Nov. 25, 1969 L. J. CHU 3,480,233

MISSILE GUIDANCE METHOD AND APPARATUS

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FIG. 3

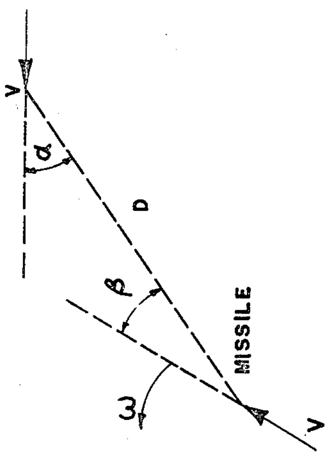


FIG. 4

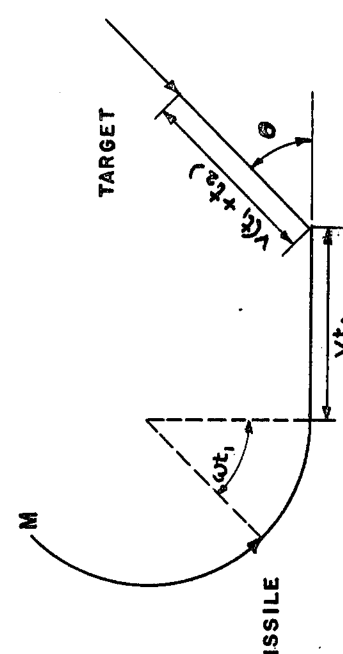


FIG. 2

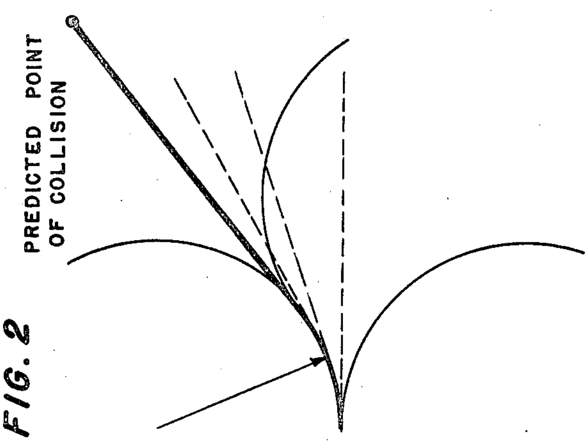
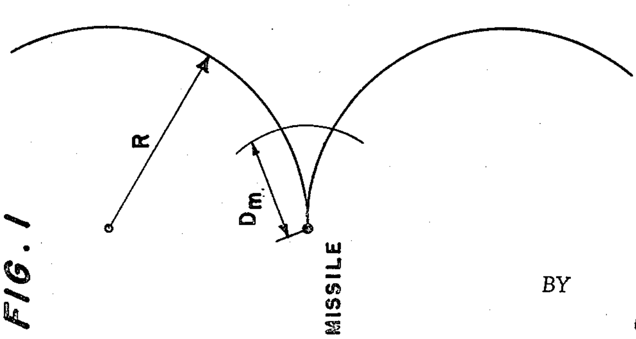


FIG. 1



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MISSILE GUIDANCE METHOD AND APPARATUS

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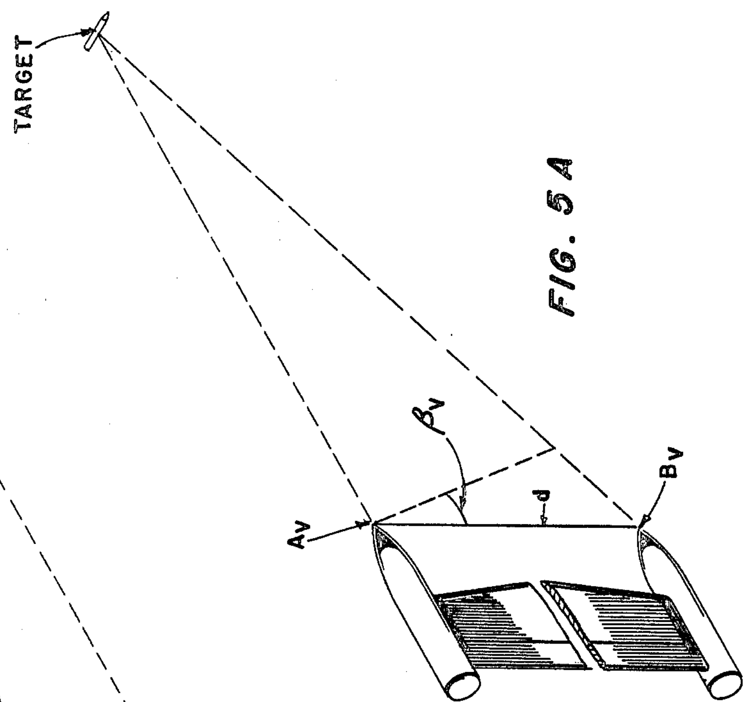

FIG. 5A

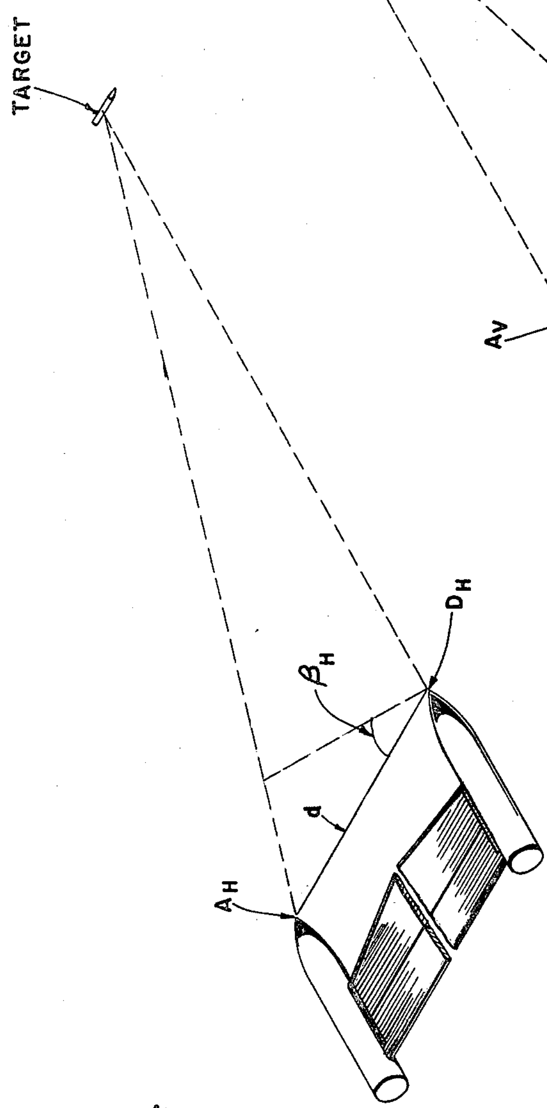

FIG. 5

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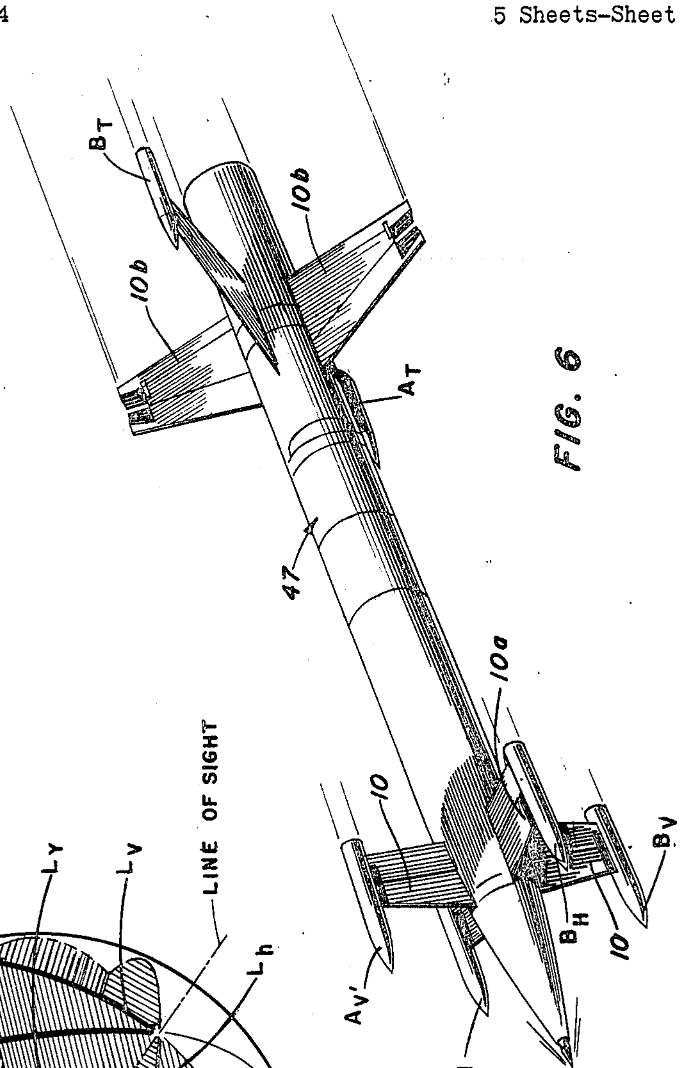

FIG. 6

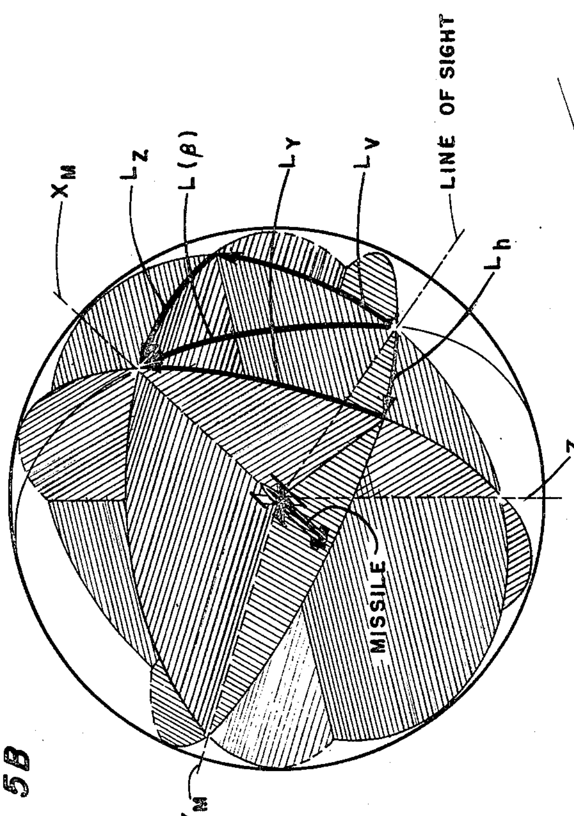

FIG. 5B

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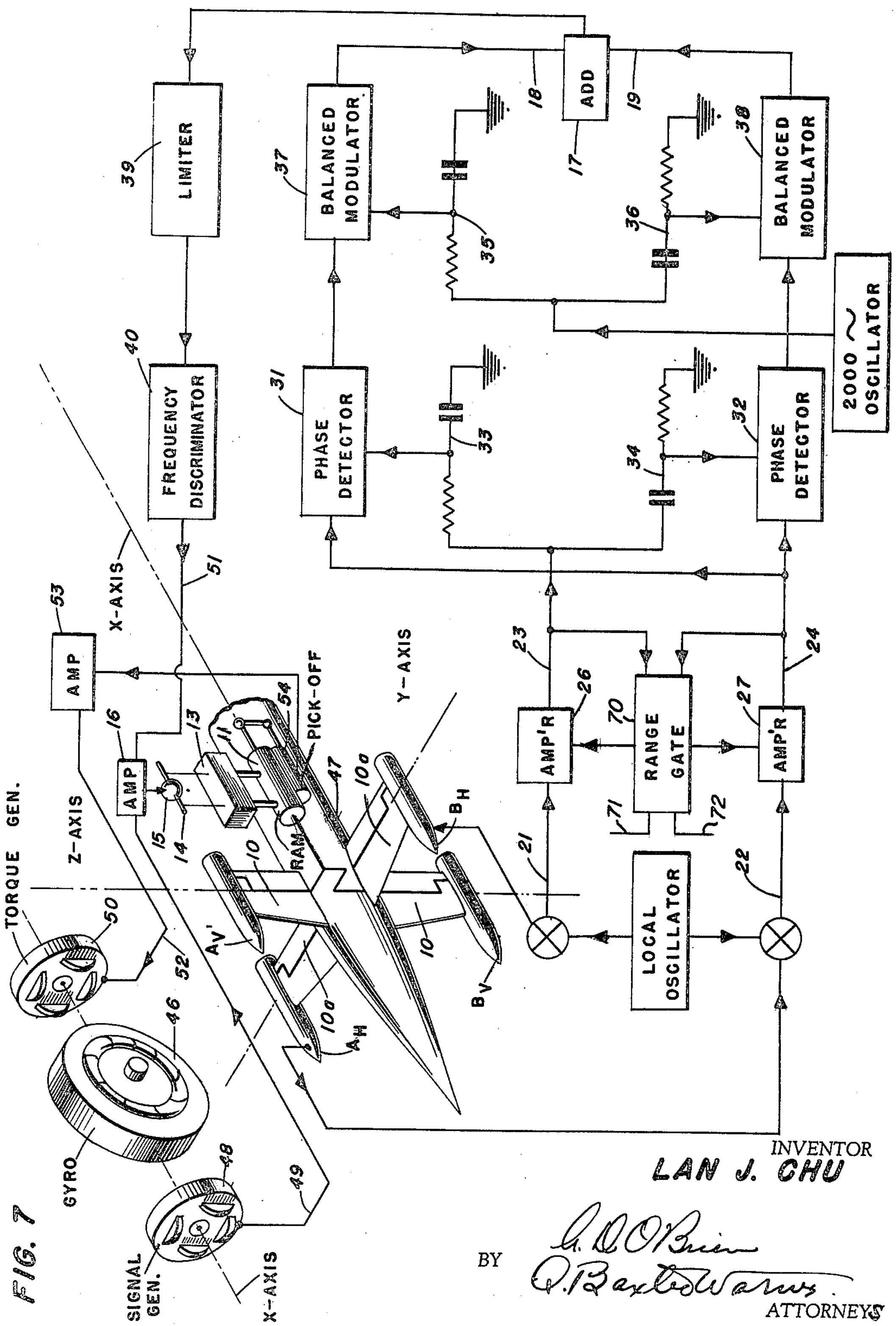

FIG. 7

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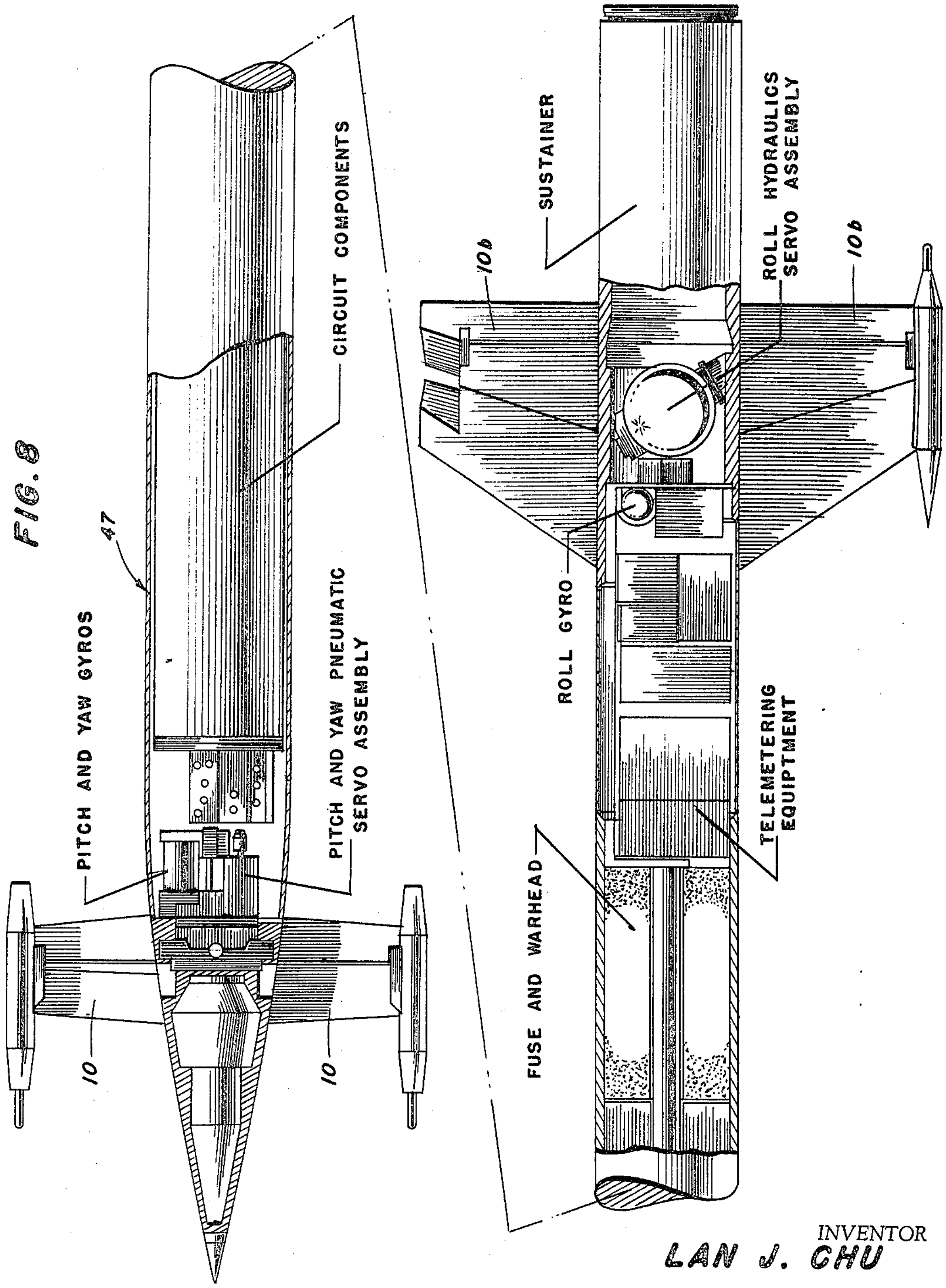

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MISSILE GUIDANCE METHOD AND APPARATUS

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Filed July 21, 1954, Ser. No. 444,931
Int. Cl. F41g 7/00; F42b 15/02; G06f 15/50
U.S. Cl. 244—3.18 6 Claims

ABSTRACT OF THE DISCLOSURE

The disclosure is directed to a missile guidance system incorporating radiant energy detection devices, which feed signal processing circuitry for phase differentiating of the signals in a manner to provide steering signals for application to the control surfaces of the missile for course changing to effect continuous pursuit of a target by the missile.

This invention relates to the guidance of missiles, and particularly to the problem of directing and maintaining a missile on a collision course with respect to a target traveling through space, which target possesses the capability of maneuvering to avoid collision with such a missile.

An object of the invention is to provide a novel method of guiding a missile toward a course being followed by a target maneuvering through space, the method disclosed being so devised as to cause the missile to continually change its course to conform to each successive deviation in the target's course.

Another object is to provide a guidable missile of novel construction, including steering apparatus responsive to phase differentiated radiant energy signals received from a detected target, and operable to change the course of the missile continually so long as the detected target continues evasive maneuvers.

A third object is to provide, as part of the novel guidance method above referred to, the step of applying lateral acceleration to a missile in response to variations in the rate of change in the significant characteristics of intelligence reflected from a detected target, the resulting lateral acceleration serving as a means to direct the missile to a "collision course," that is, a course most likely to bring about collision between the missile and the detected target, on the basis of the intelligence being concurrently received by the missile concerning the behavior of the target.

A fourth object is to provide, as part of the novel guidance method, the step of converting intercepted radiant energy into missile guiding information, by processes including the utilization of fixed antenna elements carried by the missile, which processes eliminate the need of actually computing the speed or relative direction of either the missile or the target.

A fifth object is to provide, as a specific part of the processes just referred to, the step of varying the lateral acceleration of the missile in accordance with the rate of change of phase difference between signal energy intercepted at opposite extremities of the fixed antenna array above referred to; said variation in lateral acceleration being brought about by directing said signal energy through a seeker mechanism adapted to convert said phase difference rate of change into electrical energizing impulses of appropriate polarity and magnitude to actuate a servo mechanism in the proper direction and to the proper degree required to cause the desired shift of externally mounted vanes, or steering surfaces, about the pitch and yaw axes, respectively, of the missile control system.

A sixth object is to provide a novel combination and relationship of actuating units for putting into practice the novel missile guidance method herein disclosed.

Other objects and many of the attendant advantages of this invention will be readily appreciated as the same becomes better understood by reference to the following detailed description when considered in connection with the accompanying drawings wherein:

FIG. 1 is a diagram indicating the circular course into which a missile is guided during the first stage of the novel method of operation herein disclosed;

FIG. 2 is a diagram illustrating both the first and the second stages in the missile's course to the target;

FIG. 3 is a diagram to assist in identifying certain symbols used in the description and claims;

FIG. 4 is a diagram illustrating the converging courses the missile and target may follow;

FIGS. 5, 5A, and 5B are diagrams illustrating the antenna and target relationships;

FIG. 6 is a perspective view of a missile embodying, and suitable for practice of, the invention;

FIG. 7 is a schematic representation of actuating and control components associated with one of the three control servos (namely, the yaw control servo) which operate conjointly to execute the method of control herein disclosed; and

FIG. 8 is a sectional view of the missile, showing the relative locations of the major components.

By way of introduction to the present invention, it should be pointed out that the performance of an unattended missile will necessarily be limited by the capacities inherent in the physical components carried by the missile for target-seeking, propelling and guiding purposes. The novel method of operation herein proposed is believed to be well adapted to such inherent limitations, in that it can be executed with a minimum of reliance upon factors of variable or unpredictable performance characteristics. In fact, the method can be carried out by utilization of simple steering mechanism controlled by reflected radar pulses, yet without the complications of rotating scanning antennas.

The use of fixed antennas, in lieu of the rotating scanning antennas of the prior art, is a major feature of the present invention, and is made possible only because the invention utilizes, as an underlying concept, a mathematical analysis which demonstrates that, at the instant of time when the angular velocity of the missile (ω) arrives at a value equal to the time rate of change in missile heading with respect to the missile-target line of sight, the missile is at that instant headed directly to the calculated point of collision with the target. In other words, when ω arrives at a value equal to $d\beta/dt$ (hereinafter written as $\dot{\beta}$), the missile is at that instant on a collision course with the target, β representing the angle between the heading of the target and the line joining the target and the missile and t representing time. Hence only the derivative of the missile angle and not the angle itself, need be supplied to the radar seeker circuits. This discovery makes possible the elimination of a scanning type of antenna. The steps leading to this mathematical conclusion are developed in the following analysis of the problem, which analysis explains how it is possible to cause the missile to direct itself to the collision point with the use of fixed rather than rotating antennas, and by reliance upon only one basic characteristic of the missile, namely, its ability to accelerate laterally to varying degrees, in response to mechanically induced shifting of externally mounted control surfaces. At this point it should be repeated, parenthetically, that this ability of the missile to accelerate laterally is a limited one; the extent of the lateral acceleration capacity being necessarily restricted by the mechanical design of the missile. The same is true of the other limiting quantity, namely, the missile's velocity; but the linear velocity of the missile is not readily controllable by automatic means, and, as above noted, the mathematical conclusions herein expressed make possible the development of a control system operating through a single variable exclusively, namely, the lateral acceleration of the missile.

The following analysis of the problem will further explain how it is possible to direct the missile to the calculated point of collision by varying only a single characteristic of the missile, namely, its ability to accelerate laterally.

Let V be the linear speed of the missile, and A*m* the maximum lateral acceleration that can be applied to the missile, (as by application of turning effort to the external control surfaces). If this maximum acceleration is applied at the point of detection, and maintained thereafter, the missile will follow a circular course of minimum radius R, where

$$R=\frac{V^2}{A_m} \tag{1}$$

the course proceeding either to the right or left, as shown in FIG. 1; the direction of turn depending upon whether the target is proceeding leftward or rightward from the missile-target axis.

In a three-dimensional view of the action, either circle shown in FIG. 1 will generate a tore ("doughnut") sweeping out a space whose width is twice the radius R, the axis of the tore being continuously shifted to maintain tangency with the shifting course of the missile, until the time is reached when the shifting axis points directly to the predicted point of collision (see FIG. 2). At that instant the missile's control system should discontinue turning force application, and should thereafter apply a constant bearing effort to maintain the missile on the collision axis.

Thus, the ideal course, as illustrated in FIG. 2 by the heavy line, has two stages, the first stage being a path of maximum curvature, and the second stage being a straight path tangential to the toroidal axis of the first path at that point thereon which represents the missile's position at the instant of time when the missile's angular velocity has increased sufficiently to equal its time rate of change in heading, with respect to the missile-target axis; that is, when

$$\omega-\dot{\beta}=0.$$

This course is the ideal one because it is the shortest possible course and (assuming constant missile velocity) places the point of collision on the toroidal axis of the turning missile at the earliest possible moment and hence at the greatest possible range, thereby assuring maximum striking probability. And as the "second stage" is of maximum length, the lateral acceleration requirements are minimized, so that less power need be accumulated for application to the control servos.

The control system that will steer the missile along the ideal course just described will now be considered. The following factors are involved:

V=the speed of the missile,
v=the speed of the target,
β=the angle between the heading of the missile and the line joining the target and the missile,
α=the angle between the heading of the target and the line joining the target and the missile,
D=the distance between the target and the missile,
A=the lateral acceleration of the missile, and
ω=the angular velocity of the missile (=A/V).

The following relationship obtains between these quantities:

$$\dot{\beta}=\omega+\frac{V\sin\beta}{D}-\frac{v\sin\alpha}{D} \tag{2}$$

where $\dot{\beta}$ is the time rate of change of β. The first term of the right-hand side is the rate of change of heading with respect to space. The second term is that caused by the linear motion of the missile with respect to the target. The third term is that due to the linear motion of the target with respect to the missile. The sum of the three gives the total rate of the change of heading with respect to the line joining the missile and the target. The angular velocity of the target comes into the picture as a second-order effect only.

Equation 2 can be rearranged as follows:

$$\omega-\dot{\beta}=\frac{v\sin\alpha-V\sin\beta}{D} \tag{3}$$

The term on the right-hand side involves factors, such as v and α, which are not under the direct control of the missile. If, however, it is set equal to zero,

$$v\sin\alpha-V\sin\beta=0 \tag{4}$$

it is the equation for a constant-bearing collision course. Each term is the common altitude of a triangle with angles equal to α and β and sides proportional to v and V, respectively. This is exactly what is needed for the second stage of the ideal missile course. However, as the proper value of β depends not only upon the velocity ratio but also upon the target angle α, the proper value cannot be determined by direct measurement.

When the right-hand side of Eq. 3 is not equal to zero, it gives a measure of the deviation from a collision course. It follows that

$$\omega-\dot{\beta} \tag{5}$$

also measures the deviation from the collision course. Both terms in (5) can be measured in the missile itself. The first, the angular velocity of the missile, can presumably be measured by a rate gyro. The second term is the rate of change of the missile angle and can be measured by the response of the radar seeker circuits to the phase differences impressed upon the spaced antennas. The entire quantity,

$$\omega-\dot{\beta}$$

is the rate of change of bearing with respect to a fixed line in the plane of maneuver.

Let it be assumed that the missile carries gyro mechanisms and radar detection means capable of measuring it gives a measure of hte deviation from a collision course.

$$\omega \text{ and } \dot{\beta}$$

respectively. At the instant the radar in the missile detects the target, and the quantities

$$\omega \text{ and } \dot{\beta}$$

are measured, the deviation from the collision course can immediately be determined qualitatively. If (5) is positive,

$$V\sin\beta<v\sin\alpha$$

and β should be increased so as to equalize the two terms. In other words, a positive $\dot{\beta}$ is needed. This can be accomplished by increasing ω as shown in Eq. 2. As mentioned hereinbefore, it is desired to have maximum acceleration during the first stage of the approach. If the missile is traveling along a straight line initially $\dot{\beta}$ will change, after the application of the acceleration, from a negative to a positive quantity. This maximum acceleration is to be maintained until (5) vanishes, at which instant the missile is heading along the collision course. Then

$$\omega=\dot{\beta}\neq 0$$

Since ω is not now equal to zero, the missile will overshoot towards the other side of the collision course, and expression (5) will become negative. A force is then applied through the servo to reverse the acceleration. With such a system, the missile will eventually oscillate about the collision course. The period of the oscillation is determined by the time lag present in transferring the information regarding

$$\omega-\dot{\beta}$$

from the radar system to the control mechanism, and the time it takes to reverse the acceleration. The shorter the period of oscillation the less the maximum deviation from the collision course.

If the target is maneuvering, the operation is not affected during the first stage of the approach. The missile is kept on the maximum lateral acceleration course until a collision course is reached based upon the values of α and β at that instant. After that, the lateral acceleration required for the missile is equal to the lateral acceleration of the target multiplied by a factor $\cos\alpha/\cos\beta$. The missile will oscillate about a course such that the average acceleration is equal to that required to compensate for the maneuvering of the target.

As a numerical example, reference is made to FIG. 4 in which the target is assumed to travel along a straight line.

Target velocity—1000 ft./sec.
Missile velocity—2000 ft./sec.
Maximum missile angular velocity—5°/sec.=87 mil./sec.
t_1—the time during which the missile travels along a circular course
t_2 the time during which the missile travels along a straight-line course
θ—the angle between the target and missile heading at the point of collision

The absolute value of

$$|\omega-\dot{\beta}|$$

will be about 2°/sec, assuming a value of

$$|\dot{\beta}|$$

of about 3°/sec, for the period before the missile gets on the collision course.

$$\text{For } t_1>0,\ \omega-\dot{\beta}$$

remains fairly constant and then approaches zero rapidly when the missile is near the collision course.

In order to reduce the amount of overshoot, the lateral acceleration must be reduced when the missile nears the collision course. This may be done by making the instantaneous angular velocity ω proportional to

$$\omega-\dot{\beta}.\ \text{The ratio of } \omega \text{ to } \omega-\dot{\beta}$$

must be several times unity for a fast moving target. When

$$\omega-\dot{\beta}$$

is large, the angular velocity of the missile is limited by the maximum available acceleration. When the missile is near the constant-bearing course, the angular velocity is reduced proportionally with

$$\omega-\dot{\beta}$$

The amount of overshoot can also be reduced by applying an acceleration that is positive or negative depending upon the sign of

$$k\omega-\dot{\beta}$$

instead of that of

$$\omega-\dot{\beta}$$

where k is a positive constant less than unity. When the missile is on the circular course, $\dot{\beta}$ has the same sign as ω but is numerically less than ω. As the missile approaches the constant-bearing course, $\dot{\beta}$ approaches ω as a limit. Therefore, if k is less than unity

$$k\omega-\dot{\beta}$$

will go through zero sooner than will

$$\omega-\dot{\beta}$$

The acceleration starts to change just before the missile reaches the collision course. This scheme, however, does not prevent the oscillation of the missile about the collision course, but it does reduce th eperiod and eventually damp out the oscillation. The proper choice of the constant k depends upon the time lag in the radar and servo system, and the time necessary to reverse the acceleration.

From the preceding discussion, it is obvious that the only information required of the radar is the rate of change of the angle β, that is, the "lead angle" between the missile heading and the line of sight to the target. Since the angle information is not essential for the pursuit of the target, an antenna system that is fixed with respect to the missile may be used, with a resultant simplification of the antenna and mount problem, for either a ramjet or rocket type of missile.

As shown in FIGS. 5, 5A and 6, four identical antennas are mounted symmetrically on the sides of the missile. The separation distance between each pair is approximately equal to the missile diameter plus the length of diametrically opposed wings, which distance is small compared to the distance from the missile to the target. When a signal from the target arrives at the missile, there is a phase difference of $(2\pi d/\lambda)\sin\beta$ between the two received signals at the antennas. As the angle β changes, the phase difference varies at a rate

$$\left(\frac{2\pi d}{\lambda}\cos\beta\right)\dot{\beta}$$

In other words, the two received signals will differ by a frequency

$$\left(\frac{d}{\lambda}\cos\beta\right)\dot{\beta}$$

which is proportional to $\dot{\beta}$. This frequency difference, which is a measure of the time rate of change in the angle β, determines the time interval during which the missile should be maintained in the circular path constituting the first stage of maneuvering, as heretofore explained. Therefore, the two out-of-phase signals received at vertical plane antenna elements A_v and B_v (and similarly as to those received at horizontal plane antenna elements A_h and B_h) need only be fed into suitable detection and phase timing circuits, superimposed upon a suitable carrier wave, and then frequency modulated, combined, and the resultant voltage selectively applied, at the selected frequency, and with proper amplification, to operate a "pitch" control servo unit (or yaw control servo unit, as the case may be). One such arrangement is illustrated in FIG. 7 and controls the swing of rudder surfaces **10—10** hingedly mounted on the vertical fins carrying the antenna elements A_v and B_v. In the embodiment here described there are three such servo units, one for "yaw," one for pitch, and one for roll control, that is, one servo for each of the three reference axes, X, Y and Z of FIGS. 5B and 7. The servo unit **11** for control of yaw, only, is shown in FIG. 7. The other two, not shown, are preferably of the same design, with each including a fluid pressure actuated piston or ram **11** to which hydraulic fluid is supplied from an accumulator (not shown) at a rate, and in a direction, determined by the movement of the control valve **13** operated by a crossbar **14** swinging about the axis of the shaft of a torque motor **15** to which is delivered energizing voltage whose direction and magnitude is controlled by push-pull amplifiers in an electronic unit represented by block **16** in FIG. 7.

The signals from horizontal antennae A_H and B_H are fed to circuitry for measuring their rate of change in phase and produce command signals for the yaw servo unit in response thereto. This circuitry is shown by block diagram in FIG. 7. The signals are each reduced to an intermediate frequency by a common locol oscillator and a pair of mixers. After separate amplification in amplifiers **26** and **27** the signals are compared in phase by applying the signal from one amplifier to one input of a pair of phase detectors **31** and **32** and from the other amplifier to another input of said phase detectors **31** and **32** through a pair of R.C. fixed phase shifters **33** and **34**. The respective outputs of phase detectors **31** and **32** will have a phase relation dependant on the instantaneous phase relation of the signals from antennae A_H and B_H. Said outputs are converted to a frequency modulated control signaled by applying each to a balanced modulator, **37** and **38**, and injecting a 2000 cycle signal in fixed phase relation to each modulator. The phase of the 2000 cycle signal at each modulator is determined by R.C. circuits **35** and **36**. The modulator outputs, when combined in block **17**, produce a control signal having 2000 cycle carrier frequency modulated in accordance with the rate of phase variation of the signals at the antannae. Amplitude modulation is removed in limiter **39** and the magnitude and sense of the phase rate is detected by discriminator **40**. The discriminator output is compared with the yaw gyro output (discussed below) and delivers a control signal to torque motor **15** in response to differences between the phase rate signal and the gyro signal.

An integrating gyroscope **46** (FIG. 7) whose precession is a measure of angular deviations of the missile airframe from the prescribed reference heading, is moun'ed in a cylindrical case (not shown) which is carried horizontally, with its axis on the X-axis of the missile, and is electrically driven to rotate about the Y-axis of the missile. As angular velocity is applied to the missile a precession torque is developed about the X-axis, thereby rotating generator **48** and causing it to produce a voltage that is applicable to amplifier **16** by way of conductor **49**. This input via conductor **49** to amplifier **16**, when added at the amplifier to the command signal entering amplifier **16** from the seeker output circuit **51**, results in the generation of quantity

$$\omega-\dot{\beta}$$

the necessary servo command required for maintenance of the angular velocity dictated by the $\dot{\beta}$ value set up by the phase comparison seeker circuits **21, 22**. The gyro unit includes a second generator **50** rotatable on the X-axis by feedback energy from circuit **52**, which circuit **52** includes an amplifier **53** triggered by a pick-off unit **54** coupled to ram **11**, to introduce a rate correcting follow-up to the signal generating action of generator **48**. A similar gyro combination will perform a corresponding compensating function with respect to the command signal voltage supplied to the pitch servo unit by the seeker circuit associated with antennae A_V and B_V, which seeker circuit will duplicate the one shown diagrammatically in FIG. 7 as associated with antennae A_H and B_H. A third gyro combination of similar design will introduce the third (roll axis) compensating function by supplying operating signal voltage, derived from the angular velocity of roll, to the roll servo unit controlling the angular deflection of the ailerons **10*b*** shown in FIG. 6.

FIG. 5B shows the geometrical relationships between the three missile axes, X_M, Y_M and Z_M and the lead angle L, which is the resultant of the two lead angle components L_h and L_V, corresponding to the β_H and β_V angles shown in FIGS. 5 and 5A, respectively.

The general configuration of the missile, as shown in FIGS. 6 and 8, is necessarily determined to a large extent by the requirements of the guidance system as hereinabove analyzed. Thus, the cruciform canard arrangement close to the forward nose, serving as a rigid mount for the four homing antennae, is the most practical structure for carrying the yaw and pitch control surfaces **10** and **10*a***, respectively, and therefore dictates the positioning of the yaw and pitch gyro combination and servo units. Similarly, roll stabilization is best assured by pivoting the wing flaps, or ailerons, **10*b*** on the coplanar surfaces adjacent the rear end portion of the missile. This again dictates the position of the roll servo and gyro units, and of the tail antennae A_T and B_T, mounted on the associated cruciform surfaces, and serving as the range gating input elements

for feeding to the amplifiers **26** and **27** (FIG. 7) the time control signal energy intercepted directly from the radar transmitter at the launching vehicle or station; that is, the transmitter sending out the radar pulses or a continuous wave, as the case might be, which are reflected back by the target to the phase comparison forward antennae A_V, B_V, A_H, and B_H. In this manner, this conventional range gate is able to determine the target range. As will be noted in FIG. 7 the inputs **71** and **72** from the tail antennas A_T and B_T to the range gate **70** are only partially illustrated but it is understood that any conventional connection can be used. The other major components then logically fall into the relative locations indicated in FIG. 8.

In lieu of supplying course maintaining signals to the control servos, the latter could be supplied with course changing signals from the same seeker system, as on any occasion when collision avoidance, rather than collision, should be desired. Also, in lieu of applying the invention to the control of a pilotless missile, to steer it on a collision course, it could be applied to the control of a piloted aircraft, to steer it automatically toward a landing point.

Obviously many modifications and variations of the present invention are possible in the light of the above teachings. It is therefore to be understood that within the scope of the appended claims the invention may be practiced otherwise than as specifically described.

What is claimed is:

1. In a homing device, receiver means including a fixed antenna array disposed in a plane in orthogonal relation to the longitudinal axis of said homing device and operative to retrieve a portion of a radiant energy train transmitted to a distanct object and reflected back to said antenna array by said object, and means for steering said homing device operatively connected to said antenna array, said steering means including a servo mechanism incorporating a power applying element whose movement is proportional to the time rate of change in the angle of incidence established by said reflected energy train as it impinges upon said antenna array.

2. In a homing device, receiver means including a pair of antenna elements in spaced relationship to the axis of the device, and to each other, said antenna elements being operative to retrieve a portion of the energy transmitted to a distance object and reflected back by said object, and means for steering said homing device operatively connected to said antenna array, said steering means incorporating a power applying element whose movement is proportional to the phase rate difference between the energy received by one of said antenna elements, and that received by the other.

3. Means for guiding a missile to a maneuvering target, comprising two pairs of identical antennas mounted symmetrically about the longitudinal axis of the missile, for reception of phase differentiated signals reflected from the target, and electronic means for converting the phase difference between said signals into a measurement of the time rate of change of the angle between the heading of the missile and the line joining the target and the missile.

4. A system for directing a guided missile to intercept an illuminated target, said system comprising a plurality of fixed antenna elements spaced with respect to one another on the missile to receive signal return energy from said target, pairs of said elements receiving signals differing in phase in accordance with angular relationship of the instantaneous missile axis path and the instantaneous target position, means responsive to said phase differing signals to provide a frequency discriminated signal representing the time rate of change of lead angle between the missile heading and the line of sight to the target and which determines the time interval during which the missile should be maintained in a circular path for first stage maneuvering, rate control means responsive to said frequency discriminating and timing means to cause the missile to effect a second stage linear path of interception with the target path of the second stage, the second stage being commenced when the angular velocity of the missile becomes equal to the time rate of change of the missile angle.

5. A system for directing a guided missile to intercept a radar illuminated target, said system comprising a first pair of antennae including a first and a second antenna and a second pair of antennae including a third and fourth atenna, all four antennae being mounted symmetrically on the sides of the missile, said antennae receiving return energy signals from the illuminated target and having a received signal phase difference between said first and second antennae in accordance with azimuth bearing of the target with respect to the missile, and having a received signal phase difference between said third and fourth antennae in accordance with elevation bearing of the target with respect to the missile, a yaw and a pitch servo unit, means to reduce said respective phase differing signals to an intermediate frequency, first amplifier means to amplify said phase differing signals, means to compare the phase of the first and second and of the third and fourth antennae received signals to produce a phase relation dependent on the instantaneous phase relation of the signals from each of the pairs of antennae, converting means to convert the output of said comparison means to a frequency modulated control signal in accordance with the rate of phase variation of the signals at the antennae, means to remove amplitude modulation of said converted signals, frequency discriminator means to detect the magnitude and sense of the phase rate, a gyroscope unit to introduce an axis deviation function, said gyroscope unit comprising an integrating gyroscope whose precession is a measure of angular deviations of a missile airframe from the prescribed reference heading, a rotating signal generator, a second amplifier, angular velocity applied to the missile causing development of a precession torque about the gyroscope axis, thereby rotating the signal generator to cause the generator to produce a voltage, said voltage being amplified in said second amplifier, a third amplifier, signals from the frequency discriminator being added to the output of the second amplifier to generate a servo command signal required for maintenance of angular velocity dictated by the phase difference of incoming signals, and means responsive to said servo command signals to accordingly deflect control surfaces of the missile.

6. Means to establish a collision course of a missile with a target, said means comprising spaced antennae means responsive to radiant energy signals phase differentiated in accordance with target to missile axis angular relation and received from the target, means to apply lateral acceleration to the missile in response to variations in the rate of change in the characteristics of intelligence reflected from the detected target, phase difference between the signals varying as the angle between the target and the missile axis changes, the frequency difference due to the changing angle being a measure of the time rate of change of the angle, to determine the time interval during which the missile should be maintained in the circular path constituting a first stage of maneuvering, a detection circuit, the two out-of-phase signals received at pairs of the antennae being fed into said detection circuit, means to superimpose the output of said detection circuits upon a carrier wave, frequency modulation means, means to generate signals representing angular deviations of the missile airframe from prescribed reference heading and means responsive to frequency modulated signals from said frequency modulation means to combine the frequency modulated signals with the signals representing angular deviations of the missile airframe from prescribed reference heading, adding of the two signals causing the necessary servo command required for maintenance of angular velocity to cause a target interception linear path.

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