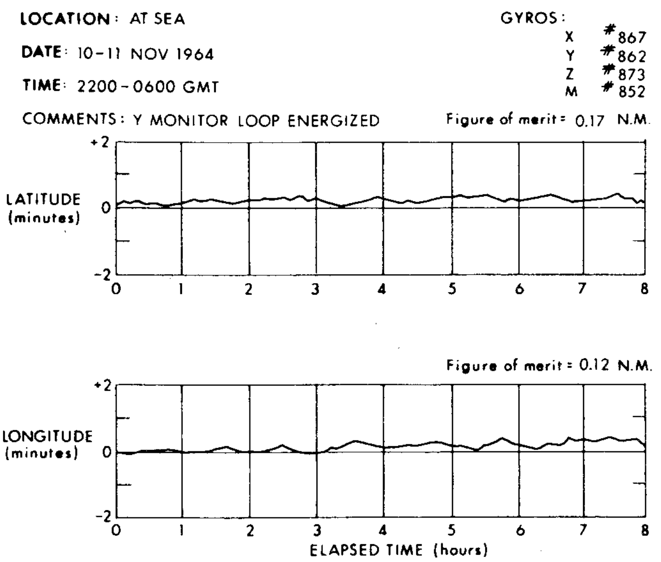


Fig. 19 TP-21 results—run 1.

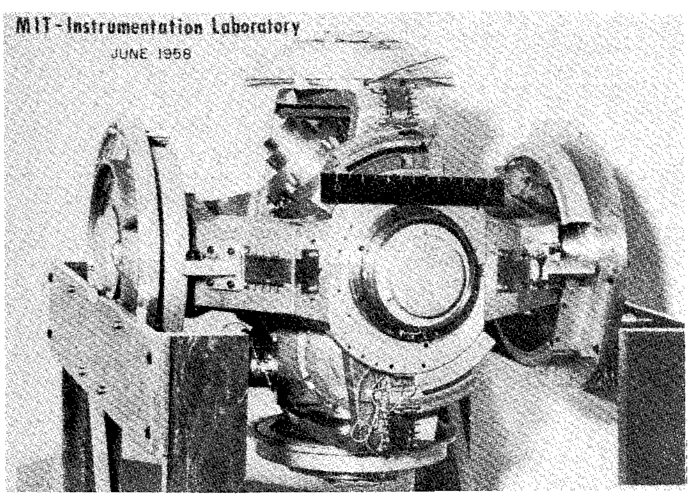


Fig. 20 Assembly, guidance gimbals (viewed along middle axis).

ship crossing the Atlantic and moving north and south along the East Coast of the U.S.

The results, in terms of latitude errors and longitude errors, were erratic and tended to have mean values in the vicinity of 5 n.mi. Figure 18, which gives error results during 34 h of a trip at sea, represents good performance from the first model of SINS.

SINS Mark IV

When work on SINS was started by the Instrumentation Laboratory during the first years of the 1950's, it was clearly recognized that the essential instruments available were not of highest quality, and that the theory associated with the problems involved had to be developed from fundamentals. It was also necessary to show that the ideas of inertial navigation for Naval purposes had to be demonstrated by operational results if they were to deserve support. For this reason, the first system was formed with existing components from which existing performance had to be accepted, with the hope of results good enough to justify further efforts.

Theoretical studies by the Laboratory, and test results during the mid-1950's, suggested that circuit refinements and the use of angular deviation sensors and specific force receivers of improved performance would very probably allow the development of operationally useful SINS. This information was discussed with interested agencies of the Navy with the result that the Laboratory was given support to design, construct, and test a new inertial navigation system to be called SINS Mark IV.

The SINS Mark IV information processing interconnections were based on functional arrangements used during tests of the first SINS model. Mechanical design features of SINS Mark IV generally were those using four gimbals, three gyro units, and three specific force receivers.

The SINS Mark IV program was started at the Laboratory during 1960, and sea tests of SINS Mark IV Mod 2 on board the U.S.S. Compass Island, and dockside, were carried out during October through December of 1964.

Figure 19 shows latitude and longitude error plots during an 8 h run at sea for SINS Mark IV Mod 2. The results show average errors of less than 0.2 min of arc in both directions for the period of the tests. A considerable number of other runs showed similar performance, a level that would have been generally satisfactory for meeting requirements. As it happened, the Navy decided to use already in-production and installed-in-operational-ships-equipment instead of continuing efforts to develop the SINS Mark IV into a production item. The ideas and technology involved remain available for future advances if they are needed for improved inertial navigation systems of the future. From the standpoint of projects for the laboratory, work on SINS was discontinued by the end of the 1960's.

Inertial Control and Guidance for Ballistic Missiles—Thor and Titan

During the 1940's it was realized that inertial control and guidance technology would be particularly useful in providing accurate information for carrying out mission paths to desired targets by the operation of self-contained systems. Accuracy tolerances had to be less than those expected in, for example, ship navigation, but the travel times involved would be short, tens of minutes, instead of the hours and days associated with usual voyages by sea or in the air. For land-based launchings, initial settings on stationary positions, would be as perfect as knowledge of the Earth's gravitational field would permit. In addition, once the launching accelerations, high compared to 1 g, were completed, relatively ideal specific force situations giving inputs formed primarily by inertial reaction effects would simplify the computing of guidance output indications.

These considerations meant that the mechanization of inertial guidance systems for ballistic missiles could follow a three gimbal, three gyro, and three specific force receiver pattern that had been introduced in the Spire Jr. At the beginning of the 1950's, computing systems still followed the mechanical component, motor, and analog electronics pattern that had been generally used during the 1940's. However, it was realized that with the emergence of solid state electronics and digital computing techniques, inertial systems with higher performance, of smaller sizes and lowered costs, would be very likely to emerge within a short time.

During the first half of the 1950's significant improvements evidently were just ahead for the technology of angular deviation sensors, specific force sensors, other components, and computer arrangements. Ballistic missiles were, by definition, not going to carry human pilots, and the need for high accuracy at thousand mile ranges was obvious. The combination of these and other factors attracted attention from a number of graduate students who were looking for thesis topics during the years at the end of the 1940's and the beginning of the 1950's. During these years, the U.S. was still using large aircraft as a primary means for implementing strategic defense. This meant that subjects associated with ballistic missiles were not subject to the rules of security, so there were no objections to students selecting ballistic missile guidance for thesis subjects. Several theses were carried out in this general field. Before these studies were completed, the U.S. introduced security for the general field so that one student, for example, could no longer have access to material he had written.

The first opportunity for the Laboratory to work on self-contained guidance for a ballistic missile came toward the middle of the 1950's, when it conceived, designed, and cooperated in building the inertial guidance system for the Thor intermediate range ballistic missile. The sensor and computer technology applied in this system was generally that of the early 1950's. The geared, motordriven PIGA arrived in time to be included.

Flight tests of the Thor missile showed that the inertial guidance system performed well and, in particular, was adapted for salvo firing of several missiles, a mode of operation that had been found unsatisfactory when radio guidance was used under the same circumstances. After results were available from the Thor tests, only inertial guidance systems were used for purposes of intermediate and long-range ballistic missile guidance.

After the Thor inertial guidance system had proved its capabilities, the Air Force arranged for the Laboratory to engineer inertial guidance for the Titan intercontinental ballistic missile. This system was designed by the Laboratory. It was placed in production by the A.C. Spark Plug Division of the General Motors Corporation. Installations and tests in operational missiles were carried out during the latter part of the 1950's.

The Laboratory also assisted with Minuteman guidance by providing gyroscopic specific force integrating receivers as components of already designed inertial systems. These sensors continue to generate satisfactory results.

Submarine Launched Ballistic Missiles—Polaris

About the middle of the 1950's, the Navy conducted studies that showed the desirability of building a fleet of submarines for underwater carrying and launching of intermediate range ballistic missiles. After discussions with the Laboratory on the engineering and technological problems associated with guidance for this mode of operation, which was given the name Polaris, a contract was awarded to the Laboratory by the Navy for the design and construction of operationally satisfactory inertial guidance systems to be ready during the first years of the 1960's.

It was decided to base the Polaris inertial guidance system on the now very familiar three gimbal, three gyro, three specific force sensor pattern, and for the first time to design and produce digital computers for guidance systems, which were reliable, light in weight, and suitable for ballistic missiles.

New sensors of reduced size and weight with improved performance gave Polaris guidance a strong position during the mid-1950's, among the new endeavors of the Laboratory. Figure 20 is a photograph of the guidance gimbals with the outer spherical shell removed. The diameter is about 16 in., and the weight for the mechanical parts about 69 lb. The weight and size have changed by a factor of approximately 40 as compared with the 2800 lb of the Febe system gimbals.

Inertial Guidance in Space—Apollo Command Module and Lunar Landing Module

In 1960, when President Kennedy inspired the U.S. to start the Apollo Project for transporting men to the moon and returning them safely to Earth during the next ten years, the Laboratory presented its capabilities for imagining, designing, building models, and playing the roles necessary for realizing satisfactory results from operational systems to control and guide all phases of the missions to the moon and return to Earth. Discussions with Administrator James E. Webb and his associates led to a contract from NASA to the Laboratory that amounted to an assignment for the conception, the engineering, the construction, the testing, and consultation during operation of the control, navigation, and guidance system for Apollo.

The principles applied for realization of the Apollo guidance system were those of using gyro sensor and specific force receivers to acquire basic information for establishing geometrical reference orientation for the inner package of a three gimbal arrangement similar to those already suggested by various figures. This arrangement provided inertial references for linear motion signals from the specific force receivers whose signals were integrated once for linear velocity information and again for linear displacements.

The inertial references for orientation were provided by initial angular settings of the gyro units and establishment of initial readings from the specific force receivers. An optical unit including a telescope and sextant with mirror systems servo-connected to the gyro angle sensors, and a manual control system was provided for aligning the inertial reference member with respect to directions from selected planetary or celestial bodies.

The overall mechanism arrangements, including computers, keyboard for entering commands, information display system, and provisions for manual control of the vehicle followed principles already discussed. Because of severe restrictions on weight and size, the equipment had to be designed for minimum bulk so that duplication of components was not feasible. This required achieving unflawed reliability from the systems as they were installed. That this objective for control and guidance was effectively achieved during all of the Apollo missions is a fact of history well known to all who followed progress in the first round-trip flights to the moon by human beings.

Inertial Guidance—Present and Future

Inertial principles certainly have provided the background for various developments that have received much attention from the Laboratory during the past 30 years. The general background of physical principles, and most active areas, has been described briefly in this paper. Many able scientists, mathematicians, designers, machinists, and test engineers have helped greatly with various advances in the theory and technology involved. In all fairness to these individuals, their names and contributions should be specifically recognized, but this paper does not provide the space required for this purpose.

Starting during the 1950's, a number of manufacturers became interested in producing and supplying inertial navigation systems for military and commercial users. In general, each manufacturer worked toward designing its own instruments for receiving angular deviations and specific force with either single-degree-of-freedom or two-degree-of-freedom operation. Laser light interference over closed paths has been applied to the sensing of angular deviations. Systems have been mechanically simplified by eliminating gimbals and substituting complex computer resolutions of signals from "strapped down" sensors. Electrostatic bearings to carry rotors have been used in generating angular momentum for two-degree-of-freedom systems.

These and other ideas have been applied to reduce size, weight, and cost of inertial guidance systems while maintaining adequate performance. Much progress has already been achieved, and it is certain that improvements will continue to appear in the field of inertial technology, which has already demonstrated its value for purposes of control, navigation, and guidance.

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