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Developments in the Field of Automatic Guidance and Control of Rockets

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Introduction

TODAY'S definitions are that "navigation" determines the vehicle's state for initial conditions as well as during flight; that "guidance" selects the maneuvering sequence to get from the instantaneous state to a required state; and that "control" executes the maneuvers called for by guidance. These definitions did not exist in the early development years. Navigation corrections were usually taken care of by flight time deviations, which were included in simple guidance functions. Fixed programs, such as flight-tilt programs and roll programs, to rotate the vehicle into its required flight plane have been considered control functions rather than guidance functions and have been exceptions to these definitions.

The development of liquid fuel and oxidizer engines made it necessary to stabilize the rockets and to provide attitude control. This included a pitch tilt program, because these rockets with their low initial acceleration had to be launched in a vertical direction. In order to aim at a target the next requirements were to constrain the rocket to a predetermined flight plane and to provide a well-defined velocity vector at propulsion cutoff of the rocket engine.

Related but less sophisticated requirements had existed for ship and airplane control. Inertial sensors, such as gimbal suspended gyroscopes and rate gyros, had been used for these systems. In 1912 Prof. Max Schuler had designed a three-gyro, three-degree-of-freedom gyro compass for ship navigation, and two-degree-of-freedom gyros were used later as airplane orientation sensors. Schemes and components derived from these systems were applied to the first rocket attitude control systems. A somewhat more difficult task was to obtain guidance sensors, which did not exist yet. For many years inertial sensors were in competition with radio guidance devices, the latter being more promising in the early years to fulfill accuracy requirements.

Accelerometers had to be used as inertial sensors for guidance. Velocity and displacement, as the information required, had to be derived; the first with normalized accuracy demands of at least 10^{-3}-10^{-4}. The derived signals had to be obtained by the only available method, which was mechanical integration. In contrast to the inertial sensors, the radio sensors measured lateral displacements and forward (radial) velocities directly. Since the guidance systems had to be used for missiles an inertial system was preferable because it is self-contained and not subjected to external interference as radio guidance systems are. The radio guidance system became available and operational first. It yielded an accuracy in range that was about 10% better than the inertial system introduced afterward. Only a few test flights with a pure inertial system took place before the end of World War II. The cross-range impact errors were approximately twice as large as with the radio guidance system. Thus for some time it

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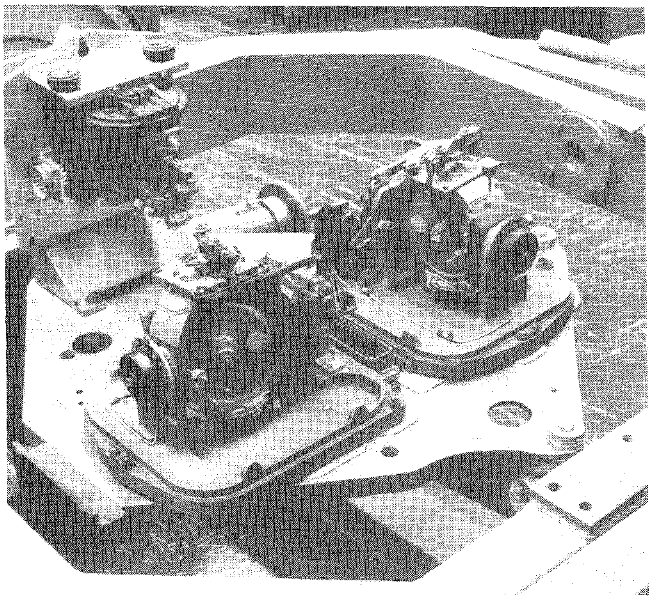

Fig. 1 LEV-3 system.

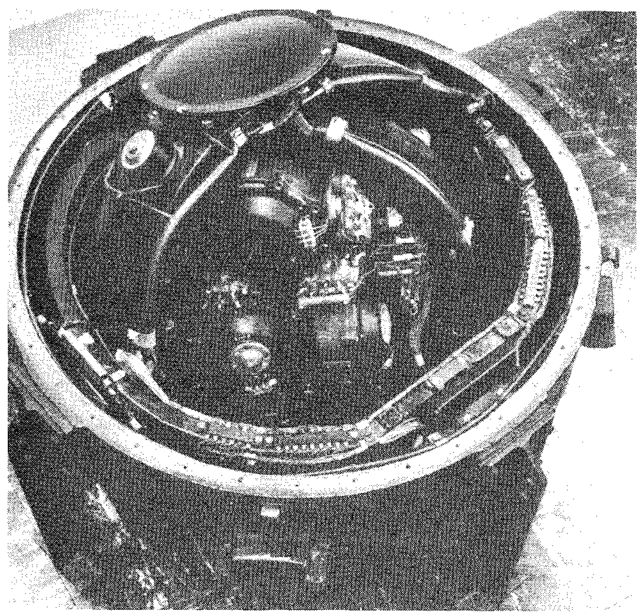

Fig. 2 Stabilized platform, SG-66.

was considered whether a combination of both systems would lead to an optimum solution. The inertial system would provide smooth information signals and the radio data would correct for the inertial sensors' inaccuracy. Because of its complexity and weight penalty such a combined radio-inertial system never became attractive; it was abandoned when, in the middle 1950's, the inertial system proved its feasibility and achieved the required accuracies.[1]

System Developments Until the End of World War II

Dr. Robert H. Goddard used the gyro in the Spring of 1935 to provide attitude control; he used the attitude signals to operate the jet vanes of his experimental rockets. A tilt program was incorporated to turn the rocket from its vertical launch direction to the flight direction.

Two years later three body-fixed rate gyros and a two-axis gyro-stabilized platform with spring-mass accelerometers and integrators were launched in the German experimental rocket A-3.[2] Four rockets were launched, but roll control by the rate gyro sensor signal only was found to be inadequate. The roll torques of the rocket were higher than expected and caused large roll deviations with a consequent tumbling of the stabilized platform.

In the following years defense requirements emphasized the demand for accurate and reliable guidance and control systems. The imminent need required short development times. This fact limited the component selection insofar as feasible components had to be used which were advanced in their design or were in manufacturing for other applications. Thus the system design was determined to a considerable extent by the state of the art of useful components. One very limiting factor for the control system was the servoactuator. A hydraulic servomotor with an integral pump, as was used in aircraft, had to be selected. Since its output power was insufficient, its hydraulic pressure had to be increased several times when the requirements were updated; still its output remained marginal, and it could not be used for position control. Its nonlinear and low velocity output at high torque loads became a most demanding factor in the design of the control loop. When new sensors or devices had to be developed, several approaches were pursued. Universities and research laboratories became engaged in these competitive developments. Thus the system design had to be very flexible to provide for the integration of the most successful components.

Two competing system approaches were taken for guidance and attitude control of the experimental rocket A-5, which served as a development and test missile for the navigation, guidance and control (NGC) system, of the subsequent strategic A-4, renamed V-2 in operation.[3] One approach was a body-fixed system, designated LEV-3, that used two, two-degree-of-freedom gyros. One was the "Vertikant," providing yaw and roll information; the other was named "Horizont," possessing a time-controlled rotation of its pickup potentiometer base and furnishing the pitch information. A pendulous integrating gyro accelerometer and the two gyros were mounted to an adjustable missile fixed plate. Figure 1 displays this system. In the other approach, a three-gimbal platform was stabilized by three single-degree-of-freedom gyros. The inner gimbal contained a large bearing for the implementation of the pitch-tilt program. The platform carried accelerometers for cross-range guidance and propulsion cutoff. Figure 2 is a picture of this stabilized platform, that carried the designation SG-66. The body-fixed LEV-3 system became operational because of its shorter development time. The stabilized platform SG-66 was successfully flight tested and became the first inertial measuring unit for a guided missile control system.

Dr. H. Hoelzer, O. Hirschler, and O. Hoberg developed the competitive radio guidance system; it was designed to provide the information for cross-range guidance and propulsion cutoff. For determining cross-range deviations, a transmitter was located in the flight plane 10-16 km behind the missile launch pad. The 50 MHz transmitter alternately fed two horizontal dipole antennas (Fig. 3) positioned symmetrically 100 m from the flight plane with a mutual phase shift of 90 deg or one quarter of a wavelength ($\lambda/4$). Thus symmetrical radiation patterns were achieved. The transmitter signal was modulated with 5 and 7 kHz, respectively, synchronously with the 50 Hz switching. The receiver in the missile sensed a modulation voltage proportional to the angle ϵ, which served as a measure for the cross-range displacement of the missile. The two modulation frequencies permitted the definition of the sign of ϵ. After the evaluated signal passed through a stabilizing electronic network, which added a derivative as well as a small integration term, it was superimposed on the yaw information for cross-range guidance. The Doppler effect was used to determine the missile velocity for propulsion cutoff. The frequency of the ground transmitter was close to 37 MHz and could be selected for interference reasons. The frequency received onboard was doubled and

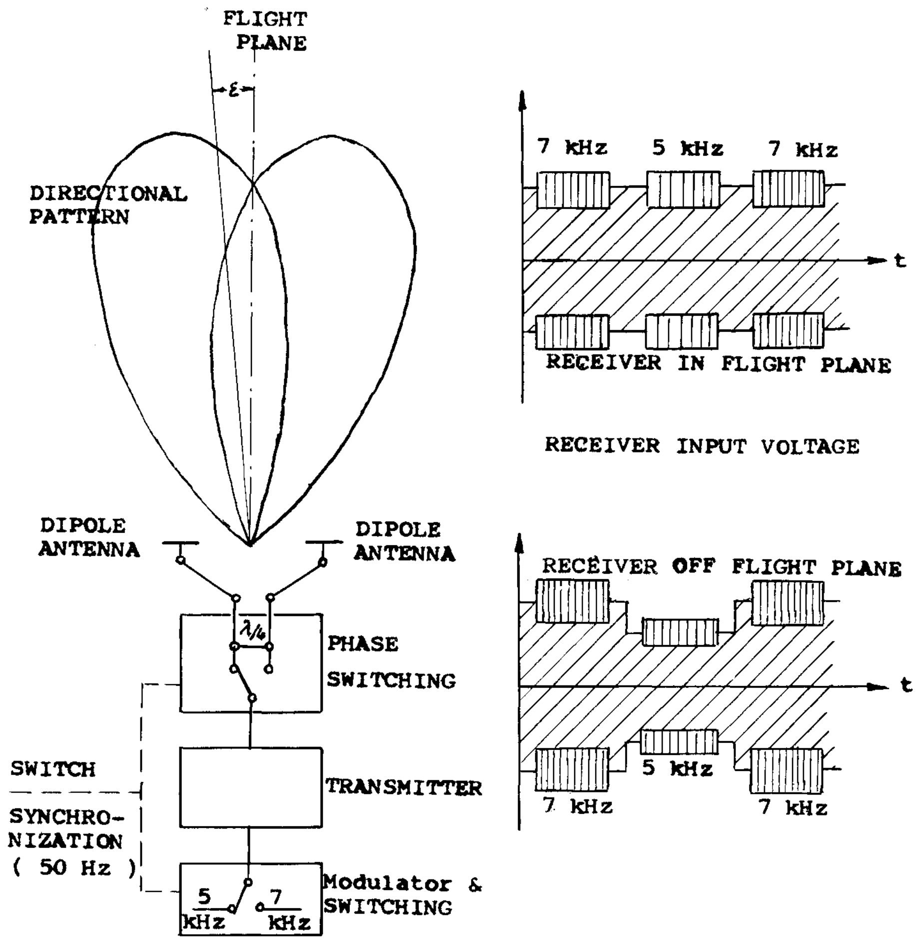

Fig. 3 Transmitting system for V-2 cross-range radio guidance.

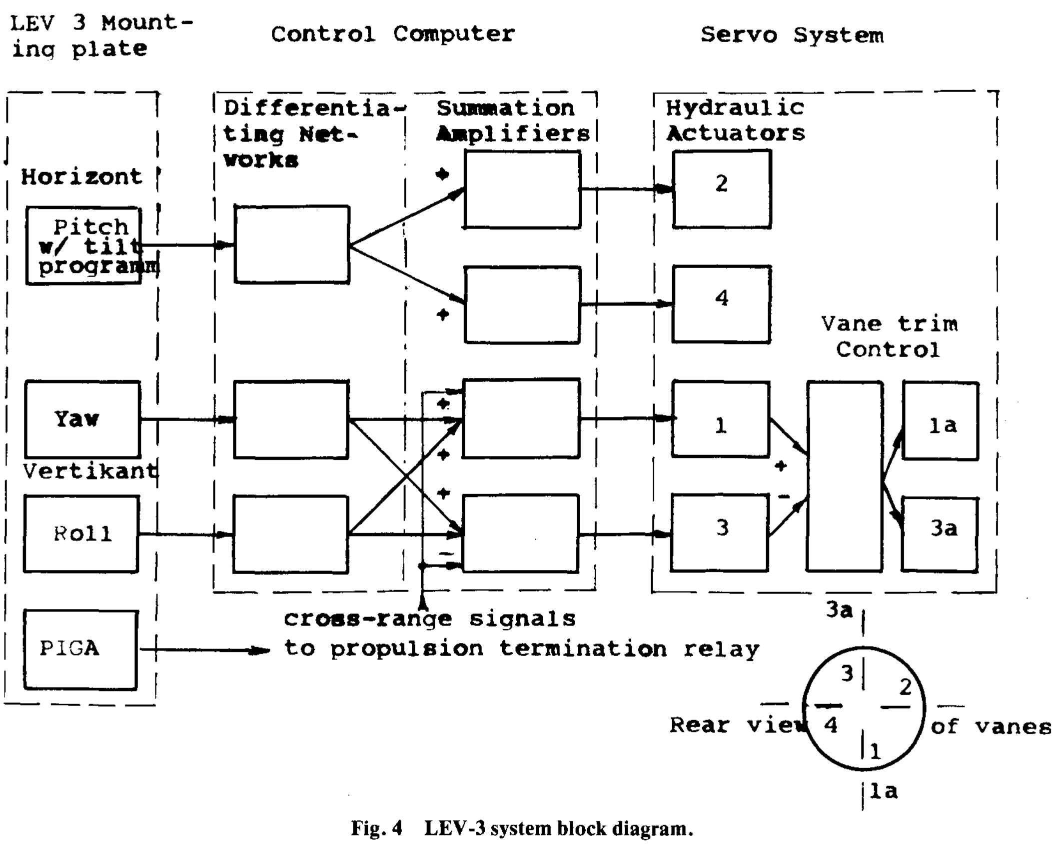

Fig. 4 LEV-3 system block diagram.

then transmitted to the ground station. The beat frequency of the return signal with the second harmonic of the frequency of the ground transmitter was passed to a Wien bridge. This bridge was preset to the beat frequency calculated for the desired cutoff velocity. When the actual beat frequency coincided with the preset frequency of the Wien bridge, a command was transmitted to the missile to initiate propulsion cutoff. The beat frequency, which was a measure of the missile velocity in the line-of-sight direction from the ground transmitter, corresponded to the correct missile velocity by locating the ground transmitter in the flight plane behind the launch pad so that its line-of-sight direction coincided with the cutoff point tangent of the desired trajectory.

During the early development of the control system, missile fixed rate gyros were used to provide attitude rate signals for stabilization. The rate gyros were soon replaced by passive differentiating networks. The control computer received the attitude signals for processing in these differentiating networks and combined the guidance information with the derived yaw signals.[4] The signals were then added as required and amplified in vacuum tube amplifiers to control the valves of the hydraulic actuators. The pitch actuators operated the jet and mechanically linked air vanes 2 and 4 (Fig. 4); the other actuators received superimposed yaw and roll signals and operated only the jet vanes 1 and 3. To compensate for missile asymmetries, that caused high roll torques, the air vanes 1a and 3a were used solely for integral roll control. The signal for this "trim control" was derived from the roll deflection of the jet vanes 1 and 3. The system block diagram is given in Fig. 4 for the LEV-3 case; the SG-66 platform had equivalent signal outputs to replace the LEV-3 signals.

Component Status at the End of World War II

Inertial sensors were limited in their accuracy mainly by the use of ball and pivot bearings. The friction of these bearings and of the potentiometer pickups caused the largest contributions in the error budget. Stabilizing and sensing servoloops for single-degree-of-freedom gyros and accelerometers used on-off contacts as powerful position sensors and provided satisfactory solutions with comparatively modest error contributions. Gyro wheels with an angular momentum up to 5 $m^2 \cdot kg \cdot s^{-1}$ were used to obtain low drift rates; a performance of somewhat better than 10 deg/h drift rate was achieved. Accelerometer-integrators yielded accuracies of 10^{-3}-10^{-4}. Considerable efforts were undertaken to obtain more accurate gyroscopic components by improving the existing design and higher precision in manufacturing, testing, and calibration. W. Angele was very successful in obtaining the required low drift rate on the LEV-3 gyros with a comparatively low angular momentum, as was Dr. F. Mueller in developing the stabilized platform SG-66.[2]

Considerable research was carried out to improve the components and to reduce their error terms. The double ball bearing with a superimposed oscillatory motion was invented by Dr. F. Gottwald; it reduced friction by almost two orders of magnitude. Magnetic amplifiers were developed by Prof. Th. Buchhold for improved frequency control of inverters needed to supply the power for the pendulous integrating gyro. Pressurized fluid bearings were conceived and investigated to provide low bearing torques to the single- and two-degree-of-freedom gyro; the latter had the form of a sphere, but possessed an axis with a maximum moment of inertia. Only a few of the numerous developments found actual application before the end of World War II, but they provided the background for future developments. The development of very sensitive accelerometers is most noticeable.

The pendulous integrating gyro accelerometer, developed by Dr. F. Mueller, is shown schematically in Fig. 5 in the simple version applied. It was used to determine propulsion cutoff. The gyro, mounted as a pendulum, maintained its

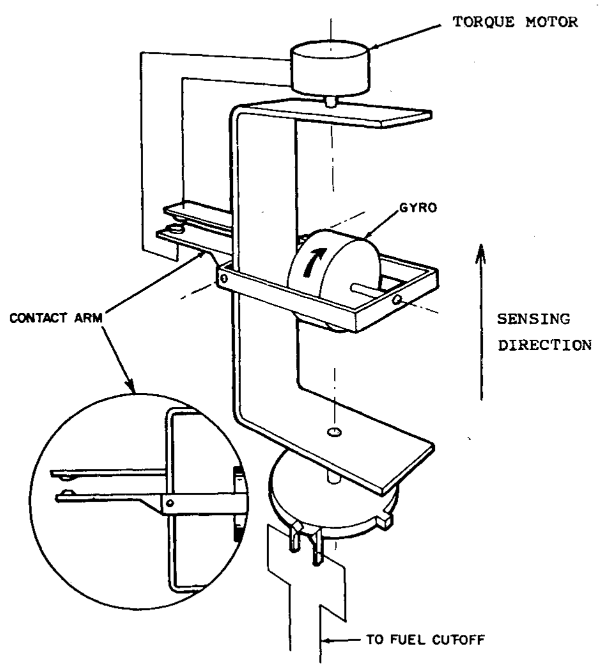

Fig. 5 Mueller pendulous integrating gyro accelerometer.

measuring direction through a simple, most effective, on-off control loop. This produced the required torques about the precession axis. Since the total precession angle of this device is proportional to the integral of the acceleration sensed, the rotation angle is proportional to the corresponding velocity. After the precession axis had rotated through an angle representing the cutoff velocity, a preset contact was closed to initiate propulsion cutoff. For a more complete cutoff equation, containing also the second integral of the acceleration (representing a displacement term), it was intended to connect a ball-and-disk integrator to the precession axis of the pendulous gyro accelerometer.

In a parallel development Profs. Th. Buchhold and C. Wagner conceived and developed a cutoff device illustrated in Fig. 6. The acceleration was sensed by a pendulous mass. Its displacement was measured electromagnetically; the sensor coils connected to a bridge circuit and a tube amplifier caused a feedback current to constrain the pendulum by a torquer. The feedback current is proportional to the acceleration experienced and passes through an electrolytic cell with silver chloride. The cell accepts a charge or discharge proportional to the current. Before the flight this cell had to be charged by a current in the opposite direction, corresponding to the value of the required cutoff velocity. In flight, the cell was discharged by the current constraining the accelerometer pendulum; it indicated its complete discharge with a small voltage jump on its electrodes. This signal initiated propulsion cutoff through a vacuum tube amplifier, which controlled the cutoff relay as depicted in Fig. 6.

The cross-range accelerometer, developed by Dr. H. Schlitt, was another version of an electromagnetically constrained pendulum. As schematically shown in Fig. 7, an electric coil, serving also as the mass of the pivoted pendulum, was inserted into the gap of a loudspeaker type magnet; the coil's constraining current was controlled by a single contact on the pendulum. The average current was proportional to the acceleration experienced by the pendulum. As shown in the figure, the contacts and a network containing the pendulum coil are elements of a bridge circuit. This bridge circuit permits the sign change of the constraining current with a single set of contacts for control; the network furnishes a signal (U) for guidance, which contains velocity and displacement proportional terms through integration in the network.

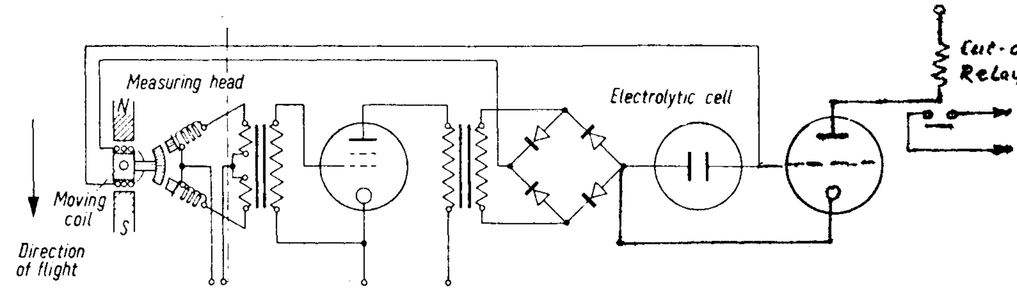

Fig. 6 Buchhold-Wagner accelerometer.

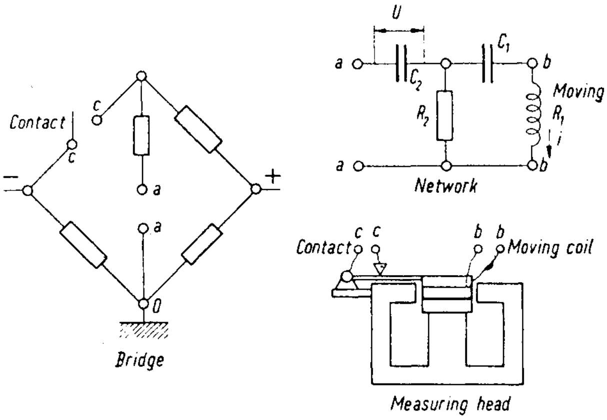

Fig. 7 Schlitt accelerometer.

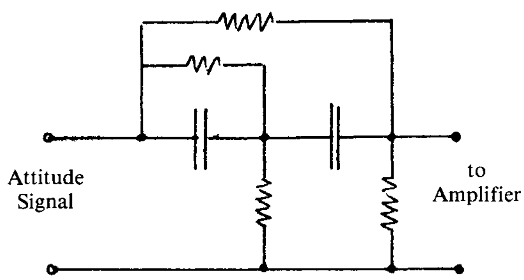

Fig. 8 V-2 pitch and yaw network.

Until 1940 all rocket control systems used vane position feedback and required a rate gyro to furnish attitude rate for stabilizing the control loops. Attempts had been made without success to stabilize vane velocity loops in obtaining the angular acceleration required for stabilization by mechanical devices. Furthermore, some problems were experienced with rate gyros exposed to the existing high vibrations. A major breakthrough was achieved that even led to the elimination of the rate gyro when Dr. H. Hoelzer[4] succeeded in obtaining the angular rate and acceleration proportional signal terms by passive electrical networks that differentiated the gyro attitude signal. In addition, these networks featured more freedom in selecting the frequency characteristics of the control loop and an adaptation to the coefficients varying with flight time. A typical network as it was used in the pitch and yaw control loops is shown in Fig. 8.

When the only available aircraft actuator that came closest to the required specifications was selected, it was accepted that its power output had to be upgraded considerably. W. Angele succeeded in fulfilling the demands, whereas other backup developments did not obtain the desired performance in time. These developments included electric motor drives with on-off relay control, motors with extremely small time responses, and continuously running electric motors with differential magnetic clutches.

Control and Stabilization Schemes and Analysis During World War II

In the early 1940s only very simple analytical methods were available for the stability analysis of control systems. Furthermore, they were limited to linear systems, whereas the actual control components and systems for missile control possessed considerable nonlinear characteristics. In addition, the coefficients of the moment and force equations of the rocket and the servotorque load varied over wide ranges during the propelled flight of the rocket. Some data were known only with comparatively wide tolerances.

In the first attempts to develop and test a control system on the ground, the test vehicle was suspended in the firing test stand such that it could move in one axis about its center of mass. It experienced prorated torques by heavy springs, thus simulating the aerodynamic stability of the missile. Of course, this method permitted only very poor flight point performance simulations, since the spring torques could not be time adjusted to the time-varying aerodynamic coefficients. Furthermore, the center of mass shifted with the fuel consumption. The method was very cumbersome as well as inaccurate because of the high friction in the suspension bearings and because no true simulation of the actuator torque load could be obtained.

A new, pure laboratory simulation of the missile was suggested by Dr. H. Steuding. In the moment equation normalized with respect to the moment of inertia

$$\ddot{\varphi}+d\dot{\varphi}+c_1\varphi+c_2\beta=\Sigma m_{\text{dist}} \tag{1}$$

with the definitions

φ = attitude angle
β = vane deflection angle
d = normalized aerodynamic damping coefficient (negligibly small)
c_1 = normalized aerodynamic moment coefficient
c_2 = normalized control moment coefficient
Σm_{dist} = normalized sum of perturbation torques acting on the missile

the angle of attack was approximated by the attitude angle of the missile. The simulation device is shown in Figs. 9a and 9b. Figure 9a indicates the position for $\varphi=\beta=0$. Figure 9b displays an instantaneous position for the platform deflected by the attitude angle φ with the servodrive shifting the c_2 spring carriage by an angle proportional to the vane deflection β. Since the total control system with the exception of the power supply was mounted on the platform, the simulation included all components. Nonlinearities due to the angular deflections and trigonometric relations were of an acceptable magnitude, and friction effects were kept within reasonable limits by the design, which used special knife edges instead of ball bearings and flexible electrical cables for power transmission. A shortcoming was that the servoactuator load could not be simulated. The mechanical motion simulator was of considerable value in the study of the control system stability despite its still time-consuming change of moment coefficient c_1 and c_2 for different instants of time.

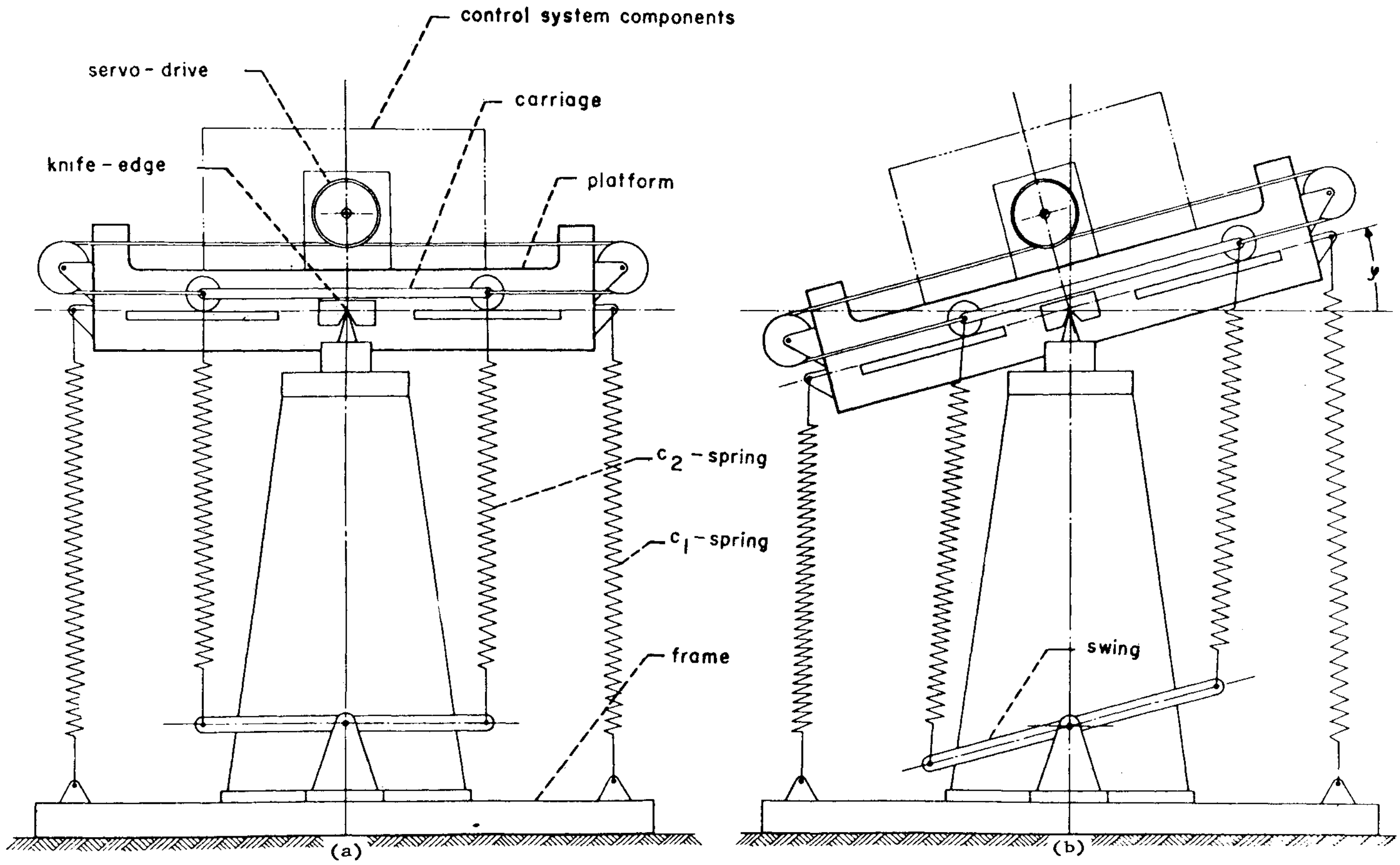

Fig. 9 Mechanical attitude motion simulator.

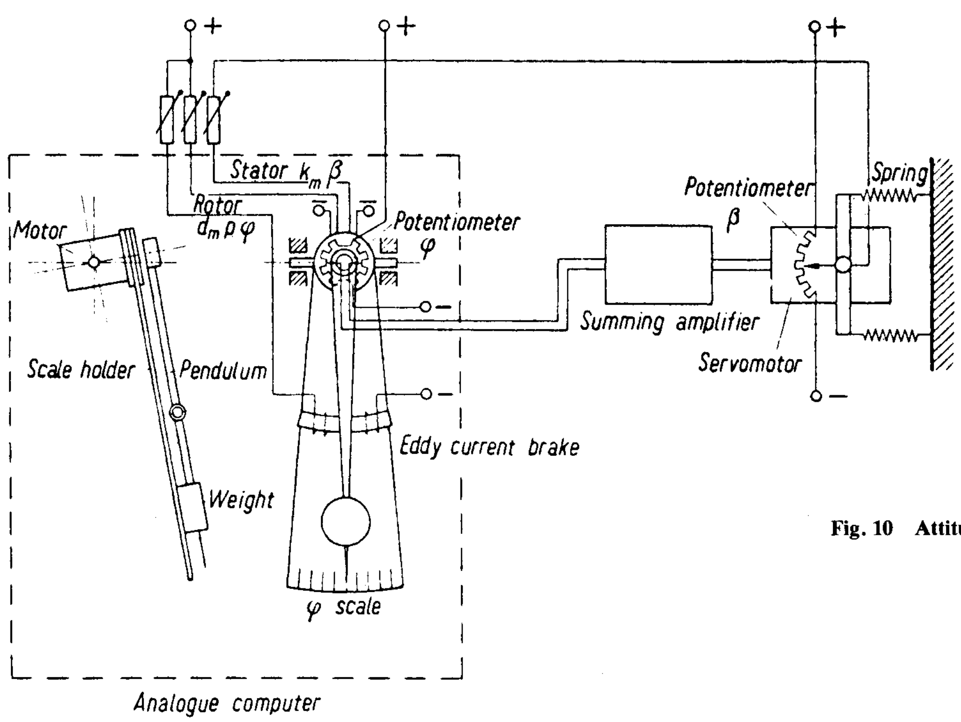

Fig. 10 Attitude control-loop simulation with pendulum.

For a more elegant simulation it became necessary to drop the requirement to subject the total control system to the attitude motion. This became possible when the rate gyro could be deleted and its information be replaced by an electrical differentiating network as mentioned previously. In the improved simulation scheme by the author, the homogeneous part of Eq. (1) was simulated by a physical pendulum and an eddy current brake (Fig. 10). Tilting the plane of motion of the pendulum about the horizontal axis permitted one to change the restoring coefficient c_l with the sine of the tilt angle, measured from the horizontal toward the stable vertical pendulum plane position. Negative restoring coefficients could be simulated, as well, by turning the pendulum plane toward the unstable pendulum position. For

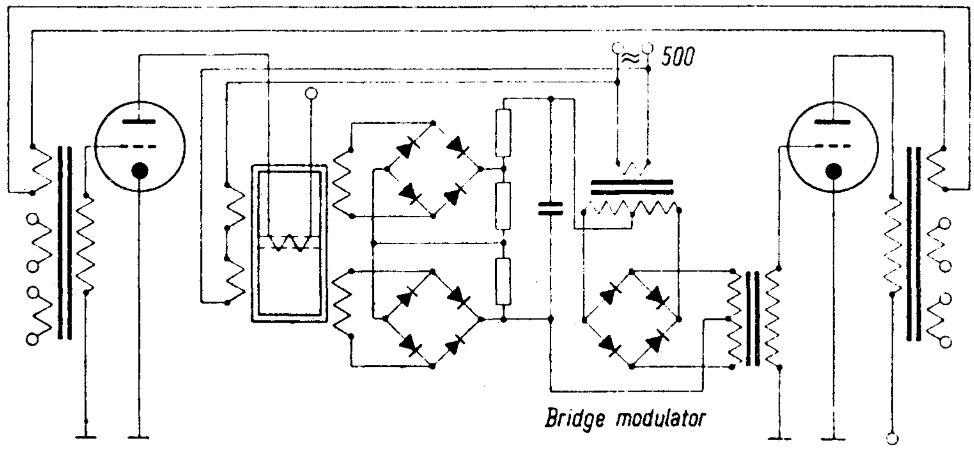

Fig. 11 Integrator amplifier (ac).

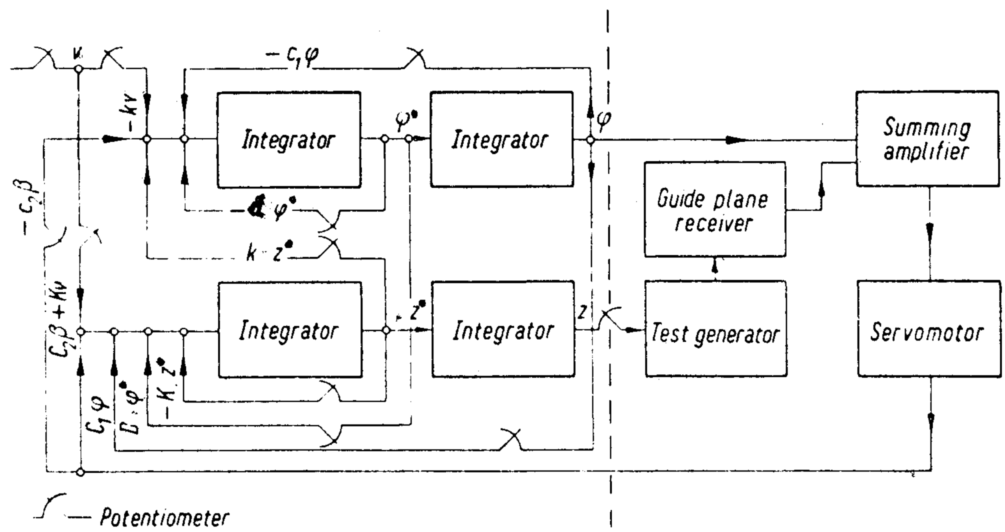

Fig. 12 Simulation of yaw and cross-range with radio guidance.

simulation of the inhomogeneous part of Eq. (1) a linear torquer was mounted to the pendulum axis; the torquer was controlled by powerful electrical signals that represented the variables in the inhomogeneous terms. The pendulum bearing friction was almost completely eliminated by a motor-driven oscillating double ball bearing. The remaining damping effect was added to the damping term of the eddy current damper to obtain the total damping coefficient. The gyroscope, behaving like an ideal component in the stability investigation, was excluded from the simulation. An equivalent attitude pickup was provided on the pendulum axis to simulate the attitude signal. Since the servoactuator was only electrically connected to the simulator, it could be loaded with hinge torques provided by leaf springs, yielding a true simulation.

The pendulous simulator could be used satisfactorily to investigate servoloops with natural frequencies up to 1 Hz; above this frequency time constants, mainly caused by the electromagnetic torquer, began to influence the simulation detrimentally. Smaller missiles, especially anti-aircraft rockets, had to operate with higher control loop frequencies, demanding the elimination of the time constants of the pendulum torquer loops. To fulfill this requirement the author provided the necessary torque by rotating the pendulum plane about the zero pendulum position axis by an angle proportional to the vane deflection, the most important variable in the inhomogeneous terms.

The simulation of the force equation to investigate the radio guidance stability also was first attempted by electromechanical means, but the necessary accuracy of the simulation could not be achieved, and the time varying coefficients could not be introduced satisfactorily. An electronic analog computer proved to be the best solution; it consisted mainly of electronic integrator amplifiers. The time varying coefficients were included in the simulation and were provided by potentiometers that were adjusted by a cam driven timer motor. Figure 11 gives the simplified diagram of the integrator amplifier, using an RC (resistance-capacitance) network for integration. A demodulator and a modulator were used for stable ac amplification. The positive feedback practically served to compensate for the losses of the capacitor and circuit resistors, causing an almost ideal integration. Dr. H. Hoelzer and O. Hirschler deserve credit for this first high-performance electronic ac analog computer development.[4] Figure 12 shows a diagram of the simulation of attitude and cross-range motions with radio guidance. The moment equation is the same as given in Eq. (1) with the disturbance torques defined as

$$\Sigma m_{\text{dist}} = k\dot{z} - kv$$

from the effects of cross-range missile and wind velocities. The force equation, normalized with respect to the mass of the missile, was

$$\ddot{z} = D\dot{\varphi} + C_1\varphi + C_2\beta - k\dot{z} + kv \tag{2}$$

The yaw control equation was of the form

$$f(\beta) = a_0\varphi + a_1\dot{\varphi} + a_2\ddot{\varphi} + e_{-1}\int\epsilon \mathrm{d}t + e_0\epsilon + e_1\dot{\epsilon} \tag{3}$$

The definitions in these equations in addition to those given with Eq. (1) are

z = cross-range displacement
ϵ = angular cross-range displacement (see Fig. 4)
v = lateral wind velocity
D = normalized damping coefficient
C_1 = normalized lateral force coefficient due to missile deflection
C_2 = normalized lateral force coefficient due to control vane deflection
k = normalized force coefficient due to cross-range missile velocity or lateral wind velocity
a_i = gain factors related to attitude angles and their derivatives
e_i = gain factors related to angular cross-range displacement and its derivatives

The vane function $f(\beta)$ in the control equation contained a predominant term of the vane velocity $\dot{\beta}$ since no position feedback was applied. A β term existed due to hinge torques from the vanes and the marginal torquing capability of the servoactuator. The function was linear only for very low vane velocities and without torque load; the nonlinearities under operational conditions made it mandatory to investigate the control loop characteristics by simulation.

Component and Systems Development

After World War II many of the German scientists and engineers accepted the offer of the U.S. Army to continue their R&D work in Fort Bliss, Texas. In 1950 they moved to the Redstone Arsenal, Alabama where their work on components and systems continued.[5] The first years were devoted to pursue ideas and concepts that had emerged in the preceding years. Thus many innovations leading to higher performance were available when the first assignment to develop the Redstone missile was given. Many young and gifted engineers from the United States joined the former German group of Dr. Wernher von Braun to form a development team. Additional support, especially in manufacturing, came from industry.

The demand for an accuracy improvement of at least one order in comparison with the V-2 accuracy could only be achieved by more precise inertial sensors. Only with their improvement did it pay to develop more accurate guidance schemes. Convinced that the required accuracy could be obtained with a pure inertial system, only this method was pursued. A thorough error analysis for the components, as well as for the systems, was performed in order to yield the greatest error contributions so that they could be reduced first in the subsequent development efforts. System and hardware had to be balanced to obtain an optimal system accuracy.

A breakthrough to obtain considerably higher accuracy of the inertial sensors was obtained in using externally pressurized, cylindrical gas bearings for the precession axis of the single-degree-of-freedom gyro, and to provide a low torque producing pickup for the gyro stabilization loop. Since the gas bearing possesses a negligible damping, a double differentiating network was required for a stable gyro servo loop.[6] The electronics needed could soon be realized using the transistors which were becoming available at that time.

The single-degree-of-freedom gyro was selected in contrast to a two-degree-of-freedom configuration because it promised simpler design and manufacturing as well as easier precision testing and error analysis. The development was started with a wheel having an angular momentum of 5 $m^2 \cdot kg \cdot s^{-1}$. A cross-sectional view of such a gas bearing is shown in Fig. 13.

It should be remarked here that the original method for providing a frictionless and extremely low torque bearing was to use an externally pressurized water bearing, as conceived by Dr. J. Gievers. Laboratory work on such a bearing had been carried out at Kreiselgeraete, the contractor for the SG-66 in Germany. Dr. F. Mueller and H. Rothe, who deserve credit for the follow up development in the U.S., envisioned a better performance by replacing water with gas. This eliminated the need for a fluid pump but required accepting a high-pressure

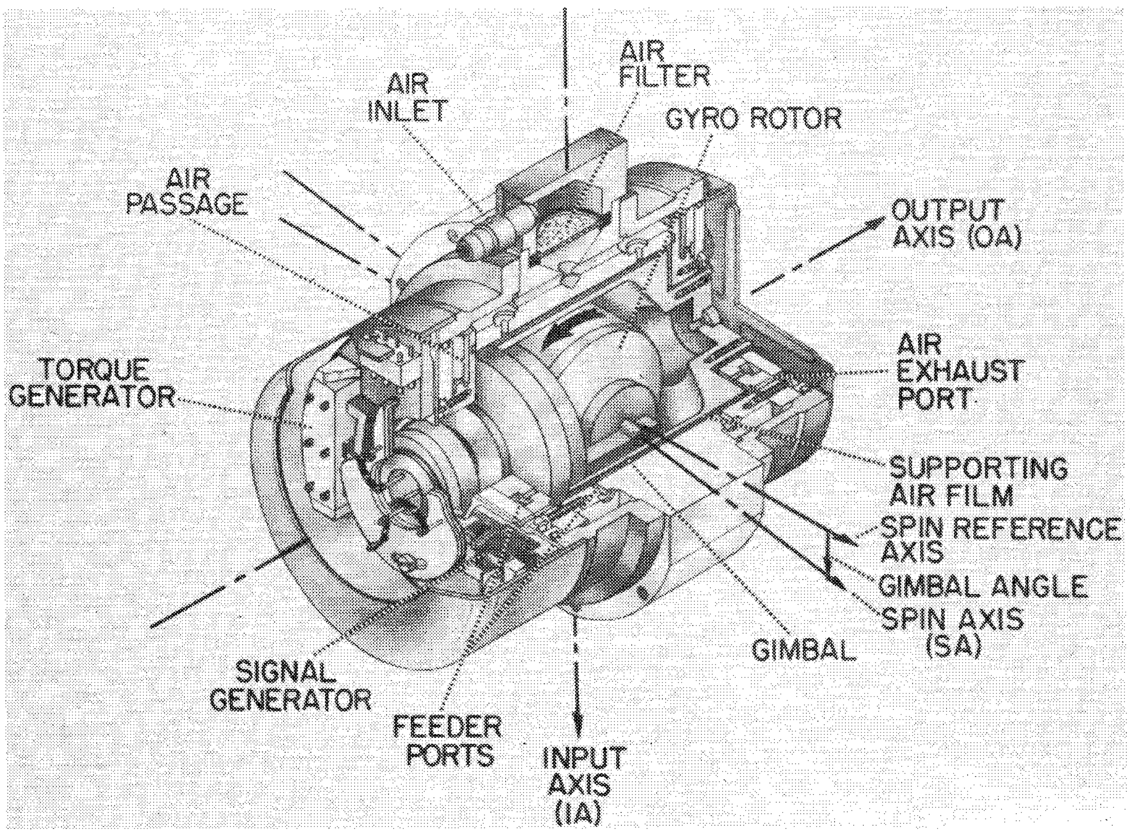

Fig. 13 Single-degree-of-freedom gyro with gas bearing.

gas supply. It was a very satisfactory solution, except that the strong desire persisted, especially later for space flight, to become independent of an exhaustible very-clean gas supply, including supply lines which had to cross the gimbals of the stabilized platform. Attempts to provide a pump for recirculating the gas in an integral unit were not successful because the gas pump required too much energy and disturbed the temperature equilibrium for the gyro. It was conceived that a low power consuming recirculating pump could be built for a liquid of low density, provided that such a liquid would give satisfactory results for the bearing torques. Engineers at the Bendix Guidance Systems Division, the contractor for the Pershing and Saturn launch vehicle inertial guidance systems, tested an unmodified gas bearing with freon. The results fully justified the idea, and the bearing did not show any performance degradation in comparison to gas as the pressurization medium. Thus today a series of integral inertial sensors is available with very modest power and volume requirements for the fluid pump. Credit for this latest development belongs to T. Morgan and H. Schulien.

The well-known three-gimbal mounting of the three-axis stabilized platform was replaced by Dr. F. Mueller with a universal joint type mounting, named the internal gimbal system. This change was possible because only the pitch axis required a large angular freedom close to 180 deg, whereas an angular freedom of ±20 deg was permissive for the roll and yaw axes of the missiles under development. The internal gimbal system had the advantage of an easy access and exchangeability of the platform components; it led also to a smaller size and lower weight of the stabilized platform system. Figure 14 shows the stabilized platforms featuring the internal gimbal system for the Redstone, Jupiter, and Pershing missiles.

For several years the development of the double integrating accelerometer was pursued. The principle of this accelerometer is to sense the acceleration by an unbalanced motor stator, which is supported by a low torque bearing, such as an externally pressurized gas bearing. The current of the rotor is servocontrolled by a sensitive pickup measuring spurious deflections of the pendulous stator such that the resulting torque counteracts the stator's pendulous torque from the acceleration. Since in reaction the rotor is subjected to the same torque, it accelerates and its angular speed and displacement become the first and the second derivative of the acceleration measured. This basically attractive method did not yield satisfactory results. Besides the complexity of the control loop an important reason was the rotor's power consumption. Being a function of the square of the acceleration, its magnitude prevented the thermal stability required for the pendulosity of the stator. Thus a normalized accuracy close to 10^{-4} was not achievable. After a comprehensive review, Dr. F. Mueller again proposed to use the

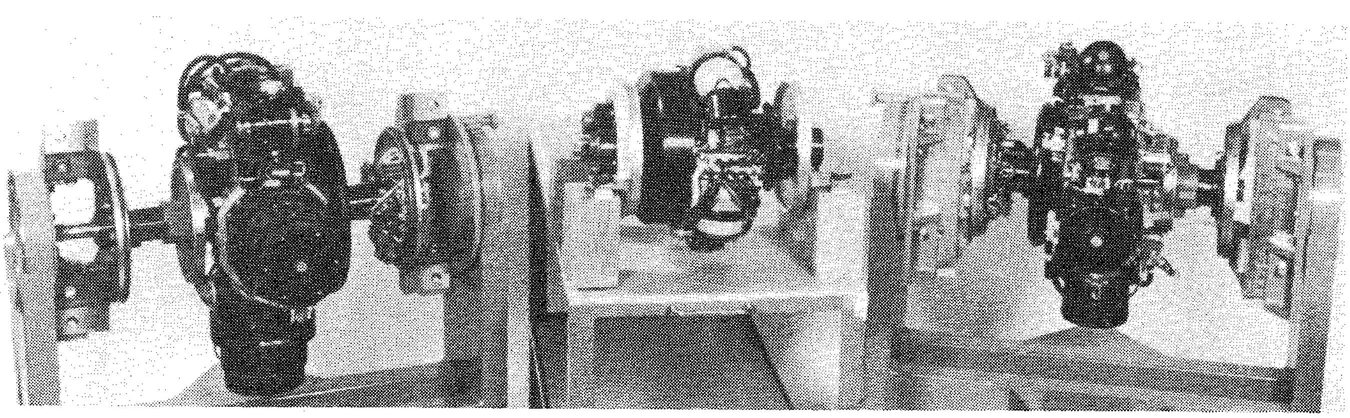

Fig. 14 Stabilized platforms for Redstone (left), Pershing (center), and Jupiter (right) missiles.

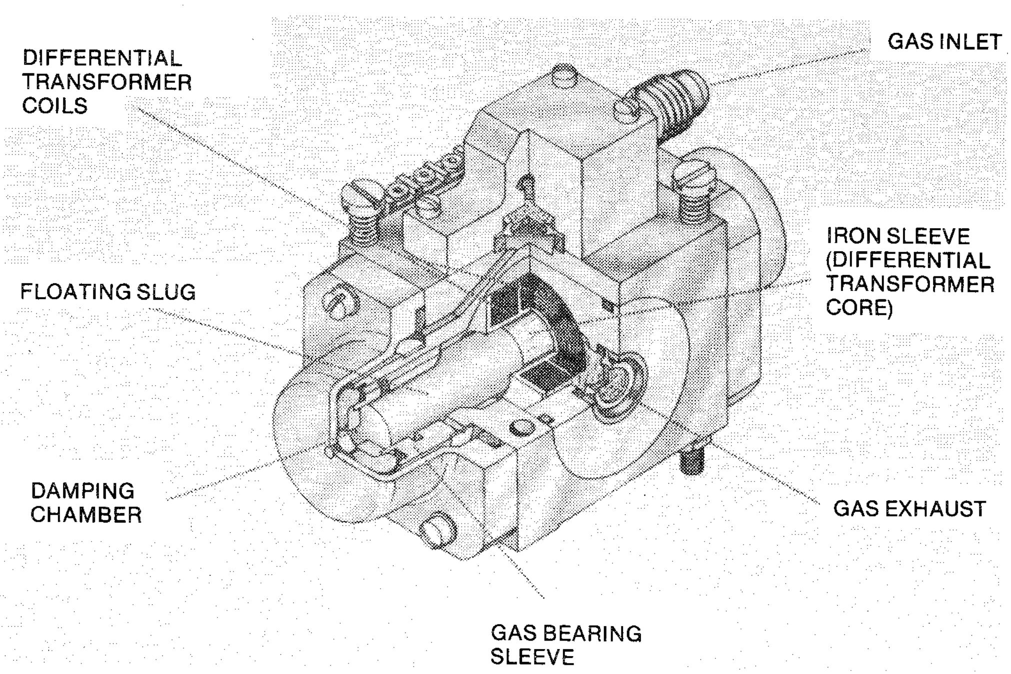

Fig. 15 Gas-bearing pendulum.

Table 1 Characteristic data of gyros and accelerometers[a]

	Redstone		Jupiter		Pershing		Saturn	
	Gyro	Acc.	Gyro	Acc.	Gyro	Acc.	Gyro	Acc.
Angular momentum, $m^2 \cdot kg \cdot s^{-1}$	5.0	0.10	1.4	0.044	0.21	0.011	0.21	0.011
Material of bearing	Monel	Monel	Ti	Monel	Al and Be[b]		Be	Be
					Be[c]	Monel[c]		
Mass of unit, kg	7.6	2.36	4.0	1.55	1.27	0.77	1.27	0.77
Flywheel power, W	40	7	32	6	10	4	10	4

[a] Wheel speed: all gyros = 24,000 rpm; all accelerometers = 12,000 rpm.
[b] Sleeve and endplates.
[c] Can or inner cylinder.

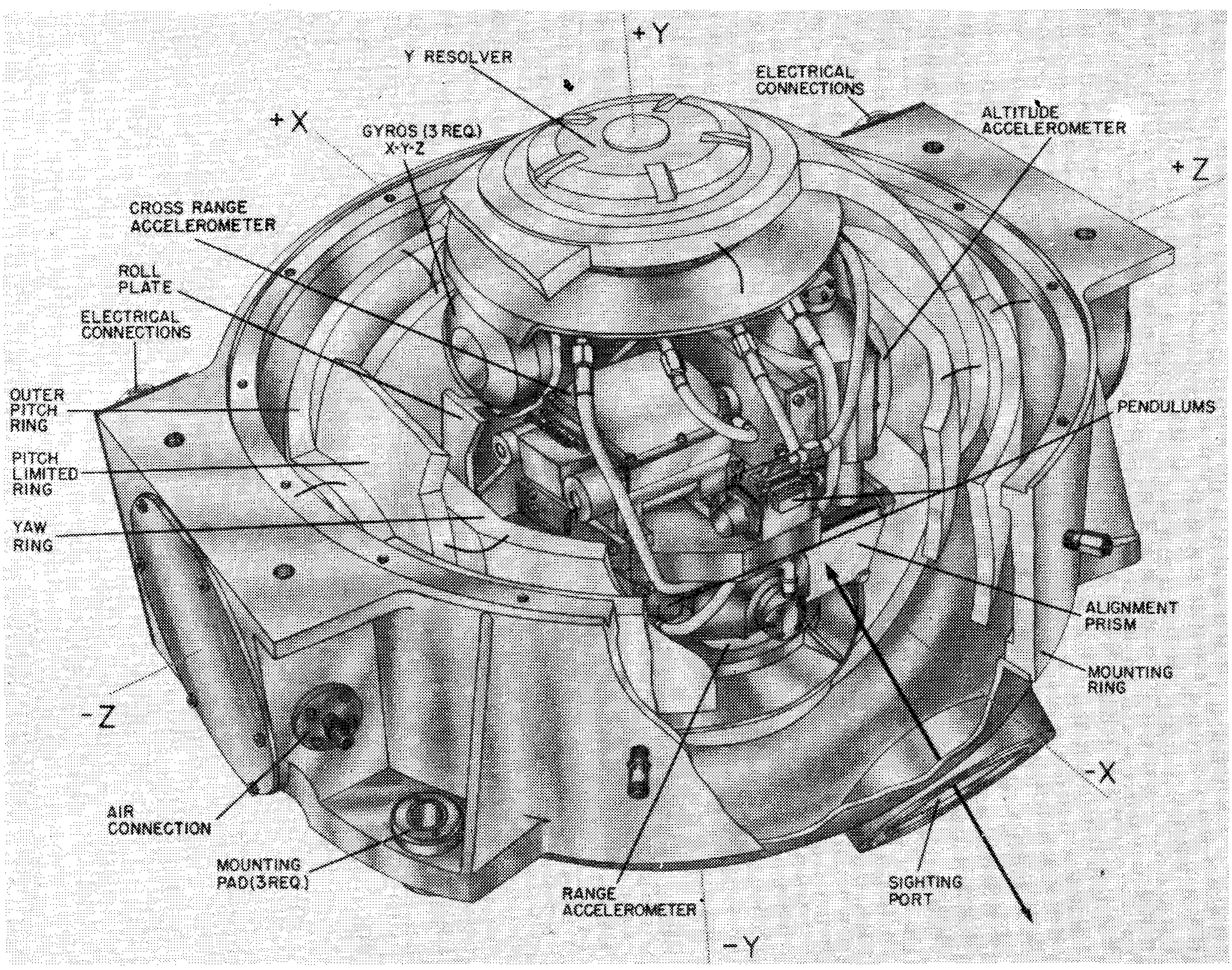

Fig. 16 Stabilized platform of Saturn launch vehicle.

pendulous integrating gyro accelerometer (PIGA), improved by a low torque, externally pressurized gas bearing for the pendulosity axis. This device became the most precise accelerometer integrator; the acceleration dependent power consumption of its torque motor is modest and has no effect on the thermal stability of the pendulous gyro.

Basically, it is possible to level the stabilized platform by its accelerometers. However, the control loop is much simpler if a separate level indicator is used. Since such a sensor can be designed for a very small range or tilt angle, it becomes simple and reliable. It can also be used for a convenient check of at least two guidance accelerometers, depending on their orientation. A gas bearing pendulum, actually a floated slug in a sleeve, as shown in Fig. 15, had been developed. Two of these pendulums have been mounted on the stabilized platform for its leveling.

The velocity from the integrated acceleration in the slant-range direction had to be transmitted from the stabilizer mounted PIGA to the guidance computer with a normalized accuracy of at least 10^{-5} to keep the error contribution from this transmission small. Electromechanical synchrotransmitters were used to bring the information to the computer, to be used for the propulsion cutoff equation and to integrate it for obtaining displacement information. This integration was accomplished mechanically with a high-precision ball-and-disk integrator. Similar components with a slightly reduced accuracy were used for cross-range and slant-altitude guidance.

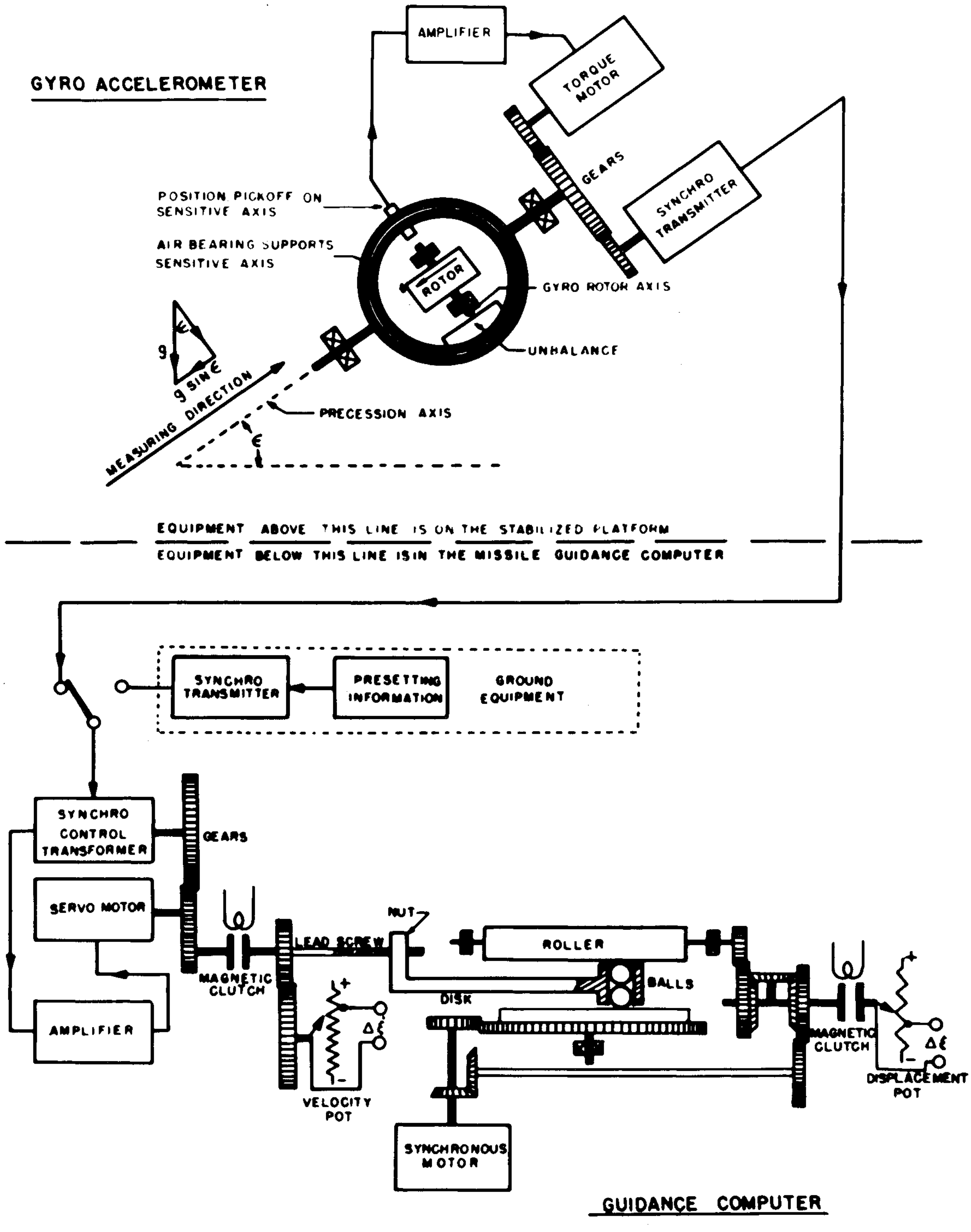

Fig. 17 Simplified diagram of the slant-range delta-minimum guidance system.

The described devices were the main guidance components for the Redstone, Jupiter, and Pershing missiles. The short-range Redstone missile, which used only cross-range guidance and slant-range information for precision velocity cutoff, became the first ballistic missile with a highly accurate inertial guidance system and proved the feasibility and accuracy of it. For the following medium-range Jupiter missile the accuracy was further improved not only by more precise inertial sensors of smaller size but also by adding slant-altitude guidance. This complete guidance method was retained for the medium-range Pershing; despite a further miniaturization, the accuracy could be increased. To illustrate the successive improvements Table 1 gives characteristic data of the gyros and accelerometers used in these missiles and the Saturn launch vehicle, which had Pershing size inertial sensors. The materials for the bearing sleeve and can were changed for improved structural and thermal stability. The low gas bearing torques were further reduced for the Saturn launch vehicle application. Since these versatile launch vehicles required larger gimbal freedom about all three gimbal axes than the previous missiles, a return to the external gimbal configuration became necessary. Figure 16 is a picture of the Saturn's stabilized platform with the external gimbals.

At the beginning of the 1960s the technology for high-precision digital encoders permitted replacing the electromechanical synchrotransmitter of the PIGA by optical digital encoders. digital information was then used directly for guidance computations in Saturn launch vehicles.

Delta-Minimum Guidance Scheme

In phase with the development of inertial sensors with higher accuracy a more precise guidance scheme had to be developed. Professor Th. Buchhold decided that the missile be constrained to its nominal flight path (hence the name "delta-minimum guidance"[7] later given to the scheme) by the guidance commands like the old V-2. For propulsion termination a condition was applied as it was conceived for an improved V-2 missile cutoff. The cutoff equation was of the form

$$\Delta\dot{\xi}_c + (1/T)\Delta\xi_c + \dot{\xi}_{rt} = 0 \tag{4}$$

containing a velocity difference and a displacement deviation term. The first term $\Delta\dot{\xi}_c$ accounted for the slant-range velocity difference, the second $(1/T)\Delta\xi_c$ for the slant-range displacement of the actual cutoff point from the nominal cutoff point. The value T corresponded closely to the flight time from cutoff to impact. A third constant term $\dot{\xi}_{rt}$ accounted for the slant-range velocity gained due to the remaining average thrust impulse after initiation of cutoff.

The guidance scheme for the slant-range direction ξ is shown in Fig. 17 with the gyro accelerometer measuring in the slant-range direction, elevated by the angle ϵ against the horizon; this angle was optimized for the described cutoff Eq. (4) to obtain minimum range errors. Figure 18 displays the velocity diagram in the slant-range direction simplified for a homogeneous gravitational field. It indicates the presettings in velocity ($\dot{\xi}_0$) and displacement (ξ_0) needed to obtain the velocity and displacement deviations for Eq. (4). It should be mentioned that the third term in the cutoff equation was taken care of by combining it with the presetting of the velocity term.

Cross-range guidance was designed as a pure nulling system, using the cross-range velocity and displacement information as additional input signals for the yaw control channel. To constrain the missile within the flight plane in the slant-altitude direction a velocity program had to be generated to counteract the effect of the accelerations measured by the slant-range accelerometer, if the missile would fly the nominal trajectory. With this program incorporated, the slant-altitude guidance computer furnished only flight path deviations into the pitch control channel for zeroing the deviations.

Guidance and Control for Explorer I

Plans to bring the first U.S. satellite into orbit with the Redstone missile as launch vehicle had been worked out as early as 1954, several years before it could be accomplished. The planned satellite launch vehicle, named Jupiter C, was conceived as a three-stage solid-propellant rocket cluster, rotating about its symmetry axis for attitude control, on top of the Redstone missile.[8]

Dr. W. von Braun recognized early that the concept could be tested in combination with nose cone ablation tests needed for the medium-range and intercontinental missiles. The ablation test required a high re-entry velocity of a nose cone into the atmosphere. It could be accomplished with the first two stages of the solid propellant rocket cluster as planned for the satellite configuration. The ignition of the first solid propellant stage had to occur after apex to obtain the desired high re-entry velocity. Guidance requirements did not exist except that the ignition of the first cluster stage had to be given as required for either configuration. This was accomplished by using the one-way Doppler effect to measure the velocity for initiating ignition by radio command. The two-way Doppler scheme as used for the V-2 was prohibitive; it would have added too much mass.

The control system requirements were practically the same for both purposes; only the pitch angle at the instant of firing the solid propellant cluster was different. Analytical as well as flight simulation investigations were carried out to determine the gyroscopic effects of the high-angular momentum of the spinning rocket cluster, which had to be overcome by the attitude control system during the tilt program.[9,10] A common requirement was that all hardware should be of minimum mass and, of course, inexpensive. Thus the LEV-3 system, of which a sufficient number still was available, was used with an extended tilt program as the only modification. The gas jet reaction control system for the Redstone missile's terminal flight was found satisfactory for actuation.

The Jupiter C test flights in 1956 and 1957 were highly successful. They assured all participants that the vehicle, just by adding the third solid propellant stage (consisting of one single rocket), could launch the first Earth satellite. As is well-known, the first attempt to launch a U.S. satellite with the Redstone-solid-propellant-cluster configuration, later named Juno I, was successful.

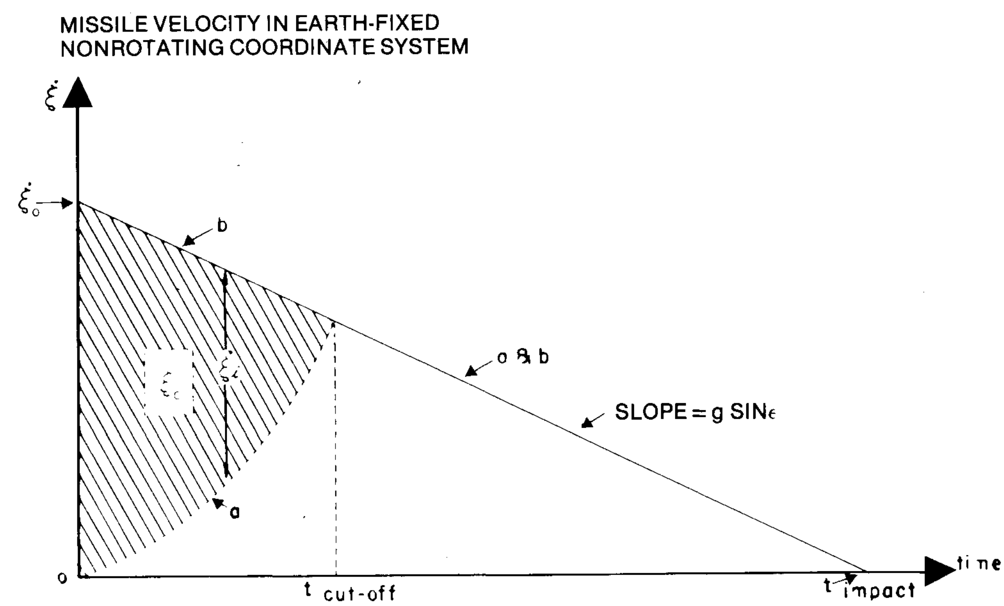

Fig. 18 Velocity in slant-range direction: a = standard missile velocity; b = reference missile velocity, if missile would not be subjected to other than gravitational forces.

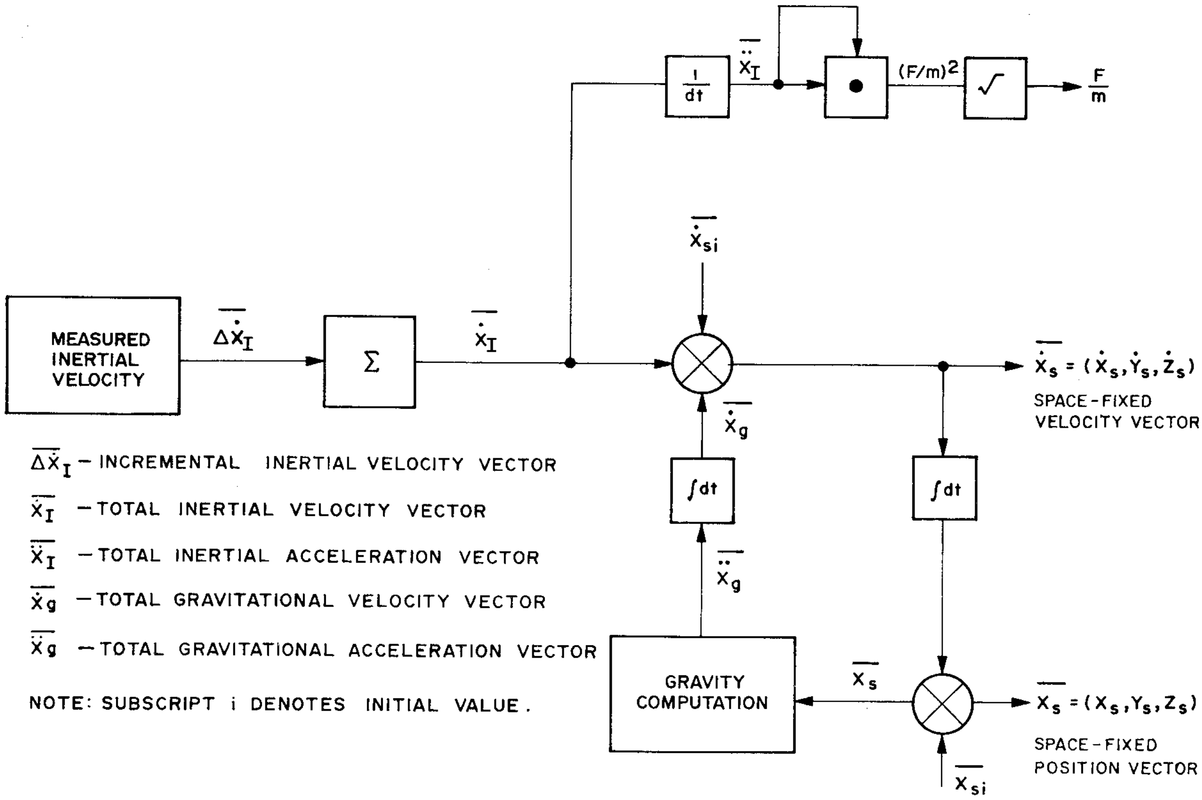

Fig. 19 Navigation computations.

Navigation and Guidance of the Saturn Launch Vehicles

The rapid development of digital computers in the late 1950's permitted digital onboard computations of navigation and guidance functions with negligible error contributions from the computations. Thus new schemes were investigated by members of the Aero-Astrodynamics Laboratory, directed at that time by Dr. E. Geissler. For propellant economy the iterative, path adaptive guidance mode was developed and selected. The flight vehicle is guided along a minimum propellant trajectory to obtain the required end condition, such as the required velocity vector and vehicle position in orbit or the velocity vector for translunar injection.

With digital computation it became possible to transform the information measured by the gravity insensitive accelerometers into a space-fixed coordinate system. Figure 19 shows how the vehicle state is derived from the velocity, measured by the accelerometer integrator, in the form of position and velocity vectors in a space-fixed coordinate system which has its origin at the center of the Earth. The diagram shows further the calculation of the acceleration in the space-fixed coordinate system as thrust to mass ratio, F/m.

The iterative guidance mode for flight path adaptive guidance along a minimum propellant trajectory determines the necessary vehicle attitude corrections and the propulsion termination command to obtain the required end conditions for the flight vehicle. The guidance functions for the thrust direction and for its termination were expressed by a series development of the reference attitude and of the propulsion cutoff time in terms of position and velocity components, time, and acceleration; the coefficients of the terms were determined by the calculus of variations and stored in the guidance computer. Further details concerning the development of the guidance equations are given in Refs. 6, 11, and 12. Reference 7 also contains some characteristic data of the inertial sensors and the computer system. The accuracy of the navigation and guidance system was excellent. The drift rate of the inertial components during Earth orbit was unexpectedly low. Translunar injections were of such an accuracy that considerably less impulse or none had to be given in the planned midcourse maneuvers. As a result, on subsequent missions the payload could be increased repeatedly because of the possible reduction in fuel for the midcourse maneuvers.[13]

Redundancy in hardware components and in software were applied as far as feasible in the NGC system. Some backup modes would have given a degraded accuracy, but avoided a serious mishap by providing a safe flight continuation. Some single failure points, however, could not be avoided. It is most remarkable that none of the redundant methods in the launch vehicle system ever became operational due to a failure of the primary components, neither during the extensive time of checkout nor during the actual flight.

Control System Developments

Until the end of World War II all the rockets to be controlled were aerodynamically stable and could be considered as rigid bodies. Sloshing was investigated by Dr. E. Geissler but never did cause any problems. New missile configurations demanded the stabilization of aerodynamically unstable missiles. Laboratory simulations permitted the development of the proper control system. Additional analytical investigations had shown that either an angle-of-attack meter or a body-fixed accelerometer could be used to provide input signals into the control system such that a drift-minimum or a load-minimum condition could be accomplished for the aerodynamically unstable missile. To achieve this desirable feature on the Hermes II, a V-2 size missile pursued by the U.S. Army, an available Schlitt accelerometer as described earlier was incorporated as required by the analysis. The subsequent missile failure and flight data showed clearly that missile vibrations were sensed by the control accelerometer. Since the signal was not filtered for vibration frequencies, it saturated the control loop and thus caused the failure. In the following launch without the control accelerometer the Hermes II became the first successfully flown, aerodynamically unstable missile.

Sloshing caused an unexpected failure of an early Jupiter test missile. The analytical evaluation of damping effects had been insufficient and had to be supported by sloshing simulation. The problem was solved temporarily with the "beer can method": aluminum cans of a size that could be inserted through the man-hole of the next already assembled missiles floated on the propellant surface. Because of their friction against each other and the wall of the tank, sloshing was suppressed sufficiently. The ensuing solution, to attach baffles to the tank walls, became the final solution.

Both experiences, many years apart, brought the requirement to thoroughly investigate possible interactions of the control loop frequencies with vibration, sloshing, and bending modes and to provide possible filters to suppress undesirable control inputs. Components were thus not only subjected to the known vibration spectrum in laboratory tests during their operation in a simulated control loop, but in some cases they were included in static firing tests.

The low reliability of vacuum tubes in flight components caused Prof. Th. Buchhold, head of the guidance and control laboratory, to decide that they should be avoided wherever they could become single failure sources; different amplification methods should be applied. Magnetic summing stages and amplifiers with sufficiently small time constants were developed. Transistor developments were strongly supported; transistors were used as soon as they became reliable and available. Their first application was in low-power circuits and preamplifiers. Later attainable power transistors were used for powerful, low-time-constant amplifiers.

All the missiles mentioned were not equipped with rate gyros; in some, differentiating networks for the attitude signals provided for the stability of the control motion. In contrast, the Saturn vehicles used rate gyros to obtain the rate information. One important reason was that a bending node might be at or near the location of the stabilized platform, causing a bending mode frequency signal which would be amplified in the differentiating network. In contrast, if it were not possible to find a single antinode location for a rate gyro, several rate gyros could be mounted at antinode locations for subsequent flight intervals with a switch-over capability to obtain always clean attitude rate information.

A low time-constant motor actuator was developed for the Redstone missile, based on development work before the end of World War II, because a properly sized servoactuator was not available. A tandem arrangement was selected for reliability reasons to drive the jet and air vanes through a gear unit. The split pole motor, invented by Dudenhausen, had a very low-time constant due to the suppression of the armature's magnetic field. The author developed a polarized relay/heavy duty relay on-off amplifier with voltage feedback to control the dc drive motors. The polarized relay served also as a mixing stage for the roll superposition to pitch and yaw, and to accept the motor voltage and vane position feedback signal. The heavy-duty relay was specially developed for a short response time.

With the development of more powerful rocket engines it became mandatory to swivel these engines for pitch and yaw control. For the high torque and power requirements the development of hydraulic, position controlled, low reaction

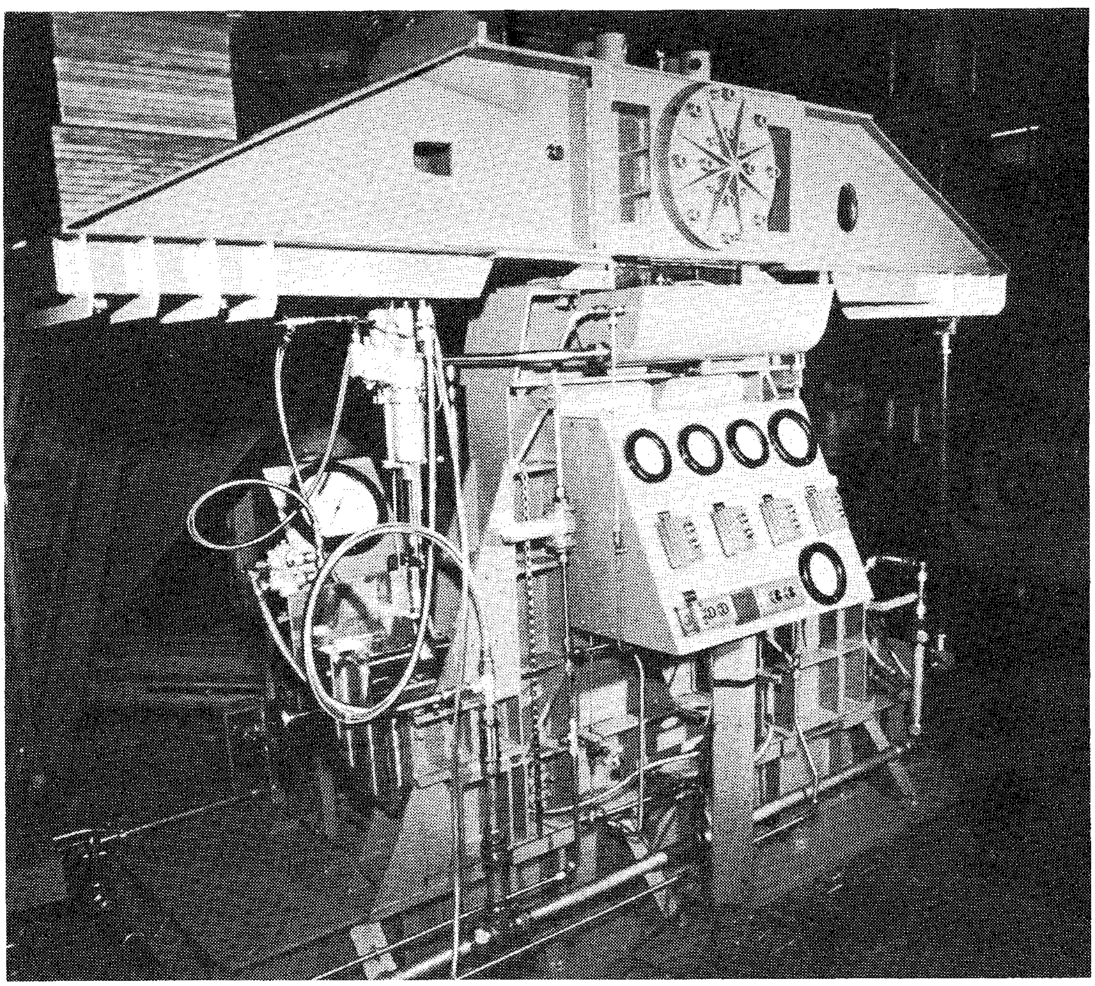

Fig. 20 Torque load simulator with hydraulic actuator for Jupiter missile.

time actuators became a necessity. The development of the actuators, including load simulators for closed-loop testing, was contracted to the Franklin Institute, Philadelphia, Pa. A very satisfactory development resulted. Figure 20 is a picture of the hydraulic actuator in the load simulator. The development was of benefit to the Jupiter as well as to the Thor missile.

Mass considerations were particularly important for the solid propellant two-stage Pershing missile. The author resolved that three vane controls could be used instead of the customary four vane control. The three vanes were symmetrically placed 120 deg from each other. The required superposition of the pitch, yaw, and roll signals was accomplished in the summation amplifier of the control computer. With a newly designed hydraulic actuator with integral motor pump, the control system proved to be most reliable and efficient.

When the powerful F-1 engine for the first stage of the Saturn launch vehicle was developed, no drive shaft was provided on the turbine for a high-pressure pump, nor was it possible to attach such a pump. Thus the hydraulic actuators for swiveling the engines had to be fed with pressurized fuel as the hydraulic medium. The pressure of only 1500 psi, in comparison to a more efficient pressure of at least 3000 psi, caused voluminous actuators with three-stage servovalves, possessing the required low-response time. Since position control had to be achieved, a mechanical feedback to the hydraulic valve's first stage torque motor was selected for reliability reasons. In the competitive development of the actuator both contractors, Hydraulic Research and Moog Inc., were equally successful and both actuators were interchangeably used.

Simulation and Analytical Tools

For several years after World War II electronic analog computers were not commercially available. Thus the V-2 motion simulator was reactivated and a larger, more flexible analog computer had to be developed.[9] This ac analog computer, developed by H. Hosenthien and O. Hirschler, became operational in 1950. It had the following features. Like the V-2 motion simulator, it used suppressed carrier amplitude modulated computer signals and a multicam time-function generator to simulate time-varying coefficients. In contrast to the fixed-wiring, special purpose type V-2 motion simulator, increased flexibility was obtained as in modern analog computers. On the front panel of the computer the individual computer elements, such as summing amplifiers, integrators, and cam-function variables could be interconnected for the various simulation tasks. Equations could be separated and reconnected by a push-button relay system for calibration and testing. Improved summing amplifiers used resistor networks with negative feedback amplification and possessed variable gain adjustment for coefficient setting. The ac integrator was modified to use the integrating capacitor in the negative feedback loop of the ac amplifier; the integrating capacitors were electronically switched at carrier frequency. This analog computer was most helpful as an analytical tool for the following decade and competed very well with the dc analog computers that became commercially available after the drift-free, chopper-stabilized dc amplifier was invented by E.A. Goldberg.

The design of a cost-effective and power-efficient control system has to permit a more or less nonlinear system. To prevent expensive overdesign linked with a linear system as well as underdesign with a very marginal performance, analytical means are normally insufficient for the performance investigation. Therefore, analog computers have been used in simulating the vehicle dynamics with the actual nonlinear hardware completing the control loop. The performance of the control system is then judged by observing the dynamic response to simulated perturbations. A complete investigation has to include all the tolerance combinations for the expected coefficient variations. In such a situation it is most helpful to determine the stability area of the control system for varying coefficients. The author developed methods to find the stability limits with a minimum of analog computer measurements, thus enabling the control engineer to judge the limits of his control system.[14] The use of these stability limits and the enclosed stability areas has been very beneficial in the stability investigations of rocket control systems; the method preceded the D-decomposition techniques developed later.[15]

Concluding Remarks

The development of inertial navigation, guidance, and control systems and techniques, leading to accuracy and reliability performances unpredictable in the beginning, has been most successful, especially in view of the extreme environmental conditions to which its components are subjected. Acknowledgment for the accomplishments must be given to the many laboratory engineers and scientists who contributed enthusiastically in a team effort, often without a personal recognition. In the same way credit must be given to the supporting industry as well as to competing institutions, in the foreground the Charles Stark Draper Laboratory, who, through their excellent, sometimes parallel development work and open-mindedness, caused the stimulation to achieve the utmost.

Acknowledgments

The author acknowledges especially the contributions and valuable comments to this report by Dr. S. M. Seltzer, by O. A. Hoberg, and by H. H. Hosenthien.

References

The following references have been kept to a minimum; most of them list numerous additional references.

[1] Farrior, J.S., "Inertial Guidance, Its Evolution and Future Potential," *From Peenemuende to Outer Space,* NASA Marshall Space Flight Center, March 23, 1962.

[2] Mueller, F.K., "A History of Inertial Guidance," Army Ballistic Missile Agency, Redstone Arsenal, Ala., 1960.

[3] Mueller, O., "The Control System of the V-2, History of German Guided Missile Development," edited by Th. Benecke and A.W. Quick, AGARD Seminar, Munich, Federal Republic of Germany, 1956.

[4] Hoelzer, H., "Guidance and Control Symposium," AAS Vol. 43, *Science and Technology,* 1976, pp. 301-316.

[5] Haeussermann, W., "On the Evolution of Rocket Navigation, Guidance and Control Systems," AAS Vol. 43, *Science and Technology,* 1976, pp. 258-269.

[6] Haeussermann, W. and Duncan, R.C., "Status of Guidance and Control Methods, Instrumentation, and Techniques as Applied in the Apollo Project," Lecture Series on Orbit Optimization and Advanced Guidance Instrumentation, AGARD, Duesseldorf, Federal Republic of Germany, Oct. 1964.

[7] Haeussermann, W., "The Delta Minimum Inertial Guidance Scheme," 13th Annual Meeting of the American Rocket Society, New York, N.Y., Nov. 1958

[8] Pickering, W.H., "History of the Cluster System," *From Peenemuende to Outer Space,* NASA Marshall Space Flight Center, March 23, 1962.

[9] Hosenthien, H.H. and Boehm, J., "Flight Simulation of Rockets and Spacecraft," *From Peenemuende to Outer Space,* NASA Marshall Space Flight Center, March 23, 1962.

[10] Haeussermann, W., "Spatial Attitude Control of a Spinning Cluster," *ARS Journal,* Vol. 29, Jan. 1959, pp. 56-58.

[11] Haeussermann, W., "Saturn Launch Vehicle's Navigation, Guidance and Control System," *Automatica,* Vol. 7, Sept. 1971, pp. 537-556.

[12] Chandler, D.C., Horn, H.J., and Martin, D.T., "An Iterative Guidance Scheme and Its Application to Lunar Landing," NASA Marshal Space Flight Center, MTP-Aero, 63-11, Feb. 6, 1963.

[13] Faget, M.A., "The Evolution of Flight Control of the Apollo Mission," AAS Vol. 45, *Science and Technology,* 1977, pp. 324-330.

[14] Haeussermann, W., "Stability Areas of Missile Control Systems," *Jet Propulsion,* Vol. 27, July 1957, pp. 787-795.

[15] Asner, B. and Seltzer, S.M., "Parameter Plane Analysis for Large Flexible Spacecraft," *Journal of Guidance and Control,* Vol. 4, May-June 1981, pp. 284-290 (this issue).