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DEVELOPMENT OF THE GERMAN A-4 GUIDANCE AND CONTROL SYSTEM

1939-1945: A MEMOIR[†]

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INTRODUCTION

The guidance of launch vehicles and spacecraft can be subdivided into two partial objectives: a) stabilize the spacecraft in the direction of its flight, and b) carry out the navigation or flight control function enabling spacecraft to reach their destination. These also make possible the navigation, guidance and flight control of conventional airliners, so it is no wonder that in 1939 the first rocket guidance systems corresponded closely to the techniques of airplane flight control modes and the inertial stabilization systems of ships. During the flight of missiles, however, flight regions are reached where the air density becomes too low to provide strong enough aerodynamic forces and moments for stabilization and flight path damping. Therefore, artificial damping modes have to be introduced to compensate for the imperfections of the guidance components, of radio signal disturbances and related complications, as well as the flight path variations they cause. This necessity led to the introduction of reaction control systems and electronic guidance components which could better provide the needed additional corrections.

With the increase in flight altitude above the Earth's surface, and of the presence of gravimetric anomalies in the gravitational field of the Earth, however, these measurements are insufficient. They require the introduction of a reference system independent of such anomalies to secure the exact definition of the flight path, and the instantaneous, true position of a launch vehicle or spacecraft, respectively. Thus, one must either introduce an inertial measuring system or apply precise and reliable ground tracking of the spacecraft or launch vehicle position, or combine both of these methods.

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The first candidate for the solution of this problem involved further development of gyro stabilization techniques to obtain a three-axial reference system that is space-fixed and not dependant on continuous tracking of the projectile or spacecraft position, or, in other words, is independant of the gravity-field variations. Gyros, by themselves, cannot accomplish this. One has also to be able to follow the state of the motion of the reference system in the spacecraft in relation to its three inertial motion axes and to its three rotational axes, in order to be able to describe precisely the trajectory or flight path of the launch vehicle or spacecraft. This can be accomplished either with body-fixed gyros installed in the projectile to measure rotations and rotational rates of the reference axes of the vehicle—as this is done conventionally in airplanes—or with equipment to monitor the motions in the directions of the body-fixed axes with integrating accelerometers that by two fold integration of the acceleration, describe the respective distances travelled in the directions of the axes inertial axes. Because, at the same time with these motions, rotations can take place, and the determination of the actual trajectory becomes very complicated and requires extremely precise calculations. Only during the past decade has this been satisfactorily achieved by on-board digital computers, without adding too much to the weight of ballistic and spacecraft vehicles. However, where the control system prevents the rotation of two of the axes, and the rotation of the third axis amounts to less than 180° , the task of the control system to keep the deviation for the two lateral axes to a minimum, for ballistic missiles particularly, a substantial simplification can be achieved.

This last described technique can be applied to preprogrammed ballistic missile and launch vehicle trajectories, and can perform the corrections for indicated deviations from the standard flight path after the separation from the launch vehicle using the spacecraft inertial guidance system. Russian launch vehicles, as well as the Saturn system, apply this solution. The Saturn system has a triple gimballed inertial platform, on which, besides stabilizing gyros, three velocimeters with integrators also operate. In the case of the Saturn system, the gimbal-positions are measured with digital pickups and reported to the digital launch vehicle control computer. This system makes position and trajectory guidance available from the same reference system. But for the present, let us return to the history of these systems.

The necessity to create a new reference system fixed in space and independent of Earth reference points caused Professor Schuler, in Goettingen, for the first time to propose a stabilized ship gyro platform, to guide ships for long periods, independent of the motion of the sea and also during darkness and absence of visibility of the stars, on a constant, precalculated course. Such ship gyros were developed according to Schuler's proposals, under the direction of Captain Boykow, by the Krieselgeraete A.G. in Berlin, and introduced into maritime navigation. Since the gyro retains its rotational plane

precisely, the larger the ratio of the gyro impulse to the disturbing impulse the better, and so very heavy gyros with high impulses were used for the stabilization of the ship gyro compass equipment.

In the 1920s Professor Schuler also developed a theory explaining that a gyro pendulum is independent of accelerations affecting its suspension point when the imaginary length of the pendulum corresponds to the Earth radius. Today such a pendulum is called a "Schuler tuned pendulum." The period of such a pendulum is 84 minutes, and for many years the Schuler pendulum technique has found application in the inertial navigation of airplanes. However, to arrive at the expected guidance precision with weights acceptable for flight instruments, the original ship gyro units had to be made much smaller, and their drift rates also had to be reduced significantly. This effort took many years and led to the replacement of the ball bearings for the gimbal supports of the gyros in their sensitivity axis by liquid suspension or gas bearings, with very refined premium ball bearings used in the less critical applications. It was Boykow who suggested the use of accelerometers for air navigation. Tests with such gyro stabilizers combined with accelerometers began in 1930, and were flight tested later.

But large navigation deviations were observed during these tests because two-fold integration of even very small residuals in the measurement of the acceleration-vector led very rapidly to large error readings of the integrals, and the gyros available between 1930 and 1940 were by no means ready for this measuring objective. However, investigators realized that as soon as the quality of the gyro stabilization and accelerometers improved, this method would become very promising. This disadvantage, however, appeared much less severe for applications in launch vehicles and ballistic missiles because they possessed much shorter propulsion and flight durations. It appeared to be the way to become completely or partly independent of radio tracking and electronic navigation aids, since these often showed anomalous operating conditions and could also be influenced by electronic noise or even countermeasures.

In 1939 aircraft navigation employed compass-supervised directional and azimuth gyros (or artificial gyro horizons)—"supervised" by an erection pendulum. They made instrument flying possible, but were too inaccurate for navigating long flights unless the actual flight path could be "updated" through radio bearings. Continuous monitoring of the gimbal angles of the horizon platform by the Earth's gravity vector makes the gyro platform independent of the Earth's rotation since the vertical reference vector is always again restored by the erection pendula. This technique, however, is also not applicable in satellites or spacecraft moving under physically weightless conditions, since the gravitational force is just compensated by the centrifugal force, voiding the supervisory effect. On the other hand, such a platform will retain an inertial orientation or "space-fixed axis," if drift rates can be compensated or remain within adequate limits.

THE CONTRIBUTION OF PEENEMÜNDE

Requirements for the A-4 Flight Control System

When the first ballistic missiles were developed at Peenemuende in Pommerania, inertial guidance systems were not far enough advanced to take over the guidance task of the A-4 missiles. However, the powered trajectories of these missiles lasted only sixty-seven seconds, as contrasted with launch vehicles designed to deliver satellites or spacecraft, and so made the use of inertial guidance components less problematic. The powered portions of the flight were subject to high accelerations, compounding the problem of systems errors, but the short time of powered flight in terms of error build-up for first and second integrals of the acceleration made these effects less severe. Therefore, it appeared possible to use accelerometers in an inertial reference system, permitting ballistic missiles to become independent of fixed radio guidance stations.

The A-4 guidance system established the burn-out location of the A-4 powerplant at about 38-km altitude. The altitude tolerance as well as the velocity vector tolerance had to be small enough to limit the dispersion of the impact location. We sought to fly the missile along a precomputed optimum trajectory without guidance and control errors, allowing impact at the desired range and direction (ballistic trajectory). The accuracies of the components of the guidance equipment and the thrust dispersions of the powerplant, including weather anomalies, led to a three Sigma dispersion of less than one tenth of a percent of the range for the various guidance and propulsion cut-off techniques under consideration. In practice errors do exist, and the propulsion schedule is not perfectly maintained. The accuracy of the powerplant cut-off velocity depends on the absolute thrust level of the rocket engine, of the closing velocity of the fuel inlet valves, of the hydraulic resistance of the fuel lines and injection nozzles between the valves and the combustion chamber, and also of the hydraulic resistance of the cooling jacket of the rocket engine, to name a few of the more important contributors to thrust dispersions. These factors, besides nozzles exit resistance, mainly influence the execution time-delay of the cut-off signal from the burnout velocity control subsystem of the guidance and navigation system to the powerplant of the missile, and make thrust decay a function of time. The latter therefore is subject to dispersions. To this dispersion one must add the value of the uncertainty of velocity and altitude determination of the cut-off point, caused by cumulative errors of the control, guidance, and navigation system itself.

To assess the magnitude of the initial errors, one has to establish an error budget within which the various potential error source ranges have to be listed and confirmed by tests. The dispersion tolerances can be established by mechanical improvements and selective groupings of interacting subsystems. In the case of the A-4 the development of new fuel valves of extremely short reaction times achieved the desired minimum

dispersion of the burnout velocity. We effectively subdivided the velocity cut-off process into two stages: with a pre-cutoff signal, the thrust was reduced from 30 to 8 tons, resulting in a small residual longitudinal acceleration, causing the missile to "creep" towards the normal cut-off velocity. This signal was followed by the main cut-off signal which entirely terminated the thrust. The acceleration then went through zero, and the altitude and direction of the velocity vector assumed a mean tail-off value. All the cut-off errors remaining at this point would retain the impact point including wind dispersions during the reentry of the projectile within the acceptable dispersion range ($\sim .1\%$ of range for longitudinal and lateral displacement). This requirement still exists today for the delivery of spacecraft or payloads into a chosen orbit, but the required tolerances needed to meet the desired target accuracy of the ephemeris data would have to become still much smaller.

To solve the cut-off problem at Peenemuende we sought to improve the weight of maritime gyro platforms, and of artificial horizons and directional gyros of aircraft. The latter consisted of gyros of two degrees of freedom. There were no operational accelerometers possessing the required accuracies to meet the A-4 tolerance objectives. Considering the accelerations in powered flight, the gyro horizons produced the desired accuracies only if the erection pendula were activated, while the resulting gyro drift rates become intolerably high whenever the erection system was turned off. These methods therefore required modifications to eliminate the described disadvantages and permit the maintenance of the local vertical automatically in inertial space.

Other methods to accomplish the task involved radio tracking and radio guidance methods. They allowed flight velocity and direction to be continuously measured and updated. Theoretical investigations showed that the radius vector velocity parallel to the burnout point tangent on the flight path, measured by the doppler velocity between a transmitted signal and its return from the missile, would accomplish this. A signal transmitter on the ground, a highly stable transmitter of equal frequency on board, synchronized before take-off, achieved this. Use of this technique required a frequency bridge of a low time constant and high precision to obtain the necessary accuracy without intolerable signal infidelity. High measuring accuracy and extremely low time constants are normally diametrically opposing qualities. In the case of achievement of this rather difficult objective, the "beat-frequency" between the two signal sources would be the velocity vector from the ground transmitter to the flying missile. Further analysis showed that the two transmitter solution required a high degree of frequency stability between the two transmitters to make this method workable. However, a solution, multiplying the frequency received on the ground and retransmitting on the new frequency, while the ground-transmitter compared the signal returned with the ground multiplied frequency, achieved its objective. In this case, the otherwise unavoidable frequency drift between two transmitters was avoided and this method was adopted in practice.

This method did not permit the determination of the actual altitude or of the lateral displacement of the rocket vehicle. Additional range measuring equipment, placed in direction of the flight path plane and lateral to it, was needed to provide this data. Use of radio-navigation equipment would also make it possible to measure the deviation from the flight path plane, and, if combined with range measurement (as with a radar), provide also the approximate position and velocity in space. Additional gyros would therefore permit improvement of all measured data by the introduction of corrections.

Development of the A-4 Flight Control and Navigation System

In fall of 1939, we decided to begin development of several A-4 flight path determination and monitoring techniques based on the use of gyro and inertial references as well as of electronic ranging and tracking equipment, using concurrent single and multiple integration of flight path accelerations to compute instantaneous velocity and range vectors. We chose this approach because it was not obvious which of the considered methods or combinations of methods would be successful, particularly since the radio methods could be very much affected in their accuracy by intentional electronic countermeasures.

To plan this work on a broad basis, a number of outstanding German and Austrian university professors were invited to participate. They attended lectures concerning the flight and operational environment in which the "measuring equipment" would have to work, and observed fullscale powerplant tests on test stands. The professors unanimously concluded that the announced task to measure vehicle position and instantaneous velocity vectors accurate enough to achieve a dispersion not exceeding the value of one-tenth of one percent of the range was clearly outside the state of the art of that time, and was, therefore, impossible. However, we asked these professors to reexamine the problem.

Eventually five of the originally invited professors returned after three months and proposed solutions to the problem. These solutions were fundamental to this problem and still today belong to the methods recognized as superior for space guidance and navigation applications. This can be termed a surprising turn of events when compared with the outcome of the deliberations and conclusions of the first session. The progress made since 1939 is represented by the improvement of the achievable accuracy or reduction in dispersion for gyro subsystems and integrating accelerometers by a factor from 10^3 to 10^4 , while radio guidance and inertial navigation methods proposed even today can be met or surpassed only by use of atomic frequency standards and with digital computers of high accuracy. Some of these very pessimistic professors, as time went on, became enthusiastic participants in the search for solutions to problems which the young space guidance and navigation discipline was facing in 1939. Their contributions will be discussed later in some further detail.

As indicated earlier, accelerometers of rather high measurement precision, as well as angular rate and angular acceleration indicators which permit integration of their outputs, have played an increasingly important role in space navigation tasks. These have contributed substantially to the subsequent increase in achievable accuracy and the simplification of the equipment used for these purposes. All integrations, even if their initial errors affecting the integration constants are very small, can grow to very large numbers over time. Thus, high measurement precision and component quality have to be a major design goal. In 1930 an attempt was made to navigate an aircraft equipped with a gyro-stabilized platform and mechanically integrating accelerometers mounted on it. The flight, which departed from Berlin-Adlershof, was discontinued and the attempt terminated after three hours of flying time when the aircraft equipment indicated a position somewhere in Australia, while visual observations confirmed the aircraft position to be at the western border of Germany, near Holland. The reason for this very substantial discrepancy lay in the lack of precision equipment.

The lack of knowledge about desirable subsystem properties for the design of such navigation subsystems to achieve the desired accuracy prevented the establishment of adequate design criteria. Only after 30 years of superior and costly development efforts would the state of guidance technology advance far enough to provide commercial aircraft the ability to navigate without radio aids over thousands of kilometers and flight durations of seven to ten hours, with an average position uncertainty of only one to two kilometers. From this example one can judge the developmental tasks we faced in 1939, and why the professors hesitated to associate themselves with such an apparently hopeless endeavor that only could be attempted with youthful optimism.

TECHNICAL SOLUTIONS TO THE SPACE NAVIGATION PROBLEM AND THEIR MECHANIZATIONS

Near the Earth's surface, using the local vertical of the gravitational field, the knowledge of orientation of the horizon, and the geographic North direction and an altimeter, one can determine aircraft position accurately. This also can be accomplished with gyros providing an artificial horizon, an accurate altimeter, and a gyro compass monitored by a magnetic remote compass, or by using a gyro stabilized inertial platform and applying again gravity and the magnetic North tie-in to correct or compensate for gyro drift, and the altitude above sea level. These techniques provide aircraft location and its vectors of motion with, for most purposes, adequate accuracy.

For greater distances from the Earth's surface, without access to accurate surface fixes and horizon references, continuous accurate navigation of a spacecraft and determination of its state of motions is only possible with additional well defined and accessible references, such as accurate orientation of the Earth horizon, or knowledge of the physical center of the Earth, of directions to the center of the Sun, and/or the Moon

or remote stars or planets besides using inertial reference equipment. Therefore, in the case of the A-4, the German Rocket Research Center developed in parallel more or less sophisticated inertial and body fixed gyro references using singly or multiple integrating linear and angular accelerometers, as well as more sophisticated stationary and mobile radio navigation, radio beam- or plane-rider navigation systems, to monitor the flight control system of this ballistic missile. This was done to obtain two to three independent methods, using different physical concepts and a range of fabrication precision and costs, that could be optimized for a variety of navigation objectives.

The first group of this equipment combined an aircraft autopilot with fixed and mobile radio ranging and range-rate equipment. It used the radio position and velocity fixes to correct flight path deviations and permitted, with the aid of an inertial angular rate measurement, successful flight path stabilization. Another method, independent of radio navigation aids, consisted of strapped-down, airframe fixed, directional and rate integrating gyros, combined with accelerometers, integrating fully or partially the readings in all six degrees of motion states, and thus stabilized the flight path and rotational axes of the ballistic missile. By computing the burnout velocity of the rocket engine, referenced to a standard trajectory that combined target parameter tolerances which statistically averaged to errors of less than .1% of the range, we achieved this objective. Use of a standard trajectory permitted correction of deviations based on standard values only, and can be optimized so that the effect of the deviations is minimized.

The solutions to the standard trajectory deviations were not abstract (mathematically rigid), and we expected that the burnout errors, and consequently the target dispersion errors, would be large. Against all expectations, however, these simplifications of flight control and navigation equations permitted a closer and more accurate flight path control than would have been expected of these simplified control and navigation methods. The magnitude of the deviations was clearly a function of the thrust deviation from the standard engine thrust versus altitude relation, caused mainly by the degree of match or mis-match of the powerplant subsystems whose values are dependent upon calibration and quality-control, and subject of component matching and consideration of cumulative tolerances. Careful matching and selection of components influenced such deviations favorably.

The third kind of navigation and flight control system used a highly precise (for 1943) three-axis stabilized platform with two longitudinal accelerometers and one lateral accelerometer. The readouts were integrated once, and of one twice, to yield a powerplant cut-off velocity that approximately compensated for thrust deviations and burnout altitude errors or burning time deviations, respectively. In a sense, since the two longitudinal accelerometers integrated along two different axes slanted in the

vertical trajectory plane, these vertical and horizontal velocity components actually had their greatest sensitivities based on a desired burnout trajectory tangent.

The flight path of an A-4 equipped with this fully inertial navigation system thus used a standard trajectory. The pitch deflection followed a highly precise pre-computed time sequence which, for the nominal range, reached a burnout deflection of 42° , a value somewhat range dependent. To achieve this deflection, a precision timer rotated the base of the pitch-angle pickup so that its zero direction at all times pointed in the first approximation in the direction of the desired flight path tangent, reduced by the instantaneous angle of attack in pitch. To determine the proper time of burnout, one of the accelerometers was mounted space-fixed on the platform. Its measuring axis at take-off was parallel to the precomputed burnout tangent, and remained so throughout the powered flight. The indicated first integral of its acceleration resulted in a velocity—which was characteristic of the burnout velocity even if the burnout altitude was not correct and would lead to a shorter or longer range—depending on whether the burnout altitude deviation was negative or positive.

The second accelerometer was inclined forward so that its axis at the launch table pointed to the nominal burnout point in space. Its time-related output was integrated twice and represented a distance characteristic for the location of the burnout point in x- and z- directions. If this point in time was reached later than at the nominal burnout time and the velocity indication was approximately correct, the missile flew a steeper trajectory and, if uncorrected, overshot its target. The burnout velocity of the first accelerometer therefore had to be reduced by an approximate amount to yield the proper range. Therefore a combination of the two accelerometer read-outs after their respective integration yielded the required corrections. In reality, this case was somewhat more complex than described since wind effects deviation of its inflight mass and variations of the drag of the missile, besides azimuth angle and thrust deviations, affect the result (effect of Earth rotation versus time of missile flight). The lateral accelerometer also compensated for lateral wind effects and dynamic oscillations of the missile due to gusts and lack of aerodynamic damping, by using the second integral of the lateral deviation to keep the lateral deviations as small as possible and, using the third integral, to compute a "crab angle" to compensate for wind effects, aerodynamic unsymmetries, etc. in the lower portion of the trajectory.

The sensitivity axes of the gyros normally were equipped with ball bearings elected for the lowest friction. To reduce the sensitivity axis friction even further, our two main contractors for these gyros chose fluid suspended bearings (Siemens) and gas bearings (Kreiselgeraete Gm. b. H.) to accomplish this objective, and applied it also for the integrating accelerometers. With the exception of a Northseeking Gyro, however, these developments did not reach serial production before the end of World War II (But the Northseeking Air Bearing Gyro was used during the war for artillery purposes.) Since

the burning period for the A-4 missile amounted only to 67 seconds, the deviations of the integrating accelerometers were quite small despite the high "g"-levels of the trajectory (≤ 5 g), and almost achieved their design goals, but their reliability, including that of the control electronics, was not as satisfactory as we desired.

As an alternate to the third method, lateral deviation measurements from a radio guide plane were employed together with the gyro stabilized inertial platform system to reduce lateral dispersion. Besides the displacement itself, this arrangement also used the first and second derivative of the lateral displacement as well as the integral of the lateral displacement to compensate for wind effects and aerodynamic unsymmetries of the yaw plane. In addition, this system employed the all electronic analog flight computer (Mischgeraet or mixing computer), flight tested in aircraft before deployment in missile firings. We found that the natural frequency of the yaw response could be significantly increased, with a concurrent increase of the yaw damping, by giving lateral displacement, lateral displacement rate, and the rate of change of the lateral displacement into the roll channel of the flight control system. This latter technique was not applicable to the A-4, but would have been of great value for the winged A-8 vehicle. Since only two of these were flown without flight navigation, this technique was only applied to aircraft. The stability and effectiveness of this technique was amazing even at small distances from the guidance transmitter; the aircraft could fly stable directly over the transmitter at low altitudes without changing the gain-settings, optimum for the far field of the guidance plane.

Before lift-off of the A-4 missile, the gyro platform was erected by two pendula, a procedure which took between 20-to-30 minutes to satisfactorily level the platform. The erection system was switched off just prior to lift-off. At the same time, the displacement signal channels, zeroed out to prevent too early deflection of the control surfaces at lift-off, were switched on and the gain factors increased to their optimum value, which varied with the location on the flight path. All gyro, accelerometer, and integrator read-outs were approximately multiplied by the respective gain settings and mixed into the corresponding control axis channels by a fully electronic analog flight computer, and the signal outputs transmitted by wire to the servomotors at the jet vanes and aerodynamic control surfaces. The "fly by wire" technique was known already at that time, and was also applied to one of the two test aircraft used to develop all the electronic flight control techniques for the A-4. On the second aircraft, the control deflections were introduced thru a clutch system to the normal push rod linkages of the autopilot, which separated the hand controls in case of use of the autopilot. Both hydraulic and electric servomotors were used. By means of electric feedback of the control surface deflections and use of low inertia armatures (rotors) in the electric motors, the time constants of two types of electric servomotors could be reduced to four milliseconds, which was adequate to retain the control response frequencies above bending and torsional flutter modes of the missile structure.

Also at that time, magnetic amplifiers with feedback were developed that were unaffected by vibrations. They permitted, in several amplification stages, high amplification ratios, but avoided the sensitivity of electron tubes against acoustical and vibrational noise. A parallel effort to achieve this objective involved the semiconductor developments conducted by Professor Weise at the Technical University of Stuttgart, later the University of Michigan, that can be considered forerunners of the later transistors and diodes. I would go too far to describe all the individual efforts and solutions that advanced control and navigation techniques at that time, but among those who contributed were: the late Professor Leo Brand (1971+), also the late Professors Hueter, Kirschstein, Vieweg and Dr. H. Steuding, and also the still living Professors Buchhold, Busch, Fischel, Kramer, Carl Wagner and Wolman, as well as Dr. John Gievers, and my former colleagues Dipl. Ing. Josef Boehm (+), Dipl. Ing. Karl Bruetzel (+), Dr. Kurt H. Debus, Dr. Hans Friedrich (+), Dr. Ernst Geisler, Dr. Reinhold Schubert, Dipl. Ing. Werner K. Gengelbach, Dr. Hans Gruene, Dr. Walter Haussermann, Dr. Helmut Hoelzer, Dipl. Ing. Otto Hoberg, Dr. Joachim Muehlner, Dipl. Ing. Erich Neubert, Dr. Helmut Schlitt (+), Dipl. Ing. H. J. Gengelbach (+), Dr. Walter Schwidetzki, Dr. Ernst Stuhlinger and Dr. Hugo Woerdemann, a large number of whom became very prominent in the U.S. space effort, while others helped to usher in the European space effort. It was a true and remarkable effort.

All these men have to be recognized for their outstanding contributions and creative engineering achievements in their respective professional disciplines that included many pioneering engineering and scientific innovations of the highest caliber in the fields of flight control, inertial and radio guidance, mathematical modelling and simulation, and navigation technology. Many of these scientists continued their outstanding contributions to the science and technology of space flight into more recent times. On the American side, the progress achieved since World War II is based on the outstanding pioneering efforts of particularly Dr. Charles Stark Draper, Director of the MIT C.S. Draper Laboratories, and his close associates Prof. Walter Wrigley, Prof. R. H. Battin, Dr. David Hoag and the many outstanding members of this excellent Laboratory and others, who conceived and developed the extremely complex Apollo flight and navigation system.*

I would like to emphasize three particular technical solutions that have influenced even recent technical developments. In 1939-1940, Dr. Hoelzer, the Chief of the Flight Control and Guidance Division of the Peenemuende Rocket Research Center, developed a fully electronic analog computer that appears to have preceded all similar developments reported in the professional literature. Within a month it was applied to flight path simulation of ballistic missiles and ground-to-air missiles (the "Wasserfall" Anti-Aircraft Guided Rocket), as well as a training device for joy-stick control of optically guided missiles. During the same period it was adapted to serve as the all electronic

*See C. Stark Draper, "The Evolution of Aerospace Guidance Technology at the Massachusetts Institute of Technology, 1935-1951: A Memoir," in this Volume-Ed.

flight computer of the A-4, its main task. Two of these sets were converted to flight test units on two aircraft where responses to flight maneuvers were recorded. Based on the success of these units, we decided to use them as autopilots for these aircraft rather than the factory installed units. This decision led us next to superimpose on the autopilot inputs the signals of the radio guidance signals. Both applications turned out to be a great success.

The Kreiselgeraete Corporation developed a pendulous gyro accelerometer which integrated the acceleration produced by the thrust of the A-4 powerplant to the longitudinal velocity vector of the A-4. The numerical value of this accelerometer was used to cut off the powerplant at a predetermined value. Since the sea level thrust variation of these powerplants did affect the burnout velocity, it became necessary to apply burning time corrections to reduce the dispersion in the range of the ballistic missile. This was achieved by making a contact position of the velocity cut-off device variable with time, so that the integrator cut off at a lower velocity if it took longer to reach the preset speed. This integrator, a very simple device, showed in laboratory tests to be accurate in terms of a velocity uncertainty at burn-out, corresponding to .1% uncertainty in range dispersion, fuel valve dispersions included. Also, under flight conditions of the A-4, this device proved remarkably accurate.

A third invention by Professor Carl Wagner involved an electrolytic cell with silver-iodide serving as the electrolyte. It was used to integrate the electric current output of the Kreiselgeraete, Inc. and Buchhold accelerometers to the burnout velocity. Also, this cell was an extraordinarily simple integrator, remarkably accurate for the time period and the environment in which it was used. Prior to the takeoff, the cell was charged with a well defined small current, proportional to a small component of the gravitational field. By converting acceleration into a dc current we received an output proportional to the numerical value of the acceleration component. To reach the cutoff velocity specified in the firing tables, one just had to multiply the small component of the gravity to obtain the charging time necessary to store the cutoff-velocity in the cell, following the equation:

$$t = \frac{v_{\text{burnout}}}{0.1 \text{ g}} \quad \text{where } v_{\text{burnout}} = 1650 \text{ msec}^{-1} \text{ g} = 9.81 \text{ msec}^{-2}$$

$$t = \frac{1650}{0.981 \text{ g}} = 1682 \text{ sec. charging time}$$

Using a vertical dividing circle, accurate to one arcsec, a timing accuracy of .1 second, and a tilting angle of the accelerometer .60.0 degrees ± 1 arcsec, the acceleration would be integrated accurately to ± 4.22 x 10⁻⁶ msec⁻². Kreiselgeraete Inc. developed air bearing levels which could measure the horizontal plane tilt accurate to one

arc-second. Since automatic timing devices of much more than .1 second accuracy were common state of the art, the velocity uncertainty of the integration could be held to much lower values than indicated above. The charging process consisted of the electrolytic deposition of pure silver, removed from the electrolyte, on one of the electrodes. At take-off, the acceleration signal was simply reversed at the instant of lift-off and began to discharge the cell. At the instant of removal of all metallic silver from the cathode, the voltage across the terminals of the cell changed significantly and actuated the cut-off signal sequence. This integrator, except for the release, had no moving parts, was not sensitive to accelerations in the measuring direction, and proved remarkably reliable.

All power cutoff sequences at that time used a precutoff and a main cutoff command signal supplied by the flight control computer and its inertial subsystems. The precutoff signal closed the main fuel valves; a by-pass valve continued to furnish approximately 30% of the fuel, on account of residual drag at the altitude of cutoff and the direction of the flight path tangent at burnout, reducing the remaining acceleration to near zero. When the now slowly changing longitudinal velocity component reached the preset value, the main cutoff signal was activated, terminating all fuel flow. Of all the above quoted examples, the techniques introduced by Dr. Gievers and Dr. Hoelzer have found "permanent employment" in space technology. Thus, the technological advance of the state of the art emanating from the early ballistic missile developments at Peenemuende must occupy a prominent place in the history of rapid technological advancements of the twentieth century.