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TM 9-5000-3

DEPARTMENT OF THE ARMY TECHNICAL MANUAL

NIKE I SYSTEMS

NIKE I COMPUTER (U)

DEPARTMENT OF THE ARMY • APRIL 1956

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TECHNICAL MANUAL

NIKE I SYSTEMS

NIKE I COMPUTER (U)

TM 9–5000–3 }
CHANGES No. 1 }

DEPARTMENT OF THE ARMY
WASHINGTON 25, D. C., *17 December 1956*

TM 9–5000–3, 10 April 1956, is changed as follows:

Page	*Paragraph*	*Line*	*Changes*
2	3b	8, 9, and 10	Delete last two sentences.
2	4a	3	Delete "at the time of burst".
4	4c	last line	Delete "at the time of burst".
7	7b	4	Change "is" to **in**.
16	15	4	Change "axes" to **axis**.
79	90	--------------	Change all "microseconds" in the paragraph to **milliseconds**.
82	95	13	Add "loop" after "control".
92	STANDBY	4	Change "44.57" to **44.67**.
97	Computer **reset**	1a	Change "44.57" to **44.67**.

[413.44 (27 Nov 56)]

By Order of *Wilber M. Brucker*, Secretary of the Army:

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General, United States Army,
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[AG 413.44 (9 Mar 56)]

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CHAPTER 1

THE SURFACE-TO-AIR MISSILE PROBLEM

Section I. INTRODUCTION

1. PURPOSE AND SCOPE

a. Purpose. This text is a guide for the officers who are assigned to a Nike I missile battalion and its component units, or to higher headquarters which have Nike I units assigned to them.

b. Scope. Covered in this text is the operation of the Nike I computer, on a detailed block diagram level.

2. REFERENCES AND DEFINITIONS

a. References. **References to TM 9-5000-8 may be abbreviated;** for example, See figure II-85. References to other special texts will show the complete text number and, where appropriate, the paragraph number; for example, TM 9-5000-2 , par 3.

b. Definitions. See appendix I for a glossary of symbols and abbreviations.

3. GENERAL

a. History. At the close of World War II it became apparent that development of faster and more maneuverable hostile bomber aircraft and infinitely more destructive bombs, such as the atomic bomb, made it necessary to increase the capabilities of antiaircraft artillery weapons to attack and destroy such aircraft. Two disadvantages of heavy artillery (90- and 120-mm guns) became apparent: limited range and insufficient accuracy. The operating ceiling of present-day aircraft has increased to such an extent that airplanes may fly above the maximum effective range of heavy artillery. With the probability that hostile aircraft will carry atomic armament, it becomes even more urgent that the accuracy of heavy AAA be increased to insure the destruction of hostile aircraft before the bombs are released. A 90- or 120-mm projectile has a long and uncontrolled time of flight. Although the fire control equipment is capable of accurately predicting the point of impact, the ability of hostile aircraft to maneuver while the projectile is in its uncontrolled flight may result in such a great miss distance that the battery will be unable to accomplish its mission. Weapons have been developed which are capable of attacking and destroying modern enemy aircraft. One of these weapons is the surface-to-air guided missile (SAM). The surface-to-air missile has an effectiveness considerably greater than heavy artillery. It has a greater range because of its motor, and is more accurate because it is guided during flight. Surface-to-air missiles are guided throughout most of their flight paths from ground to target, and are capable of outmaneuvering any presently known hostile aircraft operating at medium or high altitudes. The Nike I missile is a missile of this type. During flight the Nike is guided from the ground by a fire control system.

b. Targets. The fire unit of the guided missile system, XSAM-A-7, is the battery. The primary purpose of the battery is the destruction of long range, bomber type aircraft. A typical target may be assumed to be a bomber having an effective radar area of 26 square meters, which is comparable in size to the present B47. This target is expected to have a ground speed of 650 knots, a ceiling of 60,000 feet, and a maximum maneuverability of 3g at 40,000 feet. These performance characteristics are in excess of those of presently known types of bomber aircraft. The mathematical computations necessary for accurate guidance of the missile are performed by the computer. Therefore, the reader must understand the problem which the computer solves.

Section II. THE SURFACE-TO-AIR MISSILE PROBLEM

4. GENERAL

a. Definition. The surface-to-air missile problem is to guide a missile launched from the ground so that it will arrive and burst in the sky close enough to a target at the time of burst to destroy the target. The steps in the solution of this problem depend upon the weapon to be used and upon the type of guidance system chosen to guide the missile.

b. The command guidance problem. In the Nike I system the direction of flight of the missile is controlled by the computer. The computer-controlled flight starts about 7 seconds after the missile is fired and ends with the burst of the missile after a burst order has been issued by the computer. The two tracking radars keep the computer continuously informed about the present positions of target and missile, and the computer issues to the missile directional commands designed to make the missile intercept the target in the shortest possible time. This method of guiding a missile by ordering it to change its direction of flight is known as command guidance. The surface-to-air command guidance problem can be posed by the following question, which the computer must answer: Knowing the present position of target and missile, and knowing also where target and missile have been, what directional order should be issued to the missile at the moment in order to bring about interception in the shortest possible time? In the formulation of the command guidance problem, past position data must be taken into account. Knowledge of present positions alone would not be sufficient for steering the missile. It is not sufficient to know where the target is at the moment; the computer must also know in which direction the target is going and how fast it is going there. Thus, the courses of the target and the missile can be predicted and their interception under these circumstances can be determined. Should their predicted courses not intercept, this future prediction tells how to rectify the missile course. To tell the direction and the speed at which an object is moving, it is necessary to observe it for a short time and remember where it has been. Hence, the computer needs a memory. The memory of the computer lies in its differentiating circuits. The computer has no control over the speed of the missile. It can tell the missile to turn right or left, up or down, but it cannot tell the missile to go faster or slower. Therefore, the fixed quantities are missile speed and target direction and speed. With these fixed quantities and with missile and target present positions, the computer must find the correct missile course (direction of flight), and must determine the fin orders which are appropriate for bringing the missile onto this correct course. This course will be correct only as long as the direction of flight and speed of the target do not change. If the target performs some maneuver, the correct course for the missile changes continuously and the computer must issue continuous steering orders to keep the missile as close as possible to the latest answer for the correct course.

c. The intercept course. The first question which arises in the study of the command guidance problem is the following: When the direction of flight and the speed of the missile as well as the present position of target and missile is known, in which direction should the missile be moving so that it will approach the target as rapidly as possible? This question falls into the general study of pursuit curves. Looking at the problem from a purely intuitive point of view, it might seem at first that the correct course would be that in which the missile flies so that it points directly at the target. A pursuit based on this simple rule generates a curve known as a dog curve, because that is roughly the path a dog follows when pursuing a rabbit. Such a pursuit curve places minimum requirements on the intelligence of the steering apparatus (ground control), and is used by a class of missiles that operate on the homing principle. A missile which flies a dog curve intercepts a moving target later than a missile flying a more advantageous course. As a result, a target might escape the range of the guidance equipment, although it could have been intercepted if a pursuit curve with a faster rate of approach were used. It is quite plausible that the pursuit time can be reduced by heading off the target, that is by flying the missile toward a point which is ahead of the target. This is done with an AA projectile when it is fired at the future position of the target in space. A pursuit curve which takes full advantage of time that can be gained by leading the target is called an intercept curve. Mathematical analysis of the pursuit problem shows that the requirement for maximum rate of approach is met by the intercept curve. The Nike I computer directs the missile so that it approaches the target along a slightly modified intercept course. Therefore, the problem which the Nike I computer solves is the determination of the correct intercept curve which the missile must follow so that the missile will arrive and burst in the sky close enough to a target at the time of burst to destroy the target.

5. ORIENTATION OF THE MISSILE

a. General. An elementary knowledge of the missile and its external guidance system is desirable as background information before discussing a solution to the SAM problem.

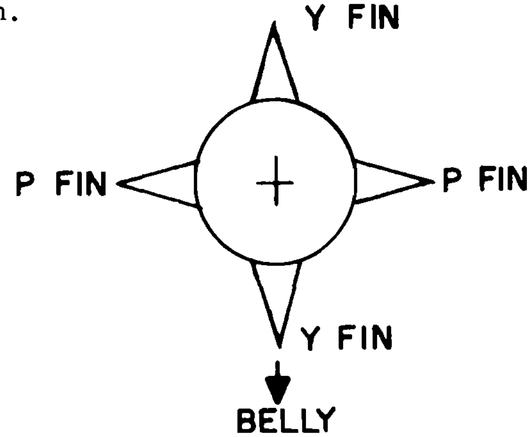


Figure 1. Rear view of the orientation of the first Nike missile.

b. The missile. The original Nike I missile was caused to turn (yaw) by applying orders to the Y-fin pair, and to dive or climb (pitch) by applying orders to the P-fin pair while the missile was traveling through space. The maximum permissible dive of the missile flying as shown in figure 1 did not permit the missile to intercept a target within the dead zone shown in figure 2. The dead zone can be reduced by orienting the position of the missile in flight as shown in figure 3. By rotating the Y-fin pair 45° counterclockwise the dead zone is reduced because the same dive order now applied to both fin pairs will result in a maximum resultant dive 1.414 times greater. For example, an acceleration order of -5g applied to the missile P-fins as shown in figure 1 will permit a maximum dive of 5g; whereas, 5g applied to both fin pairs as shown in figure 3 will result in a maximum dive of $(1.414)(5g) \approx 7g$ as shown in figure 4. Thus, the dead zone is considerably reduced. A 7g dive is the maximum dive the Nike missile can make. Fin orders to the missile are in terms of acceleration where 5g is an order which would cause the missile to accelerate 5 times faster than the acceleration due to the force of gravity (g = 10.7 yards per second per second). The missile is stabilized so that it does not roll on its longitudinal axis as it travels through space. An external view of the missile showing control fins appears in figure 5.

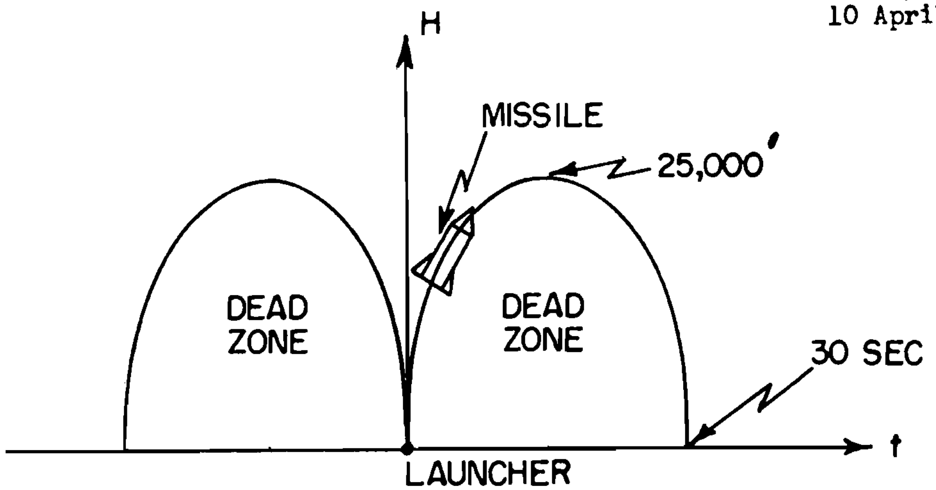


Figure 2. Intercept is impossible within the dead zone.

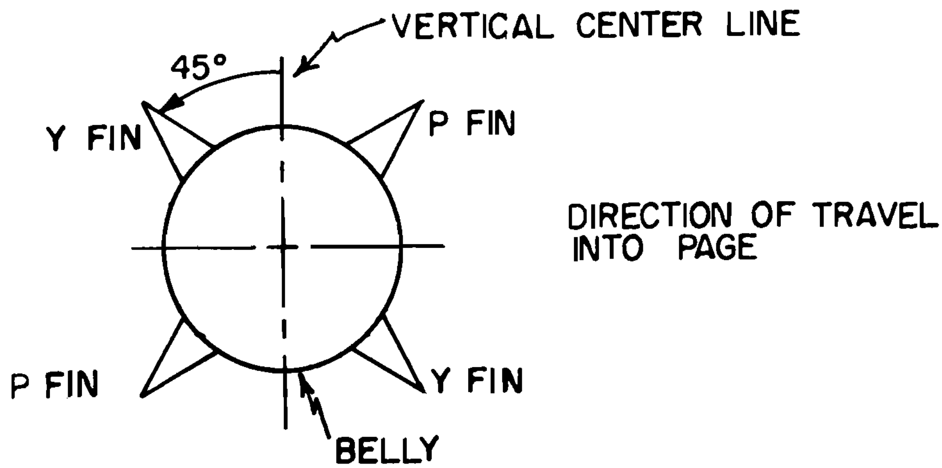


Figure 3. Orientation of the present Nike I guided missile.

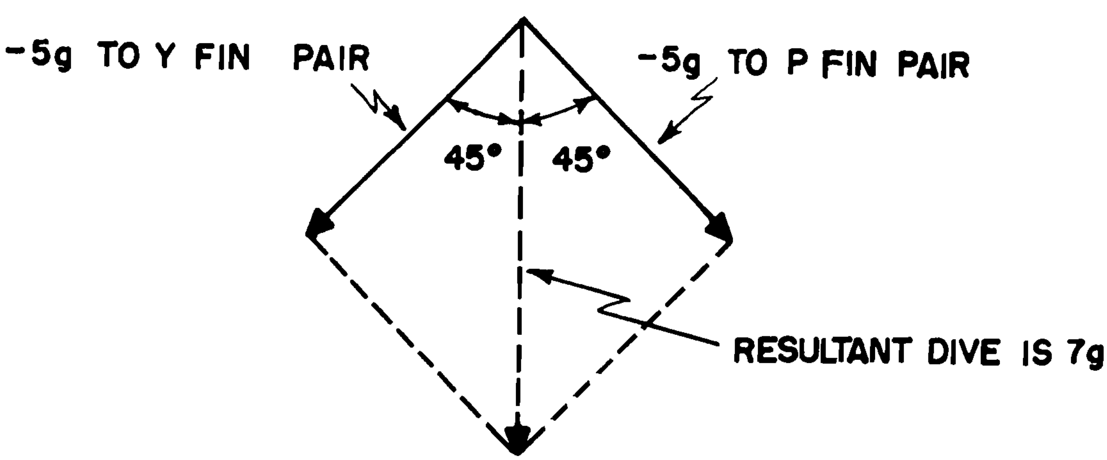


Figure 4. Resultant dive order with -5g applied to each fin pair.

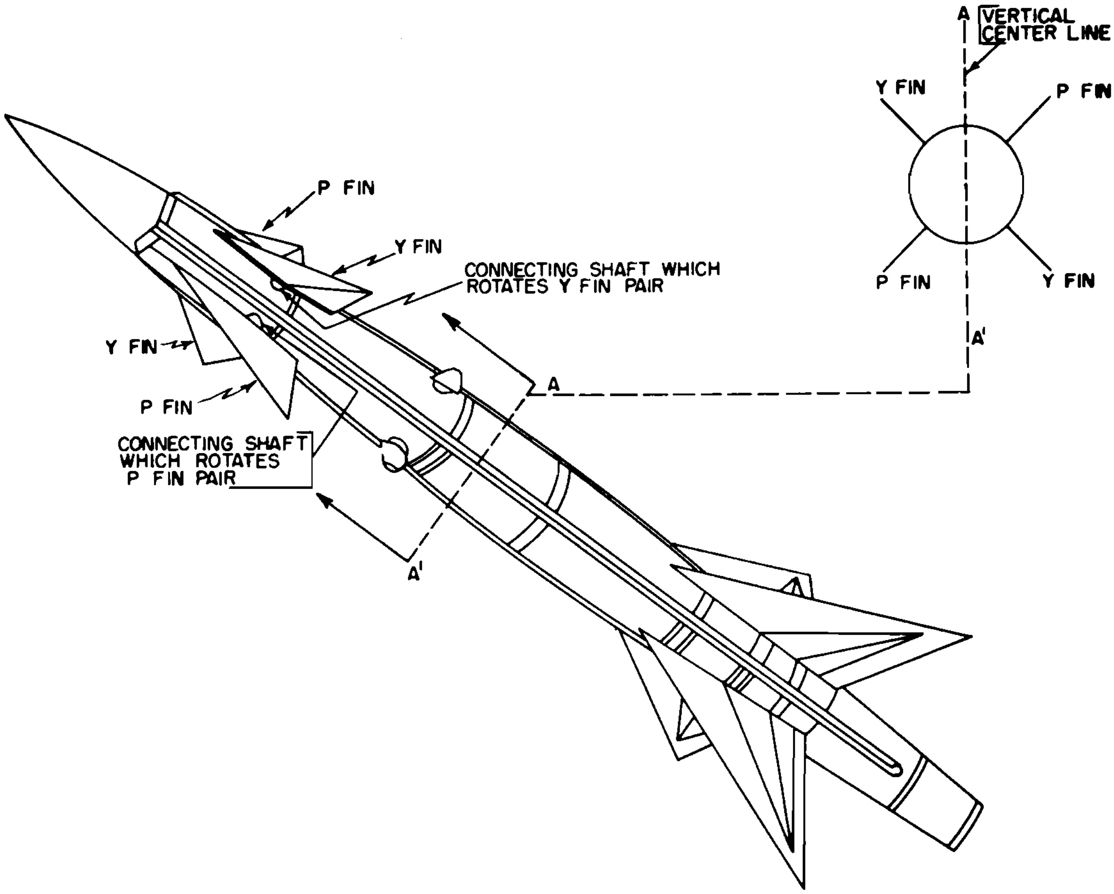


Figure 5. View of the missile showing control fins.

6. EXTERNAL GUIDANCE SYSTEM

The Nike I system is controlled during flight by a command guidance system. The command guidance system permits continuous correction of the missile flight path after launching. The guidance commands are computed and transmitted by the external ground guidance equipment of the system. The Nike I computer performs the necessary mathematical calculations to determine how much lateral and vertical acceleration must be caused by each fin pair to produce a desired result and then transmits suitable orders to the missile through the missile-tracking radar. The computer also sends at the proper time an order which causes the missile to burst. If the missile is to be guided successfully, the orders sent to the missile must cause accelerations in relation to a fixed reference common to both the computer and the missile. Once the missile is launched, this fixed reference must not change regardless of the attitude of the missile. The device used to maintain this fixed reference is the roll amount gyroscope. The use of the roll amount gyroscope for this purpose imposes a physical limitation upon the amount that the missile may turn to the right or left. If the angle through which the missile turns (yaws) exceeds a certain design limit, the gyroscope is forced into a condition known as gimbal lock. When the gyroscope is in gimbal lock the reference no longer remains fixed, but changes whenever the missile attitude changes. Consequently, if gimbal lock occurs the missile becomes uncontrollable. Because of this physical limitation the computer determines the azimuth of the intercept point before the missile is launched. The computer then transmits to the missile the intercept point azimuth which causes the plane of the roll amount gyro to be oriented in the azimuth of the intercept point. With the plane of the roll amount gyro so oriented, the missile, approximately 5 seconds after launch, is guided on the azimuth to the intercept point. If the target does not change its course and speed after missile launch, the angle through which the missile will turn on its way to the intercept point is quite small. Little time is lost in making such a small turn, and the time elapsed between launch and intercept is considerably reduced from the time which would elapse if the missile were required to turn through large angles. One advantage, then, of prelaunch determination of intercept point azimuth is the increased rate of battery fire because of the reduced launch-to-intercept time. Even if the target executes a violent evasive maneuver, the resulting angles through which the missile would have to turn will almost never be large enough to exceed the critical angle for gimbal lock.

7. MILITARY SPHERICAL COORDINATES

a. General. Points on the ground and in space are located by describing the position of such points with reference to directions or points which are known.

b. Definition. Spherical coordinates are defined as a system for locating a point in space by the length of a radius vector from the fixed origin, the angle this vector makes with a reference plane through the origin, and the angle the projection of the radius vector on the reference plane makes with a fixed line is the reference plane.

c. Military spherical coordinates. The location of a target and missile in space is determined by using the spherical coordinates commonly used by the Army. The three spherical coordinates are slant range (D), elevation angle (E), and azimuth (A). Figure 6 shows a target located by spherical coordinates. Target slant range, elevation, and azimuth are called D_T, E_T, and A_T, respectively. The spherical coordinates of the missile are D_M, E_M, and A_M.

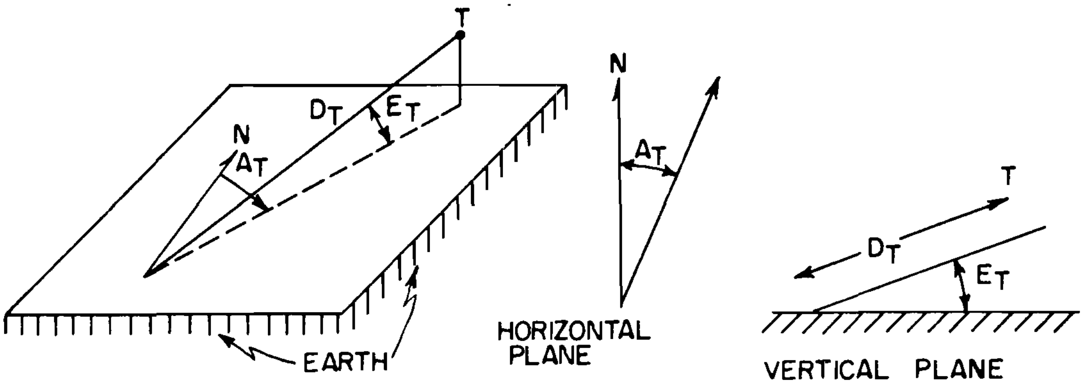


Figure 6. Location of a target using spherical coordinates.

8. RECTANGULAR COORDINATES

The mathematical computations which must be performed by the computer are more easily accomplished by using rectangular coordinates. For this reason, target and missile locations are converted from spherical to rectangular earth coordinates. Rectangular earth coordinates are commonly used by the Army. A military map is gridded with horizontal and vertical lines. Soldiers are taught during basic training to follow the rule, read right and up. The numbers on each grid line increase in magnitude when reading from left to right (west to east) and when reading from bottom to top (south to north). The most common unit of measurement in the rectangular coordinate system is the yard. The reference point in this system is known as the origin of coordinates. In 2-dimensional rectangular coordinate systems, two lines called axes pass through the origin and intersect at right angles to each other. Figure 7 shows a 2-dimensional rectangular coordinate system. The horizontal and vertical axes are designated as the X and Y axes. A 2-dimensional rectangular coordinate system is contained in a plane surface. A 3-dimensional coordinate system is formed by passing an additional axis through the origin perpendicular to the X and Y axes, and hence perpendicular to the plane formed by the X and Y axes. A 3-dimensional coordinate system is shown in figure 8. The axis perpendicular to the X and Y axes is designated as the H axis. The location of a target, T, in a 2-dimensional coordinate system is shown in figure 9. Point T is located by measuring its distance from the origin along the X and Y axes. The X coordinate is 4 units and Y coordinate is 2 units. The location of a point, T, in a 3-dimensional coordinate system is shown in figure 10. Point T is located by measuring its distance from the origin along each of the three axes. The X coordinate is 4 units, the Y coordinate is 4 units, and the H coordinate is 2 units.

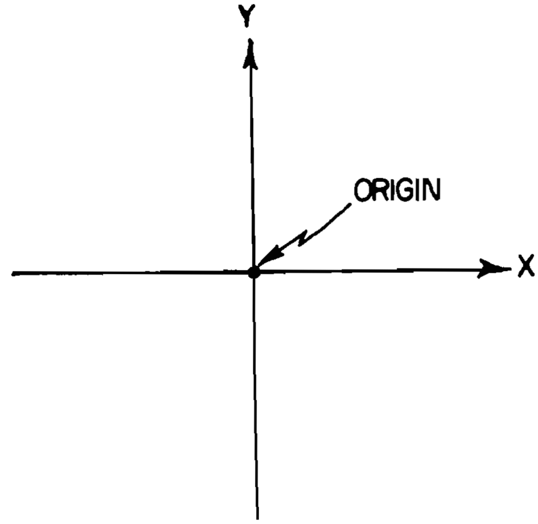


Figure 7. Two-dimensional rectangular coordinate system.

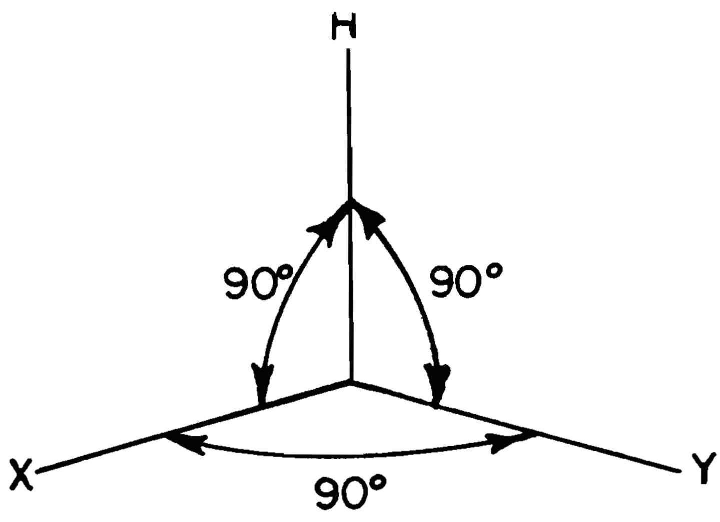


Figure 8. Three-dimensional rectangular coordinate system.

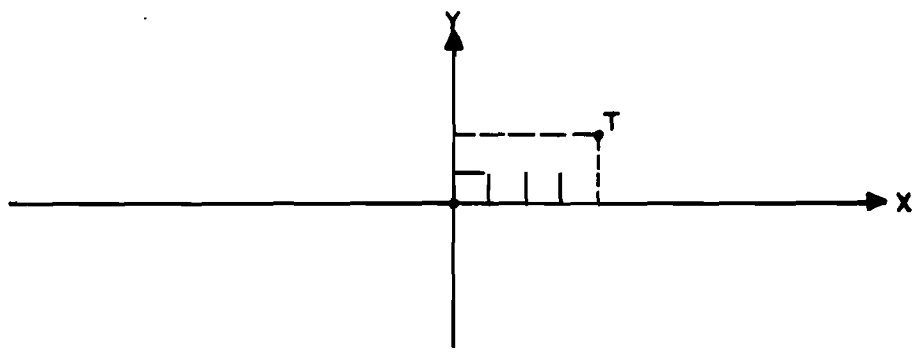


Figure 9. Location of a point in a 2-dimensional coordinate system.

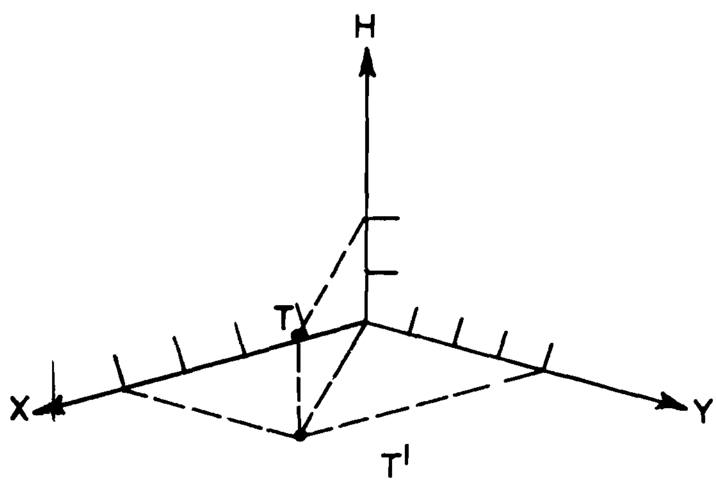


Figure 10. Location of a point in a 3-dimensional coordinate system.

9. SPEED, VELOCITY, AND ACCELERATION

a. Speed. Speed is defined as the rate at which distance changes with respect to time. Consider a sedan moving along a road. If the speedometer dial points to the number 45 the driver knows that the speed of his sedan is 45 miles per hour. In computations performed by the Nike I computer, speed is expressed in yards per second. (Conversion factor: 1 mph = 0.5 yd/sec, approximately.)

b. Velocity. The term velocity expresses both speed and direction. Velocity can be represented by a vector. If a driver in a sedan equipped with a compass is moving at a speed of 45 miles per hour as indicated by his speedometer, and at the same time his compass needle points northeast, then his sedan has a velocity of 45 miles per hour northeast. In computations performed by the computer, velocity is expressed in terms of yards per second and direction is expressed in mils measured from an established line of direction. Thus, the velocity of the sedan above is expressed as 22 yards per second, at an azimuth of 800 mils from north. A change of velocity with respect to time may imply a change in speed, a change in direction, or changes in both speed and direction.

c. Rectangular components of velocity. Many of the computations performed by the computer are made in terms of velocity expressed in rectangular coordinates. The representation of a velocity in rectangular coordinates requires that the vector which represents velocity be resolved (converted) into components which lie along the axes of the coordinate system being used. The process of resolving a velocity into its 2-dimensional components is shown in figure 11. The velocity produces X and Y components ($\dot{X}$ and $\dot{Y}$) of the vector V.

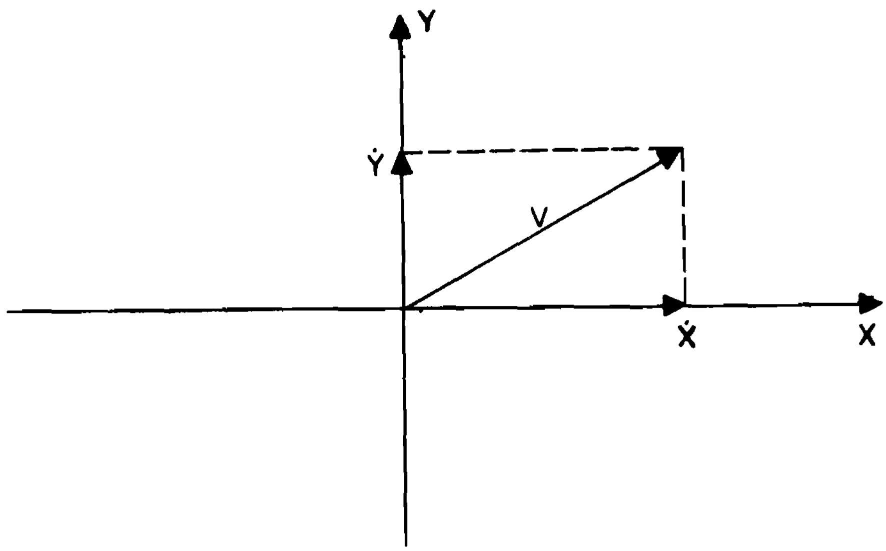
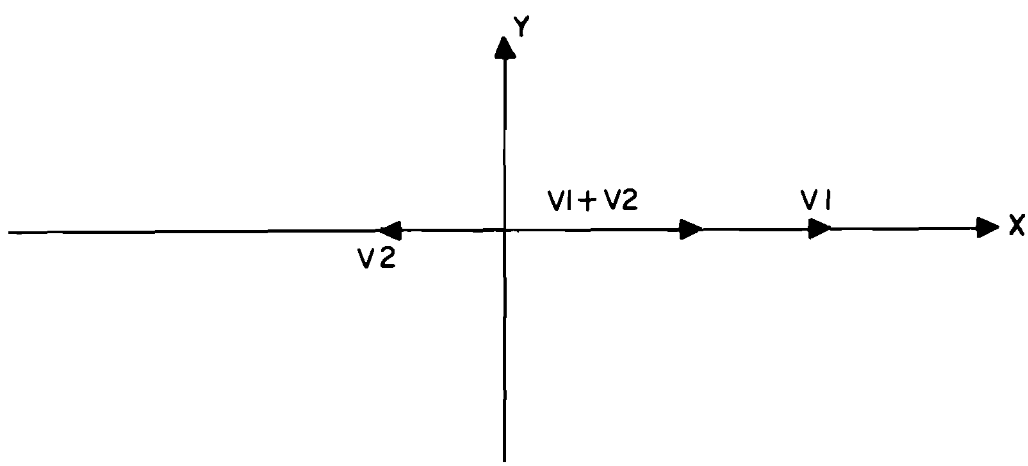


Figure 11. Resolving a vector into its rectangular components.

d. Acceleration. Acceleration is defined as the time rate-of-change of velocity as regards either speed, direction, or both. Consider the man in the sedan going 45 mph with his compass pointing northeast. If 1 hour later the speedometer indicates 50, 2 hours later it indicates 55, and 3 hours later it indicates 60, the average acceleration is 5 miles per hour per hour. As another example, consider again the sedan traveling at 45 miles per hour with the compass indicating northeast (800 mils). If 1 minute later the compass reads east (1, 600 mils), and 2 minutes later it reads southeast (2, 400 mils), the average change in direction is 800 mils per minute. In this example, the direction associated with velocity has changed with respect to time and by definition the sedan is accelerating. This is true even though the 45-mph speed of the sedan is unchanged. The process of steering the missile during its flight consists of causing the missile to change direction while in flight. Therefore, the missile is guided by causing it to accelerate and for this reason the voltages produced by the computer as final outputs are expressed in terms of acceleration. The unit used is g. It is a symbol which represents the acceleration due to the force of gravity and is equal to 10.7 yards per second per second. For example, a missile that makes a 7g turn has an acceleration 7 times greater than the acceleration due to gravity.

(1) Addition of vectors.



(2) Vector addition along coordinate axes.

Figure 12. Vector addition.

e. Vector addition and subtraction. A review of the principles of vector addition and subtraction is necessary because the majority of the quantities developed in the computer are vector quantities. One important fact to be remembered is that on paper the arrow which represents a vector may be moved to any position as long as it is kept parallel to its original position. Two or more vectors may be added by graphically attaching the tail of one vector to the head of the other. Figure 12 illustrates this process. V_1 extends from the origin 0. The tail of V_2 is attached to the head of V_1. The resultant $(V_1 + V_2)$ extends from 0 and meets the head of V_2. Vector addition is shown on rectangular coordinate axes in figure 12. Vector subtraction is performed by reversing the direction of the vector to be subtracted and then performing vector addition. Figure 13 shows this process.

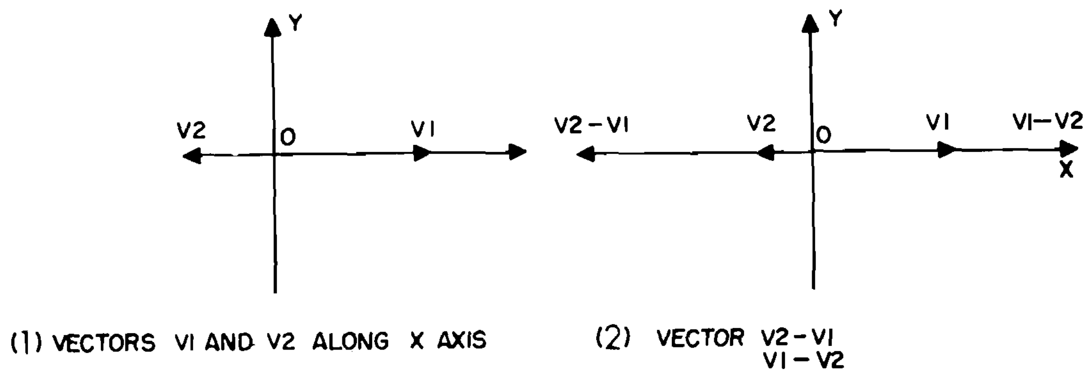


Figure 13. Vector subtraction.

10. RESOLUTION OF VECTORS

a. General. The computer in a number of instances knows the components of velocity along one set of axes and needs to know the components of velocity along another set of axes. This operation of the computer, to find the components of velocity along the new axes, is called resolution of vectors. The method of converting components of velocity from one rectangular coordinate system to another which is rotated away from the original by some angle consists of drawing the axes of the new coordinate system and using the laws of trigonometry to find the magnitude of the components in the new system.

b. Conversion of missile velocity from rectangular earth to gyro coordinates. As an example of the graphical conversion of components from one system to another, consider the conversion of missile-velocity components from rectangular earth-to-gyro coordinates. Figure 14 illustrates this method. The components of $\dot{X}_M$ and $\dot{Y}_M$ along the Y_G axis are as follows:

The component of $\dot{X}_M$ along the Y_G axis is $\dot{X}_M \sin A_G$.

The component of $\dot{Y}_M$ along the Y_G axis is $\dot{Y}_M \cos A_G$.

To find $\dot{Y}_{GM}$, these components of velocity are added vectorially, so that the vector equation for $\dot{Y}_{GM}$ is:

$$\dot{Y}_{GM} = \dot{X}_M \sin A_G + \dot{Y}_M \cos A_G. \tag{1}$$

This equation also represents the algebraic computation performed by the computer.

The component of $\dot{X}_M$ along the X_G axis is $\dot{X}_M \cos A_G$.

The component of $\dot{Y}_M$ along the X_G axis is $\dot{Y}_M \sin A_G$.

To find $\dot{X}_{GM}$, these components of velocity are added vectorially, so that the vector equation for $\dot{X}_{GM}$ is:

$$\vec{\dot{X}}_{GM} = \vec{\dot{X}}_M \cos A_G + \vec{\dot{Y}}_M \sin A_G, \tag{2}$$

and the algebraic equation is:

$$\dot{X}_{GM} = \dot{X}_M \cos A_G - \dot{Y}_M \sin A_G. \tag{3}$$

The missile velocity is in the direction of the resultant vector as shown in figure 15. The components of velocity along the earth axes are $\dot{Y}_M$ and $\dot{X}_M$. With respect to the gyro axes the resultant velocity also has components, $\dot{Y}_{GM}$ along Y_G and $\dot{X}_{GM}$ along X_G.

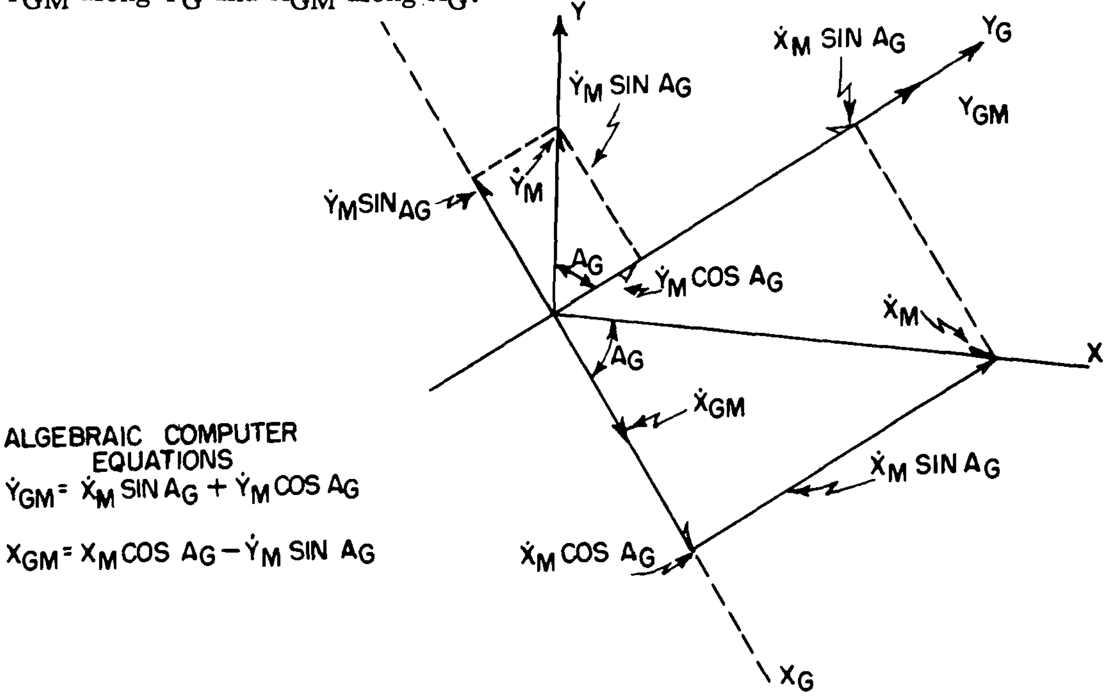


Figure 14. Conversion of missile velocity from rectangular earth-to-gyro coordinates.

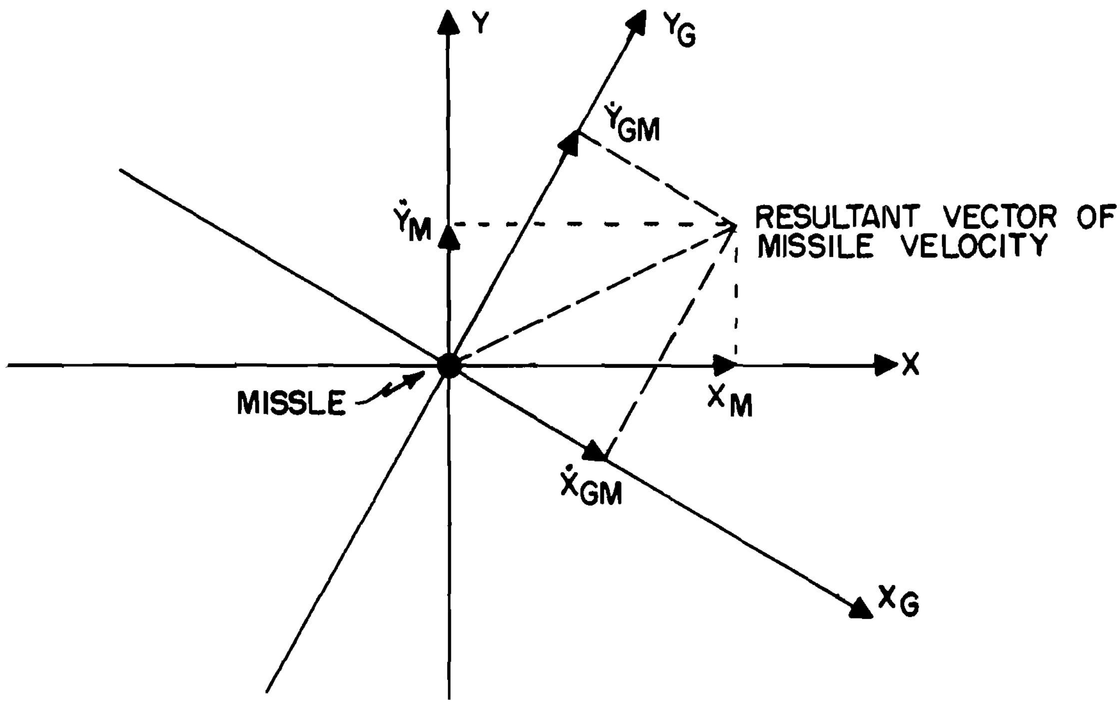


Figure 15. Components of missile velocity.

Section III. THE NIKE I COMPUTER REFERENCE PLANES AND AXES

11. GENERAL

In its solution of the surface-to-air missile problem the Nike I computer requires a system of reference. (See figure II-85) This reference system consists of four planes, three coordinate systems, and three angles. The planes are: the earth plane, the horizontal plane, the gyro reference plane, and the missile velocity slant plane. The coordinate systems are: the earth coordinate system, the gyro coordinate system, and the missile coordinate system. The angles are: the gyro azimuth angle, the climb angle, and the turn angle. A definition of each of the planes, coordinate systems, and angles is given below.

12. THE EARTH PLANE

The earth plane is the basic reference plane of the Nike I system. It is the horizontal orientation plane of the target-tracking radar and contains the earth X (east-west) and Y (north-south) axes. The earth plane is fixed when the system is oriented.

13. THE HORIZONTAL PLANE

The horizontal plane is always parallel to the earth plane and contains the gyro reference axis X_G and Y_G. The plane changes in altitude as the missile climbs or dives, but during the missile flight phase never changes its orientation.

14. THE GYRO REFERENCE PLANE

The gyro reference plane is a vertical plane, with the missile gyroscope as its center, and is the plane of rotation of the gyro rotor. The plane contains the Y_G axis, the H_G axis, the predicted intercept point, and the gyro rotor. The gyro reference plane is fixed along the gyro azimuth and changes position as the gyro azimuth changes during the prelaunch phase.

15. THE MISSILE VELOCITY SLANT PLANE

The missile velocity slant plane is a slant plane containing the X_G axis and the missile velocity axis. The missile velocity plane is not a fixed plane due to the fact that the missile velocity vector changes during the missile's flight. The plane pivots around the X_G axes.

16. EARTH AXES

The Y axis is a north-south line, while the X axis is an east-west line, lying in the horizontal earth plane and passing through the origin (target-tracking radar). The H axis is perpendicular to both the X and Y axes at their point of intersection (TTR). The following rule is used with the rectangular earth coordinate system. Distances measured eastward from the origin (along the X axis) are positive; those measured westward are negative. Distances measured northward from the origin (along the Y axis) are positive; those measured southward are negative. The H distance is positive when above the target-tracking radar and negative when below it. The X component of velocity is positive when the arrow points to the east, negative when the arrow points to the west. The Y component of velocity is positive when the arrow points northward, negative when the arrow points southward. The H component is positive when it points up, negative when it points down. The rule and the application of the rule are valid regardless of the location of the missile or target in space with respect to the target-tracking radar.

17. GYRO AXES

The gyro reference and gyro spin axes are perpendicular to each other. The gyro reference axis is designated Y_G and always points toward the projection of the intercept point on the horizontal plane before the FIRE switch is thrown. The gyro spin axis is designated as X_G. The angle between the Y axis and Y_G axis is A_G, the gyro azimuth angle.

18. MISSILE AXES

The missile axes intersect at the missile's center of gravity. The missile axes are:

a. The climb axis, which is vertical through the center of gravity.

b. The turn axis, which is horizontal and passes through the center of gravity.

c. The missile velocity axis, which is perpendicular to the other two and lies longitudinal through the missile.

19. GYRO AZIMUTH ANGLE

The gyro azimuth angle is the angle measured clockwise from north to the Y_G line. Its apex is at the designated launcher. This angle is always positive.

20. MISSILE FLIGHT ANGLES

a. General. The missile flight angles are the climb angle and the turn angle.

b. Climb angle. The climb angle is the angle between the horizontal plane and the missile velocity slant plane. This angle is always positive and is unlimited.

c. Turn angle. The turn angle is the angle between the intersection of the gyro reference plane and the missile velocity line, measured in the missile velocity slant plane. It is positive in a clockwise direction, as observed from the rear of missile, and negative in a counterclockwise direction. This angle is limited by computer circuitry to ±70°.

CHAPTER 2

THE NIKE I COMPUTER

Section I. GENERAL

21. BACKGROUND

The solution of the antiaircraft artillery problem has always been extremely complex. The movement of the target as well as the ballistics of the gun and of the projectile must be taken into consideration. With the increased performance of aircraft, the problem of obtaining a kill with conventional artillery has become almost impossible. The advantage of the Nike I system over conventional antiaircraft artillery is that the missile may be guided from the ground during flight. To perform the calculations involved in guidance, an electromechanical computer was devised. To solve the problem, it is sufficient for the computer to know the present position of the target and of the missile. However, the computer must be able to calculate the direction of travel and the speed of the target and of the missile. Thus, if the course of the target and missile can be predicted, the interception of the target by the missile under these circumstances can be determined. Should the present missile course not intercept the target, this prediction will tell how to rectify the future course of the missile. The computer must issue continuous steering orders to keep the missile as close as possible to the latest correct course. The Nike I computer is subdivided functionally into three sections: the prelaunch section, the initial turn section, and the steering section. Each section functions according to an ordered sequence of events automatically controlled by relays, switches, and timers dispersed throughout the computer.

22. PRELAUNCH SECTION

Assuming that a target has been detected and identified as hostile, the information defining its position is transferred from the battery control trailer to the radar control trailer. When the radar operators indicate that they are tracking the designated target, the computer enters the prelaunch phase. Four seconds are necessary for the computer to settle and to begin to supply smooth, reliable data after the target-tracked signal has been received. During this phase, the prelaunch section receives target present position data as well as data regarding the position of the missile to be fired. The computer predicts the intercept point position and the time of flight of the missile by combining data regarding the present position and the rate of change of target position with data known to represent the ballistic characteristics of the missile. The prelaunch section then solves for the time of flight and the azimuth angle of the ground projection of the intercept point. The time of flight of the missile is required for two reasons:

a. A time-of-flight plot enables the battery control officer to observe the progress of the engagement and to determine when to initially open fire.

b. Without the knowledge of the time of flight, the prediction of the intercept point would be impossible.

The azimuth angle of the intercept point is the clockwise angle between north and a horizontal projection of the line connecting the launcher and the intercept point. The value of this angle is required at the launcher to orient the missile roll gyro so that the gyro reference plane, which contains the initial missile trajectory, will also contain the predicted intercept point. This is necessary because the gyro imposes a limit in turn of 70° right or left of center on the missile. Therefore, the gyro azimuth is determined as accurately as possible so that the missile will never need to approach this 70° limit. The fire order initiates the flight phase and applies a brake to the gyro preset servo shaft.

23. INITIAL TURN SECTION

At the end of the boost phase, the missile will be heading upward at an uncertain angle, and it will require a rapid maneuver to bring the missile onto an approximate intercept course. Rapidity in executing this maneuver is of prime importance, since it is possible that the missile is initially moving in the wrong direction. If it is moving away from the correct course, its position with respect to the target becomes increasingly less favorable every second that passes before the missile has been swung around. The regular missile control system operating on the steering vector principle is inadequate for steering the missile during this sharp maneuver. For this reason, the initial turn section is provided for controlling the missile during the initial turn phase of computer controlled flight. The initial turn section will also steer the missile in a path that will not exceed the maximum tracking rate of the missile-tracking radar. The initial turn section will control the missile until the on-trajectory and radar-cleared signals have been received.

24. STEERING SECTION

After the initial turn section ceases to function, the steering section starts to direct the missile toward the intercept point. It determines steering errors by taking the components of the relative closing velocities, obtained by differentiating the rectangular coordinate values of the missile and target positions, and equating them to the corresponding velocities obtained by dividing the closing distance between the missile and the target by the predicted time to intercept. The steering errors are then transferred into the gyro and missile axes and resolved into acceleration orders to the missile fins. These orders are transmitted to the missile via the missile-tracking radar beam which causes the missile to fly a course in order to reduce the steering errors to zero. When the time to intercept reaches 250 milliseconds, the burst circuit is enabled and the time correction voltage is disconnected from the time-to-intercept servo, which thereafter is allowed to run down at a second/second rate. At a preselected time before the time to intercept reaches zero, the burst order is sent to the missile. Burst is ordered before zero time for two reasons: 1) there is a slight delay encountered between the time of issuance of the burst order and the detonation of the warhead; and 2) it is advantageous to burst the warhead somewhat short in certain situations. For example, if the target and the missile were approaching each other at supersonic speed, it would be advantageous to allow the burst pattern to expand and let the aircraft fly into the pattern. At present, it is believed that for an approaching target, the burst order should be sent 105.5 milliseconds before time zero. If the exploding missile destroys the target and the target tracked signal is removed, the computer returns to the standby condition and is then ready for the next problem. Otherwise, the computer is returned to the prelaunch configuration. The missile-tracking radar slews to the next designated missile or the test responder if no missile is designated.

25. LOCATION OF COMPUTER ELEMENTS

a. General. Refer to figure II-22 ; a drawing of the interior of the battery control trailer is shown. The computer is located at the immediate left of the entrance port. The first components seen are the two amplifier cabinets. Next to the amplifier cabinet assembly is the computer servo cabinet. The third component is the computer power cabinet. The three sections of the computer are on the road side of the trailer. On the curb side of the trailer are the horizontal and vertical plotting boards. These components are associated with the computer and operate on inputs obtained from the computer.

b. Computer amplifier cabinet. Refer to figure II-31 . On the left side of the diagram are the two sections of the computer amplifier assembly. When the locking handles provided are used, the doors open readily. There are two swinging frames upon which are mounted the 76 DC amplifiers, zero-setting mechanisms, testing devices, and other components. The rest of the components are mounted on the rear wall of the amplifier cabinet. Most of the components mounted on the rear wall are arranged in hinged sections which may be opened for removal or testing of the units. Located here are computer modulators, servoamplifiers, relay panels, terminal strips, and other components. The functional designations of the computer components are indicated on the individual chassis. Directly below each DC amplifier is its input and feedback network. The eight zero-setting controls are located in the center of the swinging frames. The DC amplifiers associated with one AZS control are mounted adjacent to it. At the bottom of the amplifier cabinets are terminal strips 21 through 270 on the left side, and 271 through 500 on the right side. In general, one may observe that the right portion of the amplifier cabinet contains components employed in steering error channels, target coordinates, and solution of the intercept point. The left portion contains components relating to missile coordinates, initial turn circuits, and various miscellaneous circuits.

c. Computer servo cabinet. Refer to figure II-34 At the top is the computer control panel. This panel contains many of the controls which affect computer operation. Of special importance are the COMPUTER CONDITION switch and the parallax and static test switches. See figure II-35 of ly. In the compartment below are the five computer servos. The servos may be readily removed for maintenance and adjustment. The five servos in the Nike I computer are, from left to right, top to bottom:

(1) The climb angle servo.

(2) The turn angle servo.

(3) The time-to-intercept servo.

(4) The gyro azimuth servo.

(5) The ballistic elevation servo. The dead-time unit is also in the left corner.

The ballistic resistor panel and the test voltage dividers are mounted behind the computer control panel.

d. Computer power cabinet. Refer to figure II-36 The computer power cabinet and its internal components are shown in this figure. The power control panel (fig II-37), power supplies, plugs, jacks and terminal strips concerned with the distribution of a-c and d-c power to the various units in the computer and plotting boards are contained within this cabinet. The components may be removed for maintenance and adjustment.

Section II. THE PRELAUNCH SECTION

26. GENERAL

The purpose of this section is to present to the reader a detailed block diagram discussion of the units which make up the prelaunch section of the Nike I computer. Each unit discussion will include, where applicable, a mathematical analysis of circuit operation, and a discussion of the function of the unit The detailed block diagram of the prelaunch section is shown in TM 9-5000-8, page 113.

27. TARGET COORDINATE CONVERTER

The purpose of the target coordinate converter is to convert the target present position data from spherical to rectangular coordinates. It is composed of 3 spiral potentiometers and 10 DC amplifiers. The range data potentiometer is located in the radar range and receiver cabinet in the radar control trailer. The azimuth and elevation data potentiometers are located in the azimuth and elevation data converter units on the target-tracking radar antenna trailer. The ten DC amplifiers are located in the computer amplifier cabinet. For easy identification, the position of each DC amplifier on the equipment frame is labeled according to the function of the amplifier. For example, the $-D_T$ amplifier is clearly stamped $-D_T$. There are three inputs to the coordinate converter: D_T, E_T, and A_T. These are shaft positions determined by mechanical positioning of the target antenna and range unit. Essentially there are three outputs (X_T, Y_T, and H_T); however, both positive and negative values of each of these outputs are needed. Each output may also have more than one use in the computer. The scale factor used in the coordinate converter is determined by the voltage applied to the range data potentiometer and the maximum range it is designed to represent. The voltage applied to the range data potentiometer is the regulated scale factor voltage, +106 2/3 volts. The maximum design range is 106,666 yards. The maximum range data which can be supplied by the range data potentiometer is limited to 100,000 yards by the range limitations of the TTR. The scale factor is 1 millivolt (0.001 volt) per yard.

28. MATHEMATICAL ANALYSIS OF THE TARGET COORDINATE CONVERTER

Figure 16 illustrates the method by which spherical coordinates are converted to rectangular coordinates. It can be readily proved that a mathematical relationship exists between spherical and rectangular coordinates, and that the following equations are true:

$H_T = D_T \sin E_T$ (the altitude of the target above the TTR). (4)

$R_T = D_T \cos E_T$ (the ground range of the target from the TTR). (5)

$X_T = R_T \sin A_T$ (east-west component of target position in earth coordinates). (6)

$Y_T = R_T \cos A_T$ (north-south component of target in earth coordinates). (7)

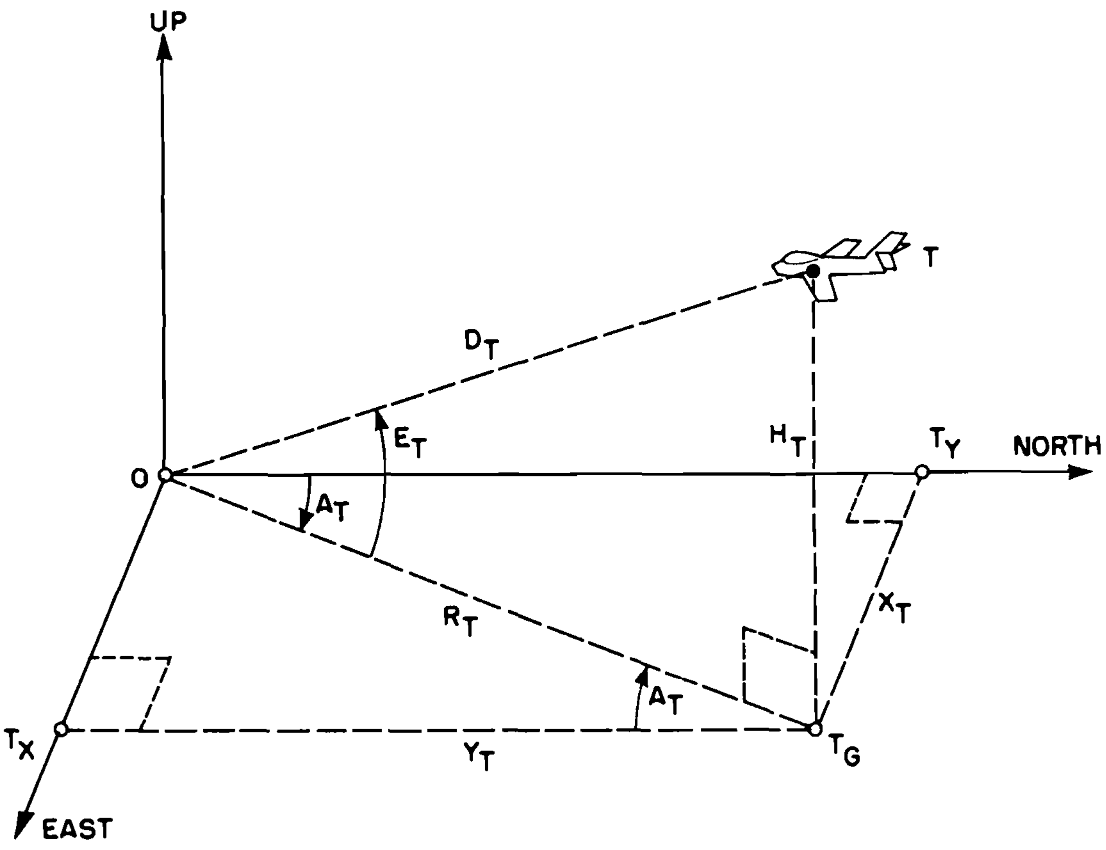


Figure 16. Resolution of target present position data into rectangular coordinates.

29. SIMPLIFIED FUNCTIONAL OPERATION OF THE TARGET COORDINATE CONVERTER

a. Range data potentiometer. The target range data potentiometer contains a linearly wound card to which +106 2/3 volts is applied. The range servo of the TTR positions a single brush arm which picks off a voltage proportional to the slant range of the target. D_T (slant range) is always positive regardless of the target azimuth, and the voltage representing D_T, taken from the range data potentiometer, is always positive. The elevation data potentiometer requires that D_T be supplied to it as both a positive and negative voltage. Two DC amplifiers are used to perform this function and to supply the positive and negative voltages representing D_T to the elevation data potentiometer; they also provide necessary isolation.

b. Elevation data potentiometer. This potentiometer is a sine-cosine spiral potentiometer. It receives two input voltages representing $+D_T$ and $-D_T$. The two brush arms (one sine, one cosine) are mechanically positioned to the elevation angle as the TTR tracks the target. The brush arms pick off voltages representing $-D_T \sin E_T$ which equals $-H_T$, and $-D_T \cos E_T$ which equals $-R_T$.

(1) Voltage signs. An explanation is required to clarify the meaning of positive or negative signs preceding the symbol for an element of coordinate data. For example, a symbol such as $-H_T$ does not necessarily mean that the voltage analog is negative, but it means that the representative voltage is negative when the coordinate element it represents is a positive value.

(2) Outputs of H_T. These outputs are taken at three separate places. Output $-H_T$ is taken directly from the elevation data potentiometer and sent to the closing speed solver of the computer steering section. Output $+H_T$ is taken from the $+H_T$ amplifier and sent to the intercept point solver. A third output, $-H_T$, is taken from the $-H_T$ amplifier and sent to the target prelaunch differentiator and to the target steering differentiator.

(3) Voltages representing $+R_T$ and $-R_T$. These voltages are developed from the $-D_T \cos E_T$ output of the elevation data potentiometer, using two DC amplifiers as in the development of $+H_T$ and $-H_T$. Voltages representing $+R_T$ and $-R_T$ are applied to the target azimuth data potentiometer.

c. Azimuth data potentiometer. This potentiometer is also a sine-cosine spiral potentiometer. The two brush arms of the azimuth data potentiometer are mechanically positioned to the azimuth angle as the TTR tracks the target. Voltages picked off by the brush arms represent $-R_T \sin A_T$ and $-R_T \cos A_T$. As explained above, the positive or negative sign preceding a symbol for coordinate data establishes the polarity of the voltage representing that element of data. $-R_T \sin A_T = -X_T$, and $-R_T \cos A_T = -Y_T$. Values of X_T and Y_T are each taken from three separate places. The quantities $-X_T$ and $-Y_T$, taken directly from the azimuth data potentiometer, are sent to the closing speed solver The quantities $+X_T$ and $+Y_T$ are sent to the intercept point solver. The quantities $-X_T$ and $-Y_T$, taken from the $-X_T$ and $-Y_T$ amplifiers, are sent to the target prelaunch and target steering differentiators.

30. MISSILE COORDINATE CONVERTER

a. Purpose. The purpose of the missile coordinate converter is to convert the missile present position data from spherical to rectangular coordinates. It is almost an exact duplicate of the target coordinate converter, even to interchangeability of components.

b. Components. The three data potentiometers are located with the MTR components; the ten DC amplifiers are located in the computer amplifier cabinet. The range data potentiometer is energized by the scale factor voltage and supplies range data as does the target range data potentiometer; however, the maximum value of D_M is limited by the MTR range servo system.

c. Input data. The inputs are D_M, E_M, and A_M and the outputs are X_M, Y_M, and H_M.

d. Output data. The scale factor for the output data is 1 millivolt per yard. This data is sent to the following places:

(1) $+H_M$ to the $\frac{H}{t}$ network is the closing speed solver.

(2) $-H_M$ to the missile away circuit and to the missile differentiator circuit.

(3) $+X_M$ and $+Y_M$ from the azimuth data potentiometer to the closing speed solver.

(4) $-X_M$ and $-Y_M$ from the $-X_M$ and $-Y_M$ amplifiers to the initial turn section, and to the missile differentiators.

(5) $+X_M$ and $+Y_M$ from the $+X_M$ and $+Y_M$ amplifiers to the initial turn section.

31. LAUNCHER PARALLAX UNIT

The purpose of the launcher parallax unit is to supply, to the intercept point solver, coordinate data from handset potentiometers representing the location of the launcher area with respect to the TTR. It is desirable that the predicted intercept point be computed with respect to the exact location from which the missile is to be launched. Until the designated missile is tracked by the MTR, the exact missile position data are not available. Therefore, the launcher parallax unit supplies data which represents the coordinates of the geometric center of the launcher area. After a missile is tracked by the MTR, the data from the launcher parallax unit are replaced by missile position data through relay contacts. Since the height of each launcher position in the launcher area will not vary greatly in most cases, it is not necessary to refine the height data to the exact missile height. The three potentiometers of the launcher parallax unit are located on the computer control panel. The maximum data setting in each coordinate is ±6,000 yards. The scale factor is 1 millivolt per yard.

32. MATHEMATICAL ANALYSIS OF THE LAUNCHER PARALLAX UNIT

To determine the intercept point and the time of flight of the missile, the distance between the target and the missile must be known. Since the launcher holding the missile may be up to 3 miles from the TTR, the launcher-to-TTR distance must be considered in the computation of the intercept point. The intercept point is then determined with respect to the launcher position. Figure 17 illustrates the problem of the effect of launcher parallax. With the TTR position as the origin of coordinates, a condition is shown where both the target and the missile location are in the first quadrant. The positions for both target and missile are resolved into rectangular coordinates. For simplicity of discussion, consider the X coordinate only. X_T is the target coordinate. X_L is the launcher coordinate. The distance desired is the X distance between the missile and target positions. This is calculated by subtracting the position of the launcher from the position of the target: $X = X_T - X_L$. Y and H are similarly determined by using $Y = Y_T - Y_L$ and $H = H_T - H_L$. Thus, to define the distance between the target and the missile the distance between the TTR and the missile must be known.

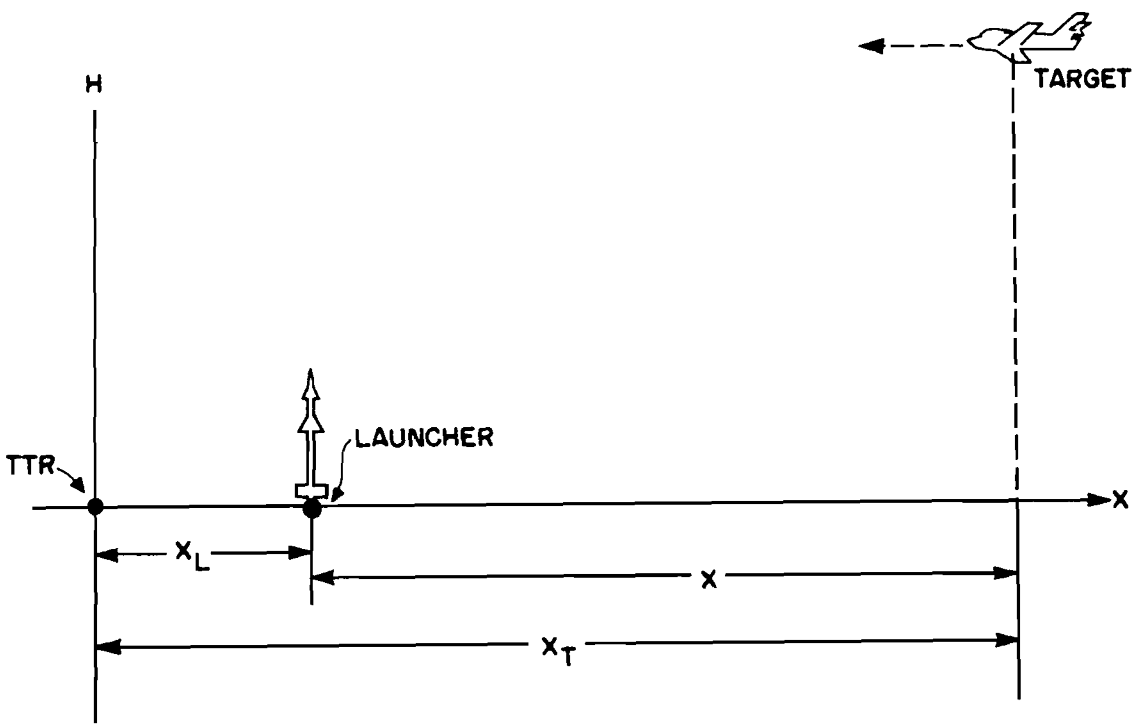


Figure 17. Calculation of launcher-to-target distance.

33. TARGET PRELAUNCH DIFFERENTIATORS

a. Purpose. Velocity multiplied by time is equal to distance. Computation of the intercept point is based upon the distance a moving target will have traveled during a given time of flight. The target coordinate converter supplies the target present position data. It is the purpose of the target prelaunch differentiator to produce data proportional to the rate of change of the target present position in order to compute the intercept point. Symbols for these data are $\dot{X}_P$, $\dot{Y}_P$, and $\dot{H}_P$.

b. Differentiator circuit. The target prelaunch differentiator consists of three DC amplifiers, with capacitor inputs for differentiation, and resistor-capacitor feedback networks for data smoothing. The amplifiers are labeled according to their output data. The inputs are $-X_T$, $-Y_T$, and $-H_T$. The outputs are $\dot{X}_P$, $\dot{Y}_P$, and $\dot{H}_P$. The output scale factor is 12.5 millivolts per yard per second.

c. Operation. The target prelaunch differentiator is enabled by relay operation when the TARGET TRACKED signal is received. The functional operation is similar to that of the target steering differentiator and the missile differentiator to be studied later. The basic difference is in the data-smoothing characteristics. The target prelaunch differentiator uses 4-second, data-smoothing networks, but the target steering differentiator and missile differentiator uses 2-second, data smoothing networks.

34. SIMPLIFIED FUNCTIONAL OPERATION OF PRELAUNCH DIFFERENTIATORS

a. General. Since the three differentiating circuits of the target prelaunch differentiator are identical, only the $+\dot{H}_P$ differentiator will be considered. The input is $-H_T$.

b. Measurement of change of target position. When the position of the target is changing in altitude, there will be an output voltage whose amplitude is proportional to the rate of change of H_T. A positive present position of the target in the H coordinate is represented by a negative voltage input to the differentiator. If the target is increasing in altitude at a uniform rate, the negative voltage representing the H position will be changing to a greater negative value, and the input voltage on the grid of the DC amplifier through the capacitor input will be negative. The DC amplifier will then have a positive output voltage whose magnitude is proportional to the H component of target velocity. Since the $\dot{H}$ coordinate is increasing, the velocity is positive. Thus, there is a positive voltage representing a positive velocity, and the DC amplifier is labeled $+\dot{H}_P$.

c. Data smoothing. The data-smoothing network is a resistor-capacitor network in the feedback circuit of the DC amplifier and is used to average the spurious rate voltages developed in the differentiator. The spurious rate voltages are a result of the uneven rate at which the TTR tracks a target, and the small voltage steps in the present position data due to the granularity of the data potentiometers. The output data $+\dot{X}_P$, $+\dot{Y}_P$, and $+\dot{H}_P$ are sent to the intercept point solver.

35. THE INTERCEPT POINT SOLVER

a. Purpose. The purpose of the intercept point solver is to solve for the rectangular coordinates (X_I, Y_I, H_I) of the point where the missile should meet the target. After the MISSILE TRACKED signal is received, the origin for the predicted intercept coordinates is the designated launcher. (Before this signal, the center of the launcher area is the origin for the intercept point coordinates.) The A_G servo computes the proper gyro azimuth, A_G. Because this azimuth is measured from the designated launcher to the predicted intercept point, the A_G servo must use intercept point coordinates (X_I, Y_I) with the designated launcher as the origin.

b. Components. The intercept point solver is used only in the prelaunch section of the computer. It is composed of three similar channels: one for X, one for Y, and one for H. Each channel consists of a time-of-flight potentiometer, a dead-time potentiometer, and an input network; the X_I and Y_I channels each contain one positive and one negative DC amplifier, but the H_I channel contains a negative amplifier only.

c. Location of potentiometers. The three time-of-flight potentiometers are located in the time-to-intercept servo assembly in the servo cabinet. The dead-time potentiometers are located in the servo and timer assembly in the servo cabinet. The five DC amplifiers and their respective input networks are located in the computer amplifier cabinet.

d. Inputs. The inputs to the intercept point solver from the target prelaunch differential are $+\dot{X}_P$, $+\dot{Y}_P$, and $+\dot{H}_P$ (representing target velocity components). The scale factor of these inputs is 12.5 mv/yd/sec. Inputs $+X_T$, $+Y_T$, and $+H_T$ (representing target present position) come from the target coordinate converter with a scale factor of 1 mv/yd. Inputs $-X_L$, $-Y_L$, and $-H_L$ come from the launcher parallax unit with a scale factor of 1 mv/yd.

e. Outputs. The outputs $+X_I$, $-X_I$, $+Y_I$, and $-Y_I$ go to the A_G servo. The $-X_I$ and $-Y_I$ outputs also go to the horizontal plotting board. The $-H_I$ output goes to the ballistic circuits and to the vertical plotting board. All outputs of the intercept point solver have a scale factor of 1 mv/yd.

36. MATHEMATICAL ANALYSIS OF THE INTERCEPT POINT SOLVER

a. Computing X_I. The intercept point coordinates are measured from the launcher to the intercept point. For simplicity, figure 18 depicts a special situation for the solution of only the X_I coordinate by the intercept point solver. The formula for X_I is

$$X_I = X_T - X_L + \dot{X}_P (t + t_d). \qquad (8)$$

Launcher parallax, X_L, is the X coordinate of the launcher position (from the target radar) in yards. By subtracting launcher parallax from the coordinate, X_T, of the present target location from the TTR, the launcher-to-target distance $(X_T - X_L)$ is obtained. Target velocity, $\dot{X}_P$, multiplied by the total time from FIRE until intercept $(t + t_d)$, gives the distance the target will travel until it is intercepted by the missile. This distance, $\dot{X}_P (t + t_d)$, is added algebraically to the launcher-to-target distance $(X_T - X_L)$, giving the coordinate X_I of the intercept point from the launcher.

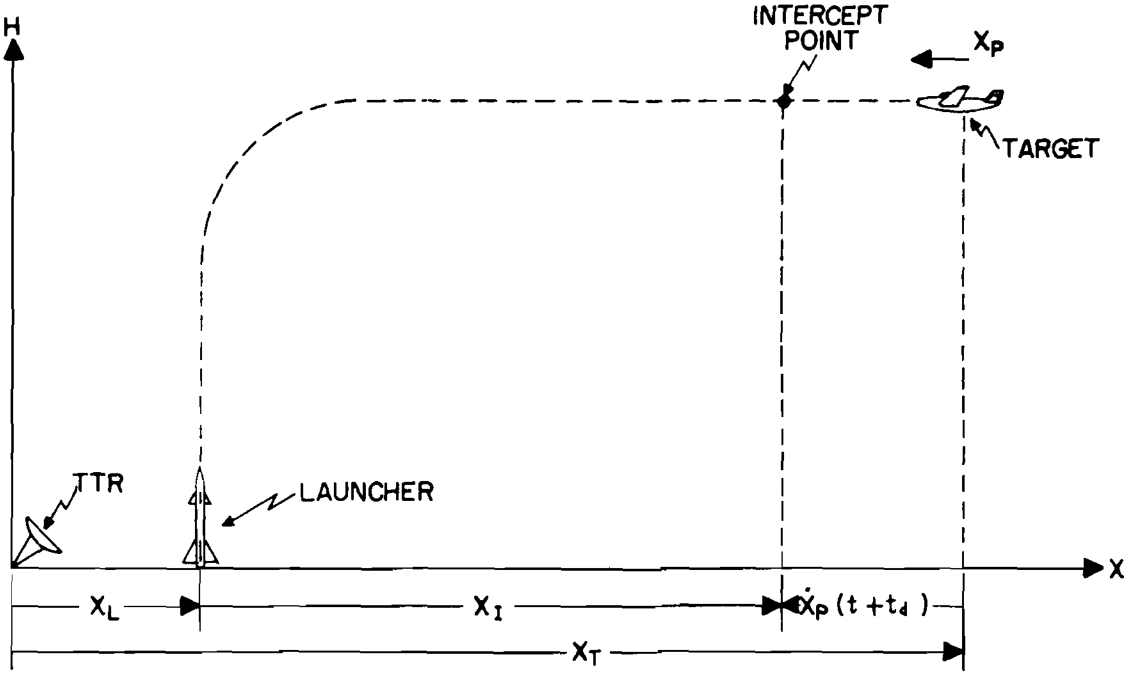


Figure 18. Computation of the intercept point coordinate, X_I.

b. Specific example. Any quantity in equation 18 can be positive or negative. The usual sign conventions apply: east, north, and up are positive; south, west, and down are negative. For example, consider the situation shown in figure 18. The launcher is 4,000 yards east of the TTR. The target is a plane presently located 60,000 yards east of the TTR and is moving westward toward the defended area with an X component of velocity of 600 mph. At this instant the dial on the time-of-flight predictor reads 63 seconds. Following is a tabulation of given data:

$$X_T = +60,000 \text{ yd}$$

$$X_L = +4,000 \text{ yd}$$

$$\dot{X}_P = -600 \text{ mph} = -300 \text{ yd/sec}$$

$$t = 63 \text{ sec}$$

$$t_d = 7 \text{ sec.}$$

Substitute these values in equation (8) and solve for the X component of the intercept point.

$$X_I = +60,000 - (+4,000) + (-300)(63 + 7)$$

$$X_I = +35,000 \text{ yards.}$$

The predicted intercept point is located 35,000 yards east of the designated launcher. Bear in mind this solution is for one instant only ($t = 63$ sec): in the computer the solution for the intercept point (during prelaunch) is continuous.

c. Computing Y_I and H_I. Y_I and H_I are computed in exactly the same manner as X_I and their equations are:

$$Y_I = Y_T - Y_L + \dot{Y}_P (t + t_d) \tag{9}$$

$$H_I = H_T - H_L + \dot{H}_P (t + t_d). \tag{10}$$

d. Dead time. Note that t_d, the dead time, is always 7 seconds during the prelaunch solution. This 7 seconds is the expected time interval between the pressing of the fire button (FIRE) and roll stabilization. This time is allocated in the following manner: 2 seconds after FIRE for the mechanical settling of the caged roll-amount gyro in the missile, 3 seconds for booster burnout and separation, and 2 seconds for the missile to accomplish roll stabilization, i.e., the pointing of the missile belly toward the predicted intercept point.

e. Time of flight. Time of flight, t, is assumed to be properly computed by the time-of-flight predictor. Note that t is measured from FIRE + 7 seconds until intercept. The dial on the front of the time servo drawer will read time of flight, t, during the prelaunch solution.

37. OPERATION OF INTERCEPT POINT SOLVER AFTER FIRE

At FIRE the A_G angle solution is fixed (frozen) and the missile is launched 2 seconds later. The prelaunch computer section cannot cease operation at this time because it is necessary to have a smooth time solution transition from the prelaunch solution to the steering time solution (if target velocity does not change) between FIRE and FIRE + 7 seconds. The dead time unit performs this function. The change in target position during this 7-second interval is compensated for by the decrease of dead time, t_d. For example, consider the preceding illustrative problem. At FIRE the computed intercept point component X_I was +35,000 yards. Now consider the same problem 6 seconds after FIRE (t_d = 1 sec) with no change in target velocity. The new target position coordinate (X_T) would be +58,200 yards. Use equation (8) which is also applicable after FIRE.

$$X_I = X_T - X_L + \dot{X}_P (t + t_d)$$

$$X_I = +58,200 - 4,000 - 300 (63 + 1)$$

$$X_I = +35,000 \text{ yards.}$$

It can be seen that throughout the 7-second interval after FIRE, the predicted intercept point solution does not change if the target velocity is unchanged.

38. DEAD-TIME UNIT

a. Purpose. One purpose of the dead-time unit is to provide for a smooth transition between the time-of-flight predictor time solution and the time solution of the time-to-intercept servo. Another purpose is to reject the missile at FIRE + 5 seconds if it has not left the launcher.

b. Sequence. The solution for time in the Nike I computer is performed by the following units for the intervals indicated:

INTERVAL	UNIT SOLVING FOR TIME
	PRELAUNCH
TARGET TRACKED to FIRE	Time-of-flight predictor (t changing); dead-time unit (t_d constant at 7 seconds).
FIRE to FIRE + 7 seconds	Time-of-flight predictor (t constant); dead-time unit (t_d decreases from 7 seconds to zero).

At FIRE + 7 seconds	Prelaunch time components cannot change the time solution. This instant ordinarily coincides with roll stabilization and MA + 4 seconds.
	STEERING
MA + 4 seconds to ON TRAJECTORY	Time is decreased at 1 second per second by bias circuit in time-to-intercept servo.
ON TRAJECTORY to t = 0.25 second	Time solution performed by time-to-intercept servo.
t = 0.25 second to t = 0 second	Time is decreased at 1 second per second by bias circuit in time-to-intercept servo.

The above data indicates that the dead-time unit controls the time solution from FIRE to FIRE + 7 seconds. The dead-time unit, in effect, freezes the intercept point solution (X_I, Y_I, and H_I) for 7 seconds if the target velocity does not change. If the target velocity does not change, the time-of-flight predictor will not change its solution for t and a smooth transition to control by the time-to-intercept servo will be affected.

c. Dead-time mechanism. The dead-time unit is located in the right side of the servo and timer assembly in the servo cabinet. An electromagnetic clutch controls the application of mechanical power from the dead-time motor to the gearing. This clutch energizes at FIRE. A dial indicating the dead time is visible from the front of the servo cabinet. A friction clutch prevents the dials and gearing from turning after the magnetic clutch has been disengaged. The dead-time potentiometers contained in the assembly are used by the intercept point solver. Microswitches control zero reset and the missile reject action.

39. BALLISTIC CIRCUITS

a. General. To determine the ballistic functions of the missile during the prelaunch phase, a special circuit is used which can represent these characteristics of the missile as a function of time. It is, in a sense, a ballistic table to which the computer can refer.

b. Components. These tables are composed physically of a DC amplifier and its input network, a modulator, a low-power servoamplifier, a servomotor generator and associated gearing, a feedback device for geometric gain control, three potentiometers, and several resistive networks. The B amplifier and input network are located in the amplifier cabinet. The B modulator and low power servoamplifier are located in the rear bay of the amplifier cabinet. A ballistic resistor panel, located at the top of the servo cabinet, contains most of the resistive networks. The servo and timer assembly mounts the mechanical parts of the A_G and B servos, and t_d unit. The portion relating to the ballistic circuits is in the center; that is, the potentiometers, servomotor generator, and gearing.

c. Inputs. The quantity $-H_I$ is supplied as one input voltage from the intercept-point solver with a scale factor of 1 mv/yd. The t servo produces a +t voltage and positions mechanically four potentiometer brush arms. From the +t voltage and the brush arms, voltages are generated which position the ballistic servo system through a normal computer servo channel.

d. Outputs. The ballistic circuits provide two output signals to the t_{p1} input network, $+R_B$ and $+D_B \cos B$, which together are analogous to a ground range to the intercept point.

40. MATHEMATICAL ANALYSIS OF BALLISTIC CIRCUITS

a. Constant time circles. When this system was developed in the laboratory, it was found that the flight of the missile could be related to constant time circles. Some of these are illustrated in figure 19. If a missile were launched from the origin of this coordinate system, it would require 40 seconds to reach any point on the 40-second curve. Thus, it would require the same time, 40 seconds, to reach a target at a 22,000-yard horizontal range and a 57,000-foot altitude as would be required to reach one at a 24,000-yard range and a 6,000-foot altitude. These curves can be approximated by circles called constant time circles, each of a different radius. The center of these circles are identified by labeled points on the locus. Because there is but one constant time circle for any one point of the locus, and because each time circle has a different radius, it is possible to plot the coordinates of the circle centers and radii as functions of time.

b. Ballistic elevation angle B. In figure 20, the circle center is represented by the coordinates R_B and H_B; the radius by D_B. Any point in the A_G plane, for example, can be designated by using H_B, R_B, D_B, and the angle at which the radius D_B must be positioned from the horizontal plane. This angle is the ballistic elevation angle B. Angle B is determined by the ballistic elevation B-servo. Angle B is needed only to obtain a correct value of time of flight.

c. The t-servo. This servo compares the computed ground range to the intercept point from the gyro azimuth servo ($-R_I$) with the empirical ground range from the ballistic circuits ($R_B + D_B \cos B$) in the equation

$$-R_I + R_B + D_B \cos B = 0. \qquad (11)$$

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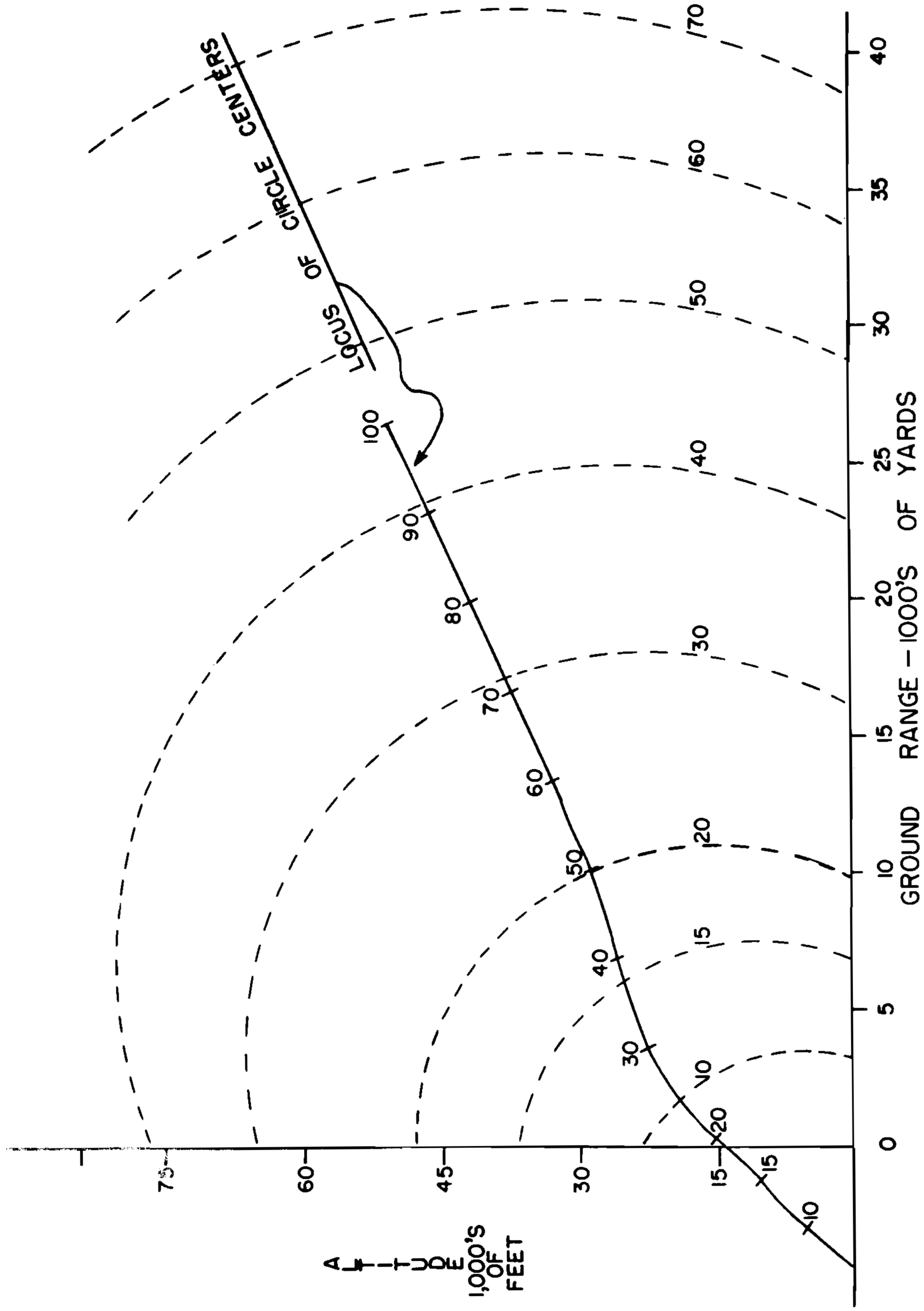

Figure 19. Constant time circles.

This solution presumes a correct value of angle B, which must be determined. $-X_I$ and $-Y_I$ are used in the A_G servo to obtain $-R_I$ for use in the t servo, so B must be found by using other parameters.

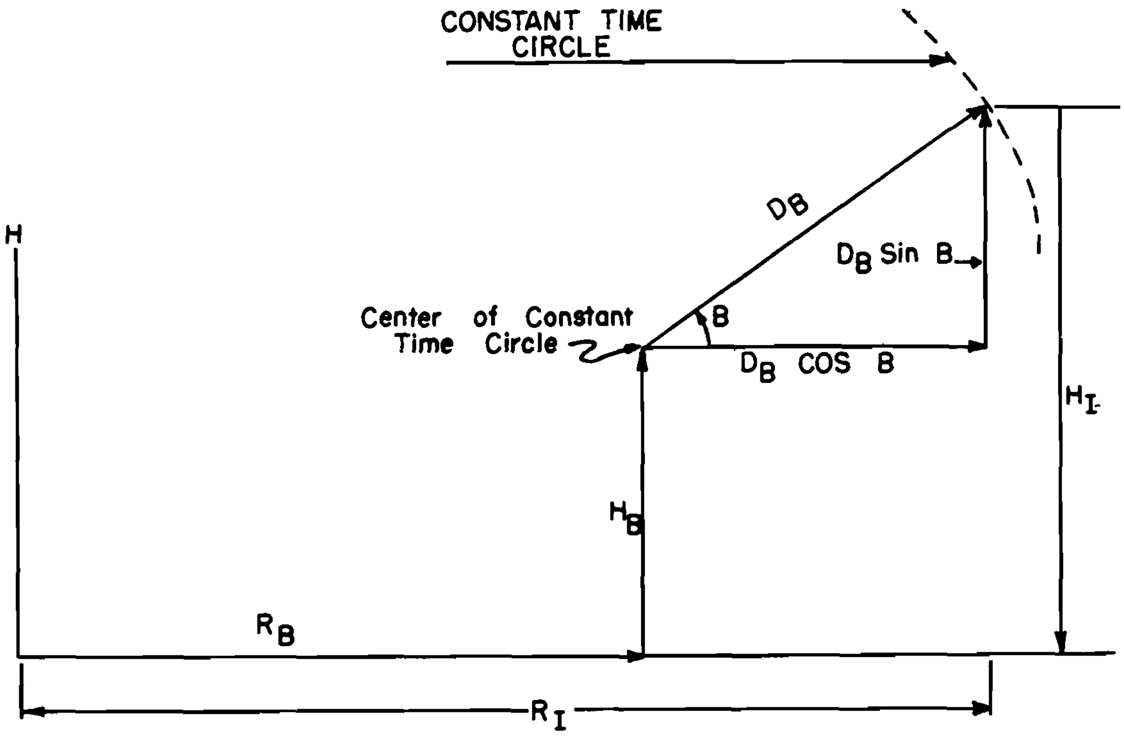

Figure 20. Calculation of ballistic elevation angle.

d. Ballistic height. In figure 20, the ballistic height of the intercept point is composed of the parameter H_B and the projection of the radius D_B on the perpendicular from the intercept point to the horizontal plane. The predicted height of intercept is

$$+H_I = H_T - H_L + \dot{H}_P (t + t_d). \qquad (12)$$

The missile and the target should coincide in the vertical plane at burst, so the B-servo uses the equation

$$-H_I + H_B + D_B \sin B = 0. \qquad (13)$$

to obtain a solution for angle B. H_B and D_B are generated in the ballistic circuits. $-H_I$ is available from the intercept point solver. This leaves sin B the only quantity of equation (13) to be controlled by the B-servo. It should be remembered that sin B may be of either polarity, positive if angle B is above the horizontal, negative if below.

41. SIMPLIFIED FUNCTIONAL OPERATION OF BALLISTIC CIRCUITS

a. B-servo system. The functional block diagram of a simple closed loop control system is drawn in dashed line in figure 21. The controlled variable is compared with the reference inputs by a comparator. The error existing between the input and output is used to energize elements in the controller, which control the output in an effort to eliminate the error. As may be seen in figure 21, the B-servo conforms to a basic closed loop control system. The reference input ($-H_I$) from the intercept point solver is applied to the comparator (B-amplifier). The second reference ($+H_B$ and $+D_B$) is generated in other parts of the ballistic circuits. H_B is fed directly to the comparator, but D_B is modified by the feedback element (a B sine potentiometer card) before being used in the comparator. If correct values are assumed for t and A_G, then D_B is the only input to be altered by feedback. Since $-H_I$ is a negative voltage, $+H_B + D_B \sin B$ must ordinarily be a positive voltage in order that equation (13) be used by this servo. When the two reference inputs are of equal magnitude, no error exists and the equation (13) is satisfied.

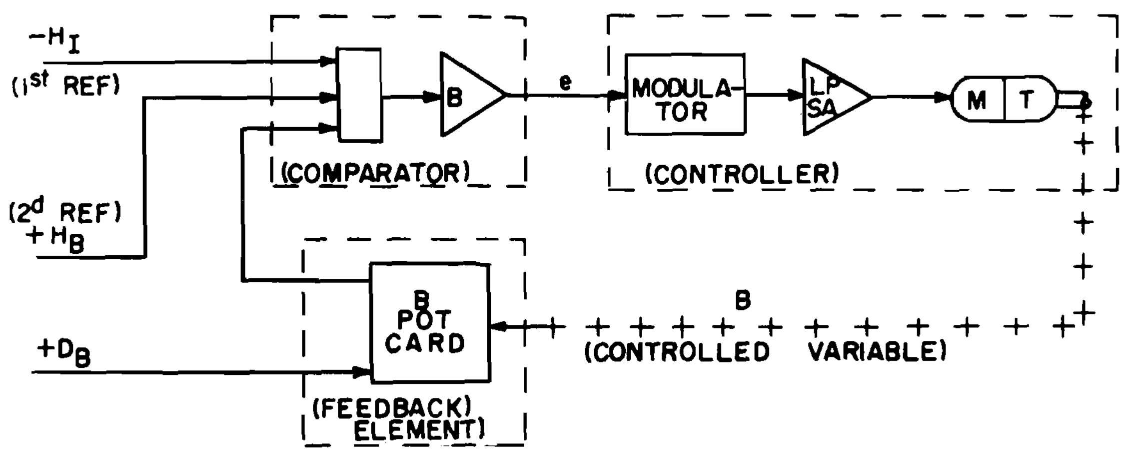

Figure 21. The B-servo on a basic closed loop control system.

b. Controller operation. Before reaching a solution, the B-amplifier applies the error (d-c voltage) to the modulator. Here it modulates a 400-cps, a-c voltage in both phase and amplitude to control the motor. This a-c voltage is raised in power level by the low-power servoamplifier and applied to the control winding of the servomotor generator. Since the generator is directly attached to the motor armature, it has as its output a 400-cps voltage that is directly proportional to the motor speed. This voltage, 180° out of phase with the voltage going to the servoamplifier from the modulator, is mixed with the a-c drive voltage as negative feedback, and is used to damp oscillations of the drive elements of the servo. The controlled variable B is a mechanical shaft position of the brush arms of five B-potentiometers. One of these potentiometer cards is used as the feedback element.

42. GYRO AZIMUTH SERVO

a. Purpose. One purpose of the gyro azimuth servo is to solve for the proper azimuth (A_G) to the predicted intercept point. A second purpose is to solve for the ground range to the intercept point ($-R_I$) which is sent to the time-of-flight predictor.

b. Components. The A_G servo consists of two sine-cosine potentiometers, a DC amplifier, a modulator, a low-power servoamplifier, a servomotor generator, and a geometric gain control. The inputs come from the X and Y channels of the intercept point solver. One output is a shaft motion that positions the A_G resolver in the gyro azimuth transmission system to transmit the gyro azimuth data to the launcher area. A second output, $-R_I$, goes from the sine-cosine potentiometers to the time-of-flight predictor. The scale factor in this servo is 1 mv/yard.

c. Geometric gain control. The servo system contains a circuit called a geometric gain control which causes the A_G servo system to have the same response when intercept points are at short ranges as at long ranges. Without the geometric gain control, the servo system would be less sensitive and would take longer to solve for the A_G angle at shorter ranges. This is undesirable since at short range, time is at a premium, A_G changes faster, and the A_G solution must be more accurate than at greater ranges. Geometric gain control in the A_G servo consists of increasing the gain of the A_G DC amplifier as the ground range to the intercept point decreases.

43. MATHEMATICAL ANALYSIS OF GYRO AZIMUTH SERVO

a. The gyro azimuth angle. The A_G servo solves for the gyro azimuth (A_G) of the predicted intercept point by comparing elements of the triangle shown in figure 22. The gyro azimuth (A_G) is the clockwise angle (measured in mils at the designated launcher) from north (Y-axis) to the predicted intercept point.

b. Example. A gyro azimuth of 5, 600 mils indicates that the predicted intercept point is exactly northwest of the launcher designated to fire. The right triangle whose legs are X_I and Y_I, define the A_G angle. This can also be stated as follows: The A_G angle is the angle the tangent of which is X_I divided by Y_I. Or,

$$\tan A_G = \frac{X_I}{Y_I}. \tag{14}$$

The tangent of any angle is its sine divided by its cosine. Therefore,

$$\tan A_G = \frac{\sin A_G}{\cos A_G}. \qquad (15)$$

By equating the expressions for tan A_G given in equations (14) and (15) equation (16) is obtained:

$$\frac{X_I}{Y_I} = \frac{\sin A_G}{\cos A_G}. \qquad (16)$$

Cross-multiplying in equation (16), it is found that

$$Y_I \sin A_G = X_I \cos A_G \qquad (17)$$

or

$$Y_I \sin A_G - X_I \cos A_G = 0. \qquad (18)$$

Equation (18) is the mathematical expression used by the A_G servo to determine the gyro azimuth. For any given X_I and Y_I, the A_G solution is correct when equation (18) is satisfied, that is, when the difference between $Y_I \sin A_G$ and $X_I \cos A_G$ is zero.

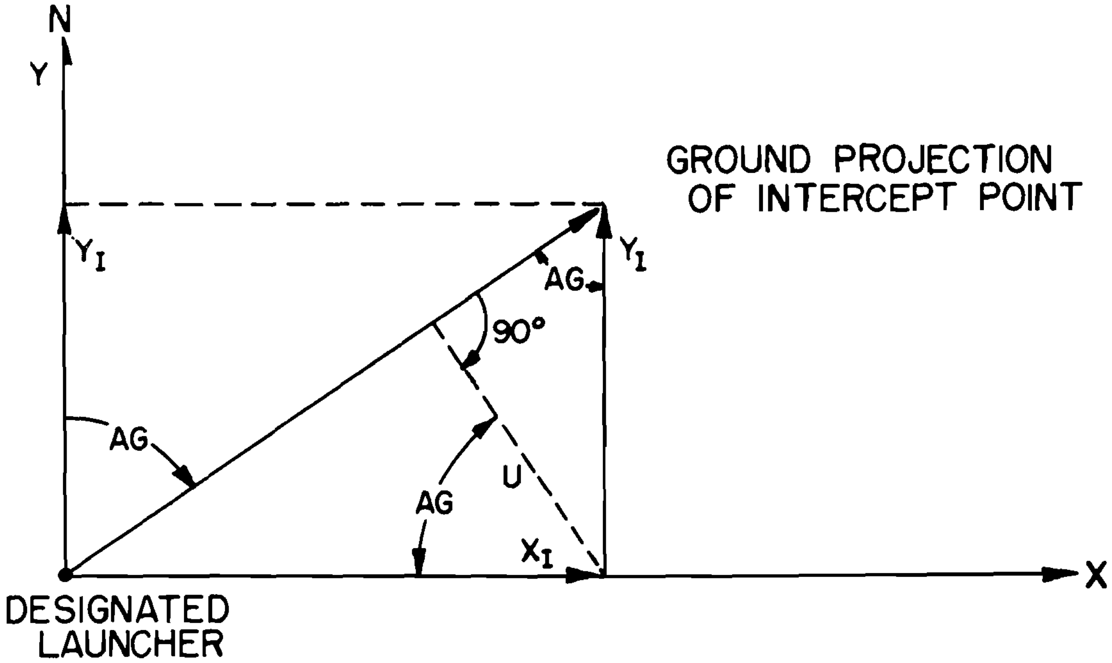

Figure 22. Trigonometry involved in A_G solution.

c. Alternate solution. Equation (17) can also be obtained by dropping a perpendicular line (U) from the right angle to the hypotenuse as shown in figure 22. Two smaller triangles are thus created. The small left triangle yields the relationship: $U = X_I \cos A_G$. The right small triangle gives $U = Y_I \sin A_G$. By equating the two expressions for line U, equation (17) above is obtained.

44. SIMPLIFIED FUNCTIONAL OPERATION OF GYRO AZIMUTH SERVO

a. General. The functional block diagram of a basic servo (closed loop control) system is shown in ST 44-161-1y, page 109. The controlled variable (output) is compared with the reference inputs by an error measuring means (comparator). The existing error between the input and output is used to actuate elements which control the output variable to eliminate the existing error. In most servos the reference inputs are constantly changing so that an actuating error always exists in order to keep the output following the inputs as closely as possible. The gyro azimuth (A_G) servo system conforms generally to the basic servo system. The elements of the A_G servo are arranged as a basic closed loop servo.

b. Inputs and outputs. The reference inputs ($\pm X_I$, $\pm Y_I$) from the intercept point solver are applied to sine-cosine potentiometers (feedback elements) whose brushes are positioned by the feedback. The outputs of the feedback elements are $-X_I \cos A_G$ and $+Y_I \sin A_G$ which are compared in the A_G DC amplifier (comparator). Note that each of the inputs to the A_G amplifier ($+Y_I \sin A_G$, for example) is the product of a reference input ($+Y_I$) and a feedback function ($\sin A_G$). When the two inputs to the A_G amplifier are equal and opposite the equation $Y_I \sin A_G - X_I \cos A_G = 0$ is satisfied, the actuating error (d-c voltage) approaches zero, and the proper solution for A_G has been obtained.

c. Servo operation. The d-c input (actuating error) to the modulator is used to control the phase and amplitude of a 400-cycle a-c voltage to properly control the servomotor generator as was described in paragraph 41b above. The output (controlled variable) is a mechanical shaft motion which positions the A_G resolver. The A_G resolver transmits A_G data to the launcher area. The feedback is a mechanical shaft motion which positions the A_G potentiometer brushes.

45. TIME-OF-FLIGHT PREDICTOR.

a. Purpose. The purpose of the time-of-flight predictor is to determine a correct time of flight of the missile from the end of fire + 7 seconds to the predicted intercept point. The time is indicated in a window on the front of the servo cabinet.

b. Dead time. At this stage of operation, there is a constant dead time (7 seconds) reckoned to provide compensation for mechanical functions which must always occur between the instants of FIRE and ROLL STABILIZATION. Dead time (t_d) also appears in a servo cabinet window. Together, these figures yield the total time which will elapse between FIRE and intercept.

c. Inputs. The inputs to the servo are $-R_I$ from the A_G servo, and $R_B + D_B \cos B$ from the ballistic circuits, all with a scale factor of 1 mv/yd. In a previous discussion of the determination of the intercept point, the time of flight of the missile was assumed. The time was needed, along with target velocity, to calculate the distance the target would move between the instants of fire and intercept. In this discussion, correct values of intercept point, gyro azimuth, and ballistic data will be assumed.

d. Outputs. The controlled outputs are mechanical positioning of potentiometer brush arms and voltage analogs of the time of flight.

e. Components. Some of the physical components which constitute the time-of-flight servo are also used in the time-to-intercept servo during the steering phase of computer action. Two identical input networks t_{p1} and t_{p2}, are located below the t-amplifier in the left amplifier frame. During the prelaunch phase, network t_{p1} feeds the error voltage to the t-amplifier. The t_{p2} network is used in conjunction with the time slew control circuits. The normal computer servo modulator and low-power servoamplifier which follow the t-amplifier are located in the amplifier cabinet. The modulator chassis is shared with the t_{SLR} circuit. The time-to-intercept servo assembly at the center of the computer servo cabinet contains the remainder of the components: servomotor generator and associated gearing and switches, a slew motor, and potentiometer cards. Two d-c amplifiers, +t and -t, are also located in the left amplifier frame.

46. MATHEMATICAL ANALYSIS OF TIME-OF-FLIGHT PREDICTOR

a. Ballistic vs computed range. The equation used by the time-of-flight predictor to solve for a time of flight is

$$-R_I + R_B + D_B \cos B = 0. \qquad (19)$$

$R_B + D_B \cos B$ is an implicit function of time discussed in paragraph 33a above. In figure 23, the ballistic ground range is shown to be composed of two parts. The first is the distance from the launcher to the ground projection of the center of the constant time circle being used. The second is the ground projection of the constant time circle radius. If the ballistic elevation is B, then simple trigonometry gives the ground projection of the radius as $D_B \cos B$, which is always a positive quantity. $-R_I$ is also a function of time, since

$$R_I = X_I \sin A_G + Y_I \cos A_G. \qquad (20)$$

and $X_I = X_T - X_L + \dot{X}_P (t + t_d)$ (8)

and $Y_I = Y_T - Y_L + \dot{Y}_P (t + t_d).$ (9)

Substituting these in equation (20)

$$\begin{aligned} R_I &= X_T \sin A_G - X_L \sin A_G + \dot{X}_P (t + t_d) \sin A_G \\ &\quad + Y_T \cos A_G - Y_L \cos A_G + \dot{Y}_P (t + t_d) \cos A_G \\ &= (X_T \sin A_G + Y_T \cos A_G) - (X_L \sin A_G + Y_L \cos A_G) \\ &\qquad + (\dot{X}_P \sin A_G - \dot{Y}_P \cos A_G)(t + t_d) \end{aligned}$$

$$R_I = R_T - R_L + (t + t_d)\, \dot{R}_P. \qquad (21)$$

This quantity is also positive by definition. Both represent the horizontal range from the launcher to the intercept point. R_I is the computed range, and $R_B + D_B \cos B$ is the ballistic range. If the missile is to destroy the target, the ballistic and computed ranges must be equal.

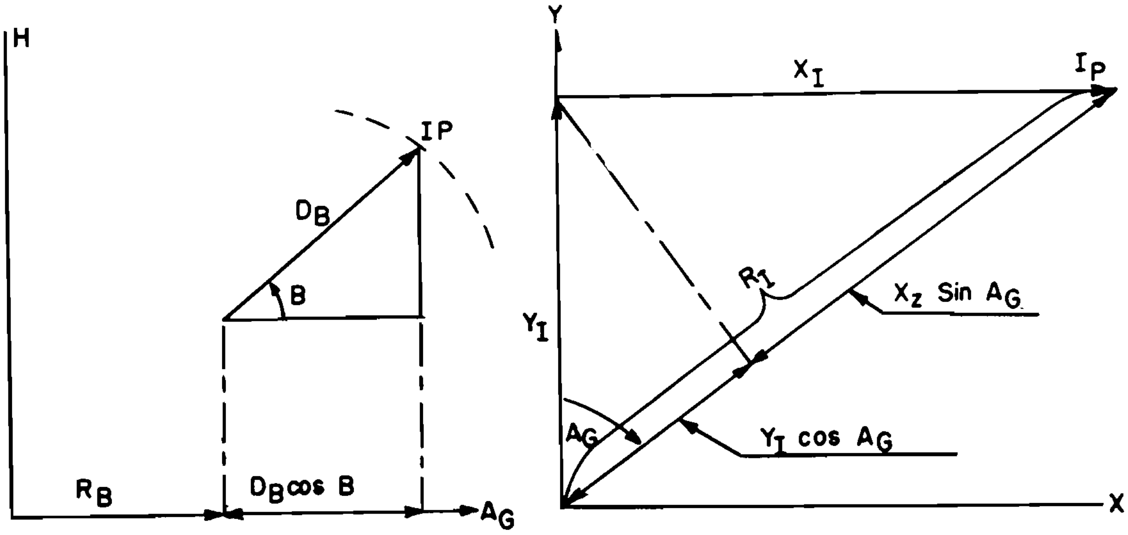

Figure 23. The horizontal range triangles.

b. Solving for time. The problem is to find a value of time that will make equation (19) true. The t servo solves the problem by the method of successive approximations. The time-of-flight predictor selects a value of t, substitutes this t in the equation (19), and tests the solution. If the solution is not zero, another value of t is used in the same process. Eventually, the correct time of flight is determined.

47. SIMPLIFIED FUNCTIONAL OPERATION OF TIME-OF-FLIGHT PREDICTOR

a. Definition. The time of flight determined by the time-of-flight predictor is the time required by the missile to reach the intercept point from its position at the end of roll stabilization. Dead time takes care of the period between FIRE and the end of roll stabilization. During the prelaunch calculations t is constantly changing and t_d is a constant 7 seconds.

b. Time servo. The circuits used to find the time of flight readily fit into the picture of a basic closed loop control system. The controlled variable t is used to modify the two reference inputs. A comparator (t-amplifier and input network) compares the two references. The difference between the references is the actuating error e. The error passes on to the drive elements of the servo. In the modulator the d-c error voltage becomes an a-c error voltage. The low-power servoamplifier raises the power to a level sufficient to drive the servo-motor generator. This motor positions a number of time potentiometer brush arms. Alteration of the brush arm positions affects both reference inputs. This is done in the feedback elements (intercept point solver and ballistic circuits). The outputs are a mechanical position of the brush arms and a d-c analog voltage.

Section III. THE INITIAL TURN SECTION

48. THE INITIAL TURN PROBLEM

a. General. Two situations exist which threaten to reduce the ability of the Nike I battery to attack targets under all conditions. These situations arise from the limitation imposed upon the Nike I battery by the need for a booster disposal area and from the maximum tracking rates of the missile-tracking radar.

b. Inclination toward booster disposal area. The launcher erecting arm is slanted 5° from the vertical toward the booster disposal area. Since the launcher is fixed, it cannot be pointed in any other direction. One situation will arise when the intercept point is in a direction far removed from the direction of the booster disposal area. By the time the missile has completed the 7g dive, it will be flying almost horizontally at a high ground speed and may be a long way from the gyro reference plane. If no order were applied prior to the completion of the dive, the steering orders required to bring the missile back into the gyro reference plane might result in turns which would exceed the 70° gimbal limits of the missile and thus prevent the missile from ever intercepting close-in targets. The initial turn section eliminates the possibility of this situation by producing outputs which cause the missile flight path, after roll stabilization, to remain in a plane parallel to the gyro reference plane. Although the missile will be a short distance away from the gyro reference plane at roll stabilization, the initial turn section prevents this distance from increasing as the missile dives. Thus, after the missile completes the dive and the ON TRAJECTORY signal is received, the steering orders required will not cause the missile to exceed the 70° gimbal limit in turning onto the intercept course.

c. Excessive tracking rates. This situation arises if the missile on its outbound flight path were to pass close enough to the missile-tracking radar to exceed the maximum tracking rates of the antenna, thus causing the radar to lose the missile. The initial turn section eliminates this possible situation by determining whether or not the missile is in danger of passing too close to the MTR. If this danger exists, the initial turn section produces outputs which cause the missile to skirt the MTR. When the missile has safely passed the MTR, the initial turn section removes the skirting turn order, and if necessary, causes the missile to fly parallel to the gyro reference plane until the ON TRAJECTORY signal is produced. The initial turn section must perform properly at all times to make certain that the two situations just discussed will never occur.

49. INITIAL CLIMB AND TURN ANGLES

a. General. Before launching any missile, the launcher erecting arm is normally tilted 5° from the vertical and the launcher is oriented in the direction of the booster disposal area. After the missile is launched, the booster separates from the missile and lands in the booster disposal area. A missile launched under these conditions never starts out vertically. Furthermore, since the booster disposal area and the intercept point are rarely at the same azimuth, the missile will seldom be headed in the direction of the intercept point. Consequently, the missile will have initial climb and turn angles different from 0°.

b. Effect of gravity. Even if the thrust on the tail-end of the missile is evenly distributed, the missile will be less vertical at roll stabilization than when on the launcher rail because of the effect of gravity on the missile. Thus, the initial climb angle and initial turn angle will be different from what they would be if gravity had no effect. Other factors which may throw the missile off the vertical are uneven booster thrust on the tail-end of the missile and transients which change missile direction when the missile and booster separate.

c. Initial turn and climb angles. Figure 24 represents the limits of initial turn and climb angles which may exist at the end of roll stabilization due to the factors which have just been discussed. These limits are ±15° for both the climb and turn angles. In the ideal situation, it would be desirable to launch the missile vertically; then, when the 7g dive is applied, the missile would execute a pure dive in the gyro reference plane. However, because the missile is not launched vertically and is always launched toward the booster disposal area, the initial climb angle will not be 90° at roll stabilization, and the initial turn angle will not be 0°. When the 7g dive order is applied, the missile will execute a pure dive, but not in the gyro reference plane. If the 7g dive order were applied and no correction made for the nonzero initial turn angle, the trajectory of the missile over the ground would be as shown in figure 25. This figure represents a situation where the booster disposal area is in a direction 3,400 mils greater than A_G. The position of the launcher is shown by the letter L. In the section of the trajectory from L to A (fig 25), the missile may move out a considerable distance from the gyro reference plane.

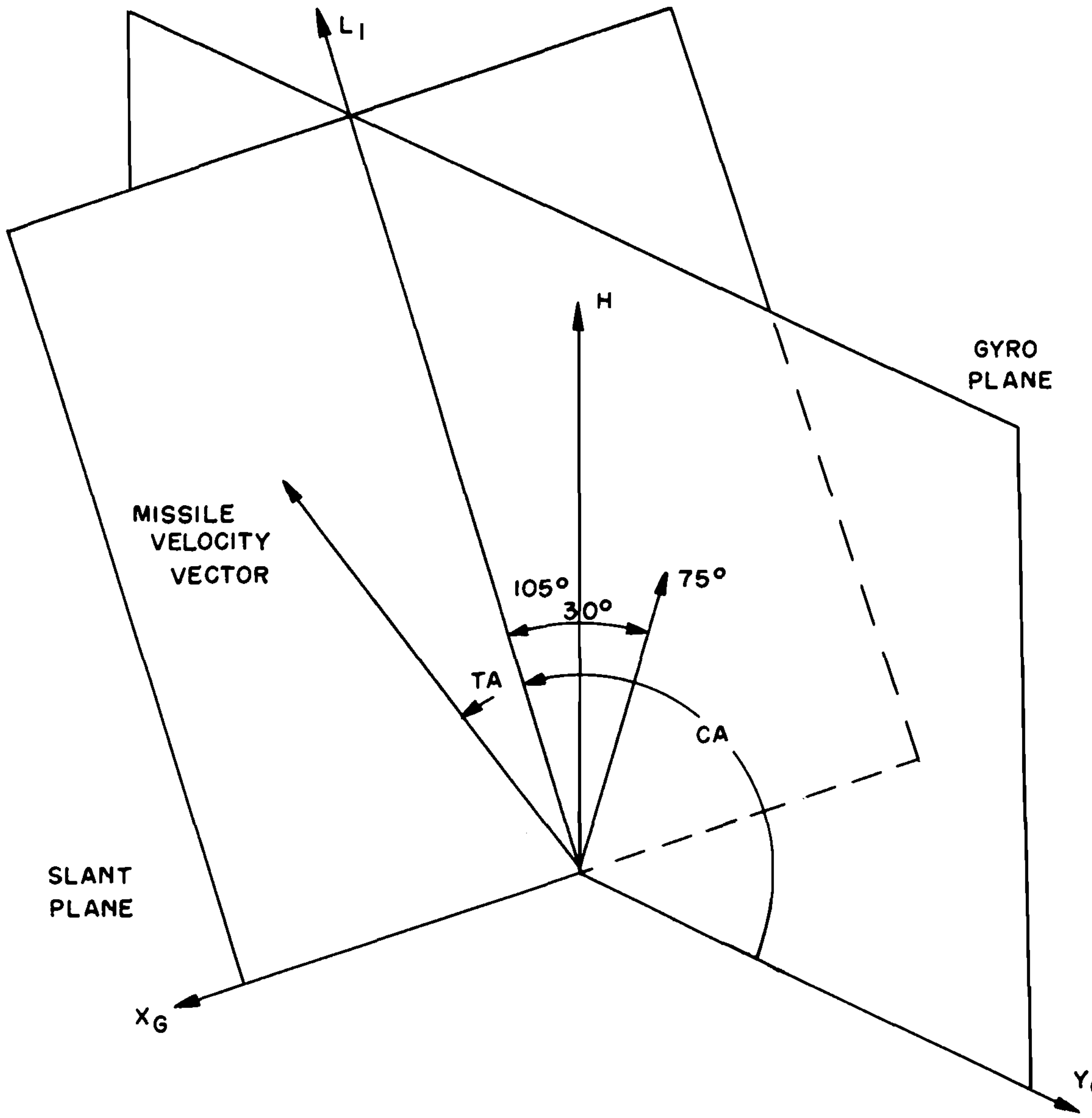

Figure 24. Limits of climb and turn angles during boost.

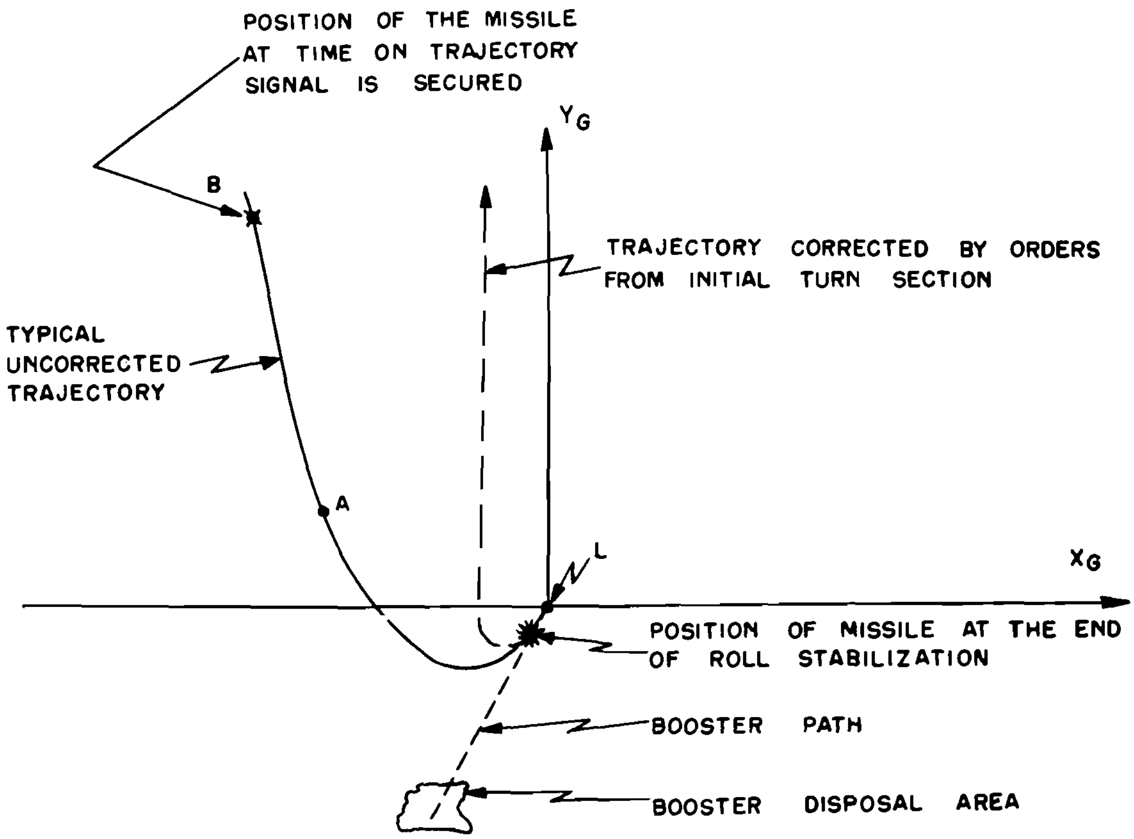

Figure 25. *Typical uncorrected missile trajectory.*

d. Initial turn orders. By the time the missile is at a ground distance of about 3 miles, the missile will be flying almost horizontally and will therefore have a high ground speed. By this time, the ON TRAJECTORY signal will have been received by the computer and the steering phase will have commenced. Turn orders are required to turn the missile back toward the gyro reference axis. However, if the remaining time to interpret is too short (target is close in), it may not be possible to guide the missile back toward the intercept point fast enough to intercept the target. The initial turn section produces appropi-ate outputs which result in turn orders that change the pure dive order to a turning dive order and results in a 0° turn angle. Once the 0° turn angle is established, the initial turn section continues to produce outputs which cause the missile to maneuver so that the turn angle remains 0°. The missile then will maneuver in a plane parallel with and close to the gyro reference plane and hence the ground path of the missile is parallel to the Y_G axis.

50. DEVELOPMENT OF THE SKIRTING TURN ORDER

If the missile passes too close to the missile-tracking radar, the missile velocity may exceed the maximum tracking rates of the missile-tracking radar and as a result, the MTR may lose the missile. This problem is shown in figure 26. The circle around the point marked MTR indicates the ground area which must be skirted by the path of the missile over the ground. This area is called the critical zone. If the missile path over the ground passes through this zone, the maximum tracking rates of the MTR antenna will be exceeded and the MTR will lose the missile. The size of this circular area depends upon several things. One factor is the maximum azimuth tracking rate of the MTR (750 mils per second). Another factor is the ground velocity of the missile as it passes the MTR antenna. The ground velocity of the missile, in turn, depends on the direction of the missile in relation to the direction of the intercept point and upon the initial climb and turn angles which exist at roll stabilization.

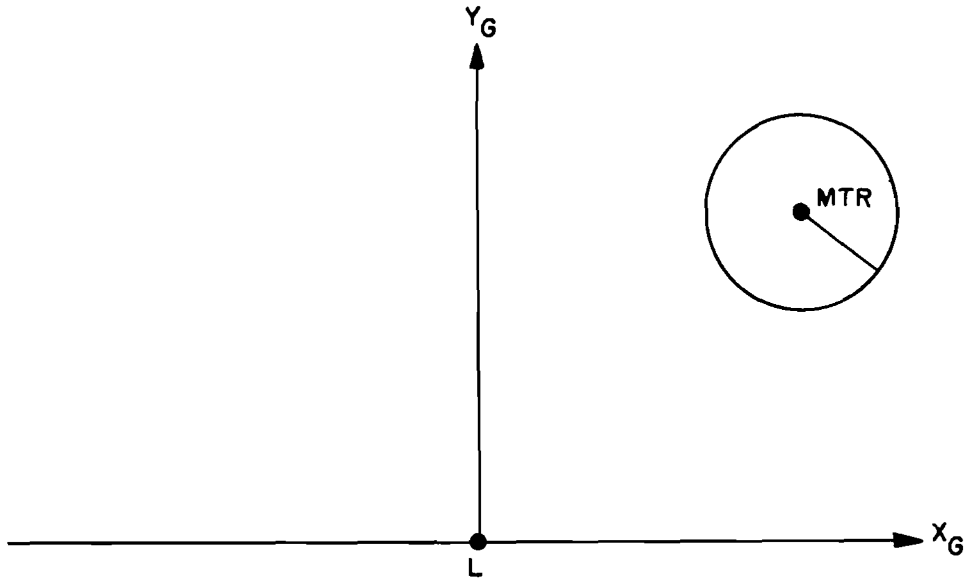

Figure 26. Critical zone of MTR.

51. CRITICAL TURN ANGLE

The critical turn angle is defined as the turn angle that, if applied at roll stabilization, would cause the missile to fly directly over the missile-tracking radar antenna. The computer determines this critical turn angle. Figure 27 shows several possible positions of the missile-tracking radar with respect to the ground position of the missile at roll stabilization, and several ground projections of the missile flight path.

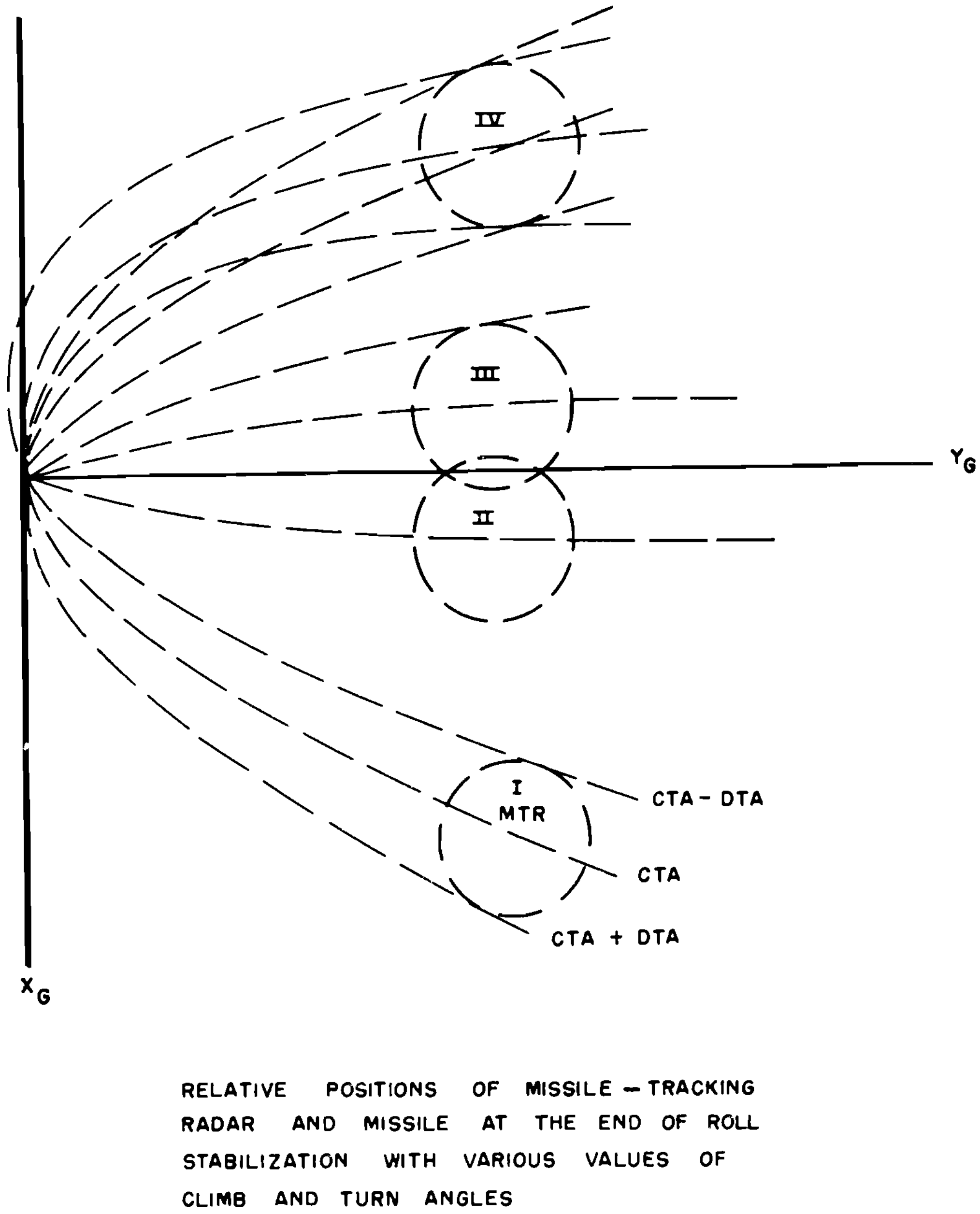

Figure 27. Ground plot of missile trajectories and MTR positions.

52. DIFFERENCE TURN ANGLE AND SKIRTING TURN ANGLE

a. General. The initial turn section also determines the difference turn angle. The DTA is the angular difference between the CTA and the skirting turn angle (STA). The STA is the turn angle which must be established and maintained by the missile after roll stabilization to make sure that the subsequent ground projection of the missile's path just skirts the critical zone on the side closest to the gyro reference plane.

b. Positive and negative values of DTA. There is always a positive and a negative value of DTA. Figure 28 shows the CTA, plus and minus difference turn angles, and the skirting turn angle. The DTA is positive when it is measured clockwise from the MV axis; it is negative when measured counterclockwise from the MV axis. The initial turn section always chooses the DTA angle which, when added algebraically to the CTA, will yield an STA that will result in the skirting ground path closest to the Y_G axis. The CTA depends on the position of the missile with respect to the MTR at the end of roll stabilization.

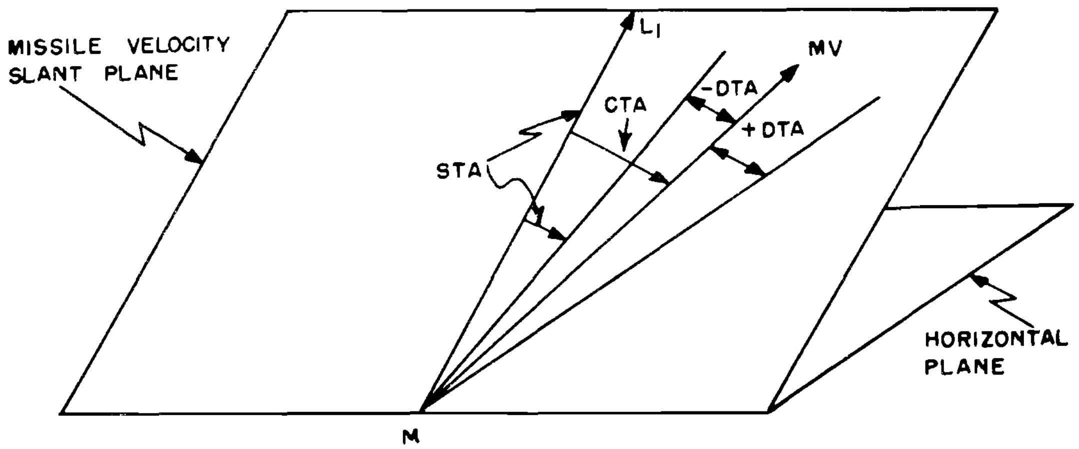

Figure 28. CTA, DTA, and STA.

c. Location of missile with respect to MTR. There are a number of ways to describe the location of the missile with respect to the MTR. The method which has been chosen in the operation of the initial turn section is to designate the missile position in terms of ground range from the TTR to the launcher, and the distance along the X_G axis from the MTR to the ground projection of the missile. In the initial turn computer, the correct value of the CTA for any given position depends on the climb angle (CA), the distance along the X_G axis from the MTR to the missile ground projection, and the ground range from the TTR to the launcher. The ground range from TTR to launcher is an approximation of MTR to launcher distance. The correction for this error in accuracy is made in the DTA amplifier circuits. Area I (fig 27) represents a position of the MTR to the right of the Y_G axis. The ground path of the missile which results when the missile attains and then maintains the CTA is shown (fig 27) passing through the center of area I. The CTA is positive when the MTR is to the right of the Y_G axis. Note that two ground paths pass through the center of area IV. The coarse broken curve represents the ground path of a missile with a CTA which depends on a climb angle equal to or less than 90°. The fine broken curve represents the ground path of a missile with a CTA which is dependent upon a climb angle greater than 90°. The critical turn angles which would result in ground paths through the center of areas III and IV are negative because the MTR is located in each case to the left of the Y_G line (X_G is positive). Figure 27 also shows the missile ground paths which would result from the establishment of initial turn angles determined by adding CTA and DTA together. The ground path which skirts the bottom of area I is represented by CTA + DTA. The path which skirts the top of area I is represented by CTA - DTA. This ground path also represents the STA for area I. The DTA depends on the radius of the critical zone and on the ground range from the MTR to the missile ground projection. In the initial turn section, the DTA depends on the value of the climb angle (CA) and on the ground range from the MTR to the projection of the missile position on the ground plane (R_M). It is not likely that the missile will pass over an MTR sited in a position such as those represented by areas I and IV. Therefore, it is not necessary that the missile fly a skirting turn trajectory after roll stabilization. However, the missile may pass over an area such as area II or area III; consequently, the missile must attain the correct STA to make sure that area II or area III is skirted.

d. Skirting turn angle solution. The initial turn section always chooses a value of DTA which is opposite in polarity to the computed CTA. The initial turn section decides whether or not a skirting turn angle must be established on the basis of the following information: Whenever the CTA is a larger angle than the DTA, the initial turn section will order the missile to fly with a 0° turn angle; whenever the DTA is a larger angle than the CTA, the initial turn section will order the missile to fly a skirting turn angle until it clears the critical zone. A simpler, more easily memorized rule is this: If the polarity of the STA (the sum of CTA and DTA) is the same as the polarity of the DTA chosen, then a skirting turn angle is required.

e. Example of the application of these rules. Suppose the MTR is sited in the center of area IV.

$$\text{CTA} = -400 \text{ mils}$$
$$\text{DTA} = 100 \text{ mils}$$

The initial turn section will select the positive value of DTA and add it to CTA.

Thus,

$$\text{STA} = -400 + (+100)$$
$$\text{STA} = -300 \text{ mils}$$

STA is not of the same polarity as +DTA; therefore, no skirting turn angle is required (the missile must fly at 0° initial turn angle). Suppose the MTR is sited in the center of area II.

$$CTA = 50 \text{ mils}$$

$$DTA = 70 \text{ mils}$$

The initial turn section will select the negative value of DTA, and add it to CTA. Thus,

$$STA = 50 + (-70)$$
$$STA = -20 \text{ mils.}$$

STA is of the same polarity as -DTA; therefore, a skirting turn angle is required.

53. INITIAL TURN SECTION BLOCK DIAGRAM

The initial turn section block diagram is found in TM 9-5000-8, page 116. The initial turn section consists of the following functional units: the missile distance converter, the critical turn angle (CTA) solver, the difference turn angle (DTA) solver, the initial turn control circuits, the turn angle reference potentiometer, the steering error converter, the fin order solver, and the radar cleared circuit. The ON trajectory circuit is discussed in relation to the initial turn section.

54. INITIAL TURN SECTION OPERATION

At MISSILE AWAY the initial turn section of the computer is enabled. However, the section does not control the missile until MISSILE AWAY + 4 (approximately 7 seconds after FIRE) at which time the missile is roll stabilized and prepared to accept orders. At MISSILE AWAY + 4, the 7g dive order circuit in the computer issues a dive order to the missile, causing it to dive toward the proper trajectory path to the target. At the same time, the initial turn section begins issuing orders which modify the dive order causing a turning dive in the direction necessary. The dive order will be continued until the missile is on the proper ½ g climb trajectory toward the target. Turn orders, either a skirting turn or a zero turn, will continue to be issued by the initial turn section until both the ON TRAJECTORY and RADAR CLEARED signals have been received. When both RADAR CLEARED and ON TRAJECTORY have been received, control of the missile becomes the function of the computer steering section. The period of missile flight which is under the control of the 7g dive order circuit and the initial turn section may be considered to be a programmed flight. Steering orders issued during this period have no relation to the steering error which may exist between the target and missile flight path, as in the steering section of the computer, but depend primarily on the relative position of the missile and the missile-tracking radar.

55. MISSILE DISTANCE CONVERTER

The solution for the critical turn angle and the determination of RADAR CLEARED require that the missile position components along the X_G and Y_G axis be known. It is the function of the missile distance converter to provide this information. The operation of this circuit is identical with those circuits in the steering section which rotate steering errors from the earth coordinate axis to the gyro coordinate axis. The outputs are X_{GM} and Y_{GM} (the initial turn section block diagram in TM 9-5000-8 , page 116, shows these as X_G and Y_G and should be corrected). The mathematical equation for X_{GM} is:

$$X_{GM} = X_M \cos A_G - Y_M \sin A_G \qquad (22)$$

for Y_{GM}

$$Y_{GM} = Y_M \cos A_G + X_M \sin A_G. \qquad (23)$$

56. CRITICAL TURN ANGLE SOLVER

The critical turn angle solver has the function of determining the critical turn angle. It receives the missile distance along the X_G axis (X_{GM}), the ground range to the center of the launching area (R_G), and the missile climb angle (CA) as inputs. From this information it calculates the critical turn angle. The output is a voltage representing the critical turn angle in mils. The actual solution is an empirical solution which approximates the value of the turn angle required to cause the missile to fly over the missile-tracking radar. Since the solution is not exact, an error is introduced into the difference turn angle solution to insure that the missile will skirt the critical zone around the missile-tracking radar.

57. DIFFERENCE TURN ANGLE SOLVER

The function of the difference turn angle solver is to produce an output which represents the difference turn angle. The unit receives as its inputs, the ground range to the missile (R_M), the missile climb angle (CA), and a bias voltage. The bias voltage represents the minimum radius of the critical zone around the missile-tracking radar and introduces the necessary error required by the empirical nature of the CTA solution. The output of the unit is a voltage which represents the difference turn angle in mils.

58. INITIAL TURN CONTROL CIRCUITS

The initial turn control circuits consist of two relay amplifiers and several relays. The critical turn angle negative relay amplifier detects the polarity of the voltage representing CTA and, through relay action, selects the opposite polarity of DTA. The skirting turn angle negative relay amplifier detects the polarity of the sum of CTA and DTA which is the skirting turn angle. Additional relay circuits make up the yes-no circuit which determines whether or not a skirting turn is required. If a skirting turn is required, proper voltages to cause this skirting turn are directed to the steering error converter. If no skirting turn is required, this fact is determined and orders requiring the missile to fly a zero turn angle trajectory are issued. Upon the receipt of the RADAR CLEARED and ON TRAJECTORY signals, the initial turn control circuits disconnect the initial turn section from the steering error converter and the steering section assumes control of the missile.

59. TURN ANGLE REFERENCE

A closed loop servo system requires a feedback signal. The Nike system may be considered to be a closed loop servo. During the initial turn phase of computer operation, the feedback signal is supplied by the turn angle reference circuit. The turn angle which the missile is actually flying is supplied from the turn angle reference circuit (a potentiometer in the turn angle servo) to the initial turn control circuits. In the initial turn control circuits, this turn angle is compared with the required turn angle. If these signals are not equal and opposite, an order requiring the missile to turn to the proper turn angle is directed to the steering error converter.

60. STEERING ERROR CONVERTER AND FIN ORDER SOLVER

The functions of the steering order converter and the fin order solver during the initial turn configuration are similar to those occurring during the steering configuration. The inputs to the steering error solver during the initial turn configuration however, are the required skirting turn angle (the sum of CTA and DTA); or a zero turn angle, and the turn angle reference. From this information the required turn order is determined and sent to the fin order solver. In the fin order solver this turn order is used to modify the dive order being sent to the missile to cause the missile to execute a turning dive.

61. RADAR CLEARED CIRCUIT

The radar cleared circuit consists of a relay amplifier and its associated relay. The function of the circuit is to determine when the missile has safely passed the missile-tracking radar. This is required because it is desirable to bring the missile onto a zero turn angle course as soon as possible after executing a skirting turn. As soon as the missile has safely skirted the critical zone the voltage representing Y_{GM} will become positive. This fact is made use of in the radar cleared circuit to operate a relay amplifier. The negative analog is used; therefore, when Y_{GM} becomes positive, the voltage representing $-Y_{GM}$ becomes negative. This negative voltage causes the relay amplifier to energize its associated relay and send the RADAR CLEARED signal to the initial turn control circuits.

62. ON TRAJECTORY CIRCUIT

The on trajectory circuit is shown on the block diagram of the initial turn section since the receipt of both RADAR CLEARED and ON TRAJECTORY is required before the steering section of the computer can assume control of the missile.

Section IV. THE STEERING SECTION

63. GENERAL

This section will provide the reader with a detailed block diagram discussion of the units which are incorporated in the steering section of the Nike I computer. Each unit discussion will include, where applicable, a general discussion, a mathematical analysis of circuit operation, and the function of the unit. The steering section block diagram is found in TM 9-5000-8, page 115.

64. MISSILE RATE CONVERTER

The missile rate converter is provided to perform the operation of rotating missile velocity components from earth to gyro coordinate axis. Missile velocities are determined originally along the earth coordinate axis. The climb and turn angles, however, are referenced to the gyro coordinate system. In order that we may use missile velocities to determine the climb and turn angle solutions we must determine the components of missile velocity which lie along the gyro coordinate axes.

65. MATHEMATICAL ANALYSIS OF MISSILE RATE CONVERTER

The components of missile velocity in earth coordinates must be resolved into components of missile velocity in gyro coordinates. In doing this, each component of velocity in earth coordinates is resolved into its components which lie parallel to the Y_G and X_G axes. The components of $\dot{Y}_M$ are: $\dot{Y}_M \cos A_G$ (parallel to the Y_G axis) and $\dot{Y}_M \sin A_G$ (parallel to the X_G axis) (fig 29). The components of $\dot{X}_M$ are: $\dot{X}_M \sin A_G$ (parallel to the Y_G axis) and $\dot{X}_M \cos A_G$ (parallel to the X_G axis) (fig 30). The values of $\dot{X}_{GM}$ and $\dot{Y}_{GM}$ are determined by summing all vectors which lie parallel to the X_G and Y_G axes. Doing this (fig 31), the following formulas are derived:

$$\dot{Y}_{GM} = \dot{Y}_M \cos A_G + \dot{X}_M \sin A_G \tag{24}$$

$$\dot{X}_{GM} = \dot{X}_M \cos A_G - \dot{Y}_M \sin A_G. \tag{25}$$

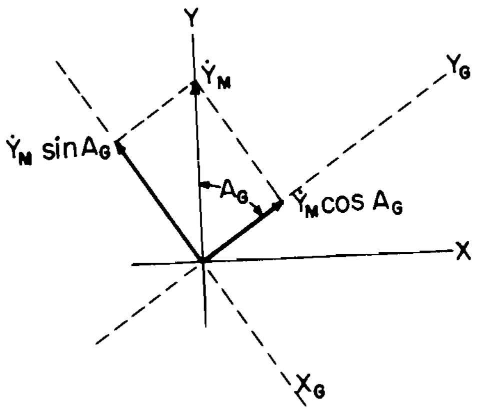

Figure 29. Resolution of $\dot{Y}_M$.

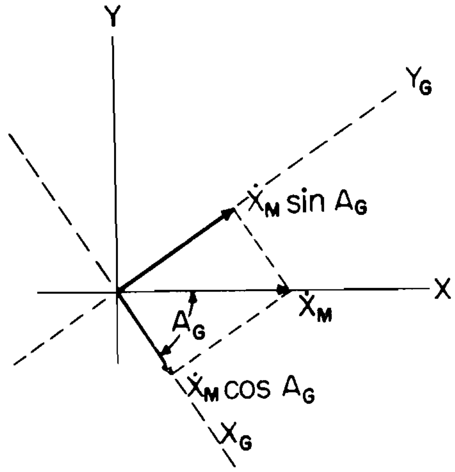

Figure 30. Resolution of $\dot{X}_M$.

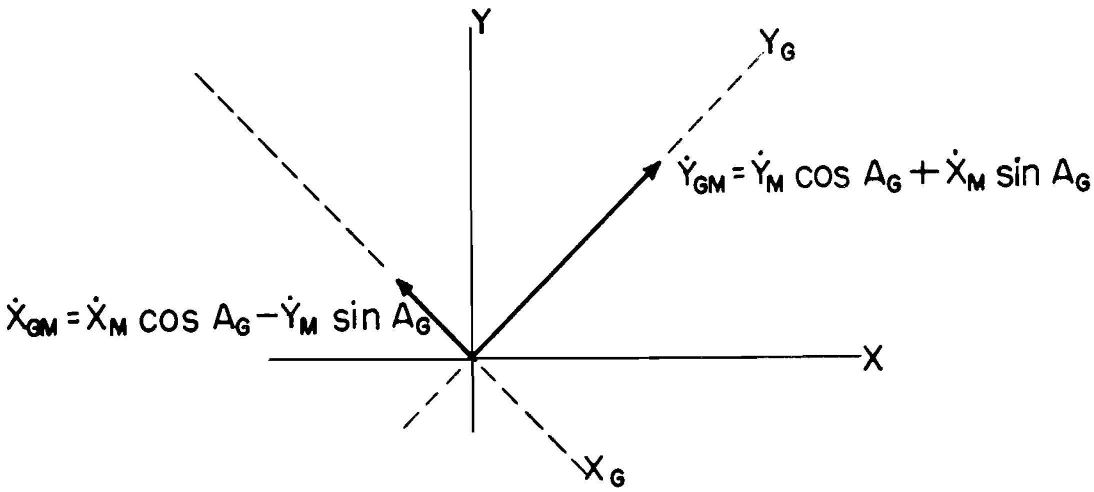

Figure 31. Result of summing of components of $\dot{X}_M$ and $\dot{Y}_M$.

66. CLIMB ANGLE SERVO

The climb angle servo has the function of determining the missile climb angle. Steering errors are determined in the earth coordinate system. These errors must be made available in a coordinate system related to the missile.

This is accomplished by rotating the error signals through the climb and turn angles. In order that this may be accomplished, the missile climb angle must be determined.

67. MATHEMATICAL ANALYSIS OF CLIMB ANGLE SERVO

The mathematics performed by the CA servo necessitates the use of $\dot{H}_M$ from the missile differentiator and $\dot{Y}_{GM}$ from the missile rate converter. The servo is constructed so that it must compare two values to arrive at a solution. The formula which is used by the CA servo may be derived by a trigonometric solution. In figure 32, the climb angle may be calculated by the formula:

$$\tan CA = \frac{\dot{H}_M}{\dot{Y}_{GM}} = \frac{\dot{H}_{GM}}{\dot{Y}_{GM}} \tag{26}$$

and since

$$\tan CA = \frac{\sin CA}{\cos CA}$$

this equation may be written thus:

$$\frac{\sin CA}{\cos CA} = \frac{\dot{H}_{GM}}{\dot{Y}_{GM}}. \tag{27}$$

When rearranged into a form which the computer can use, the equation is:

$$\dot{Y}_{GM} \sin CA - \dot{H}_{GM} \cos CA = 0. \tag{28}$$

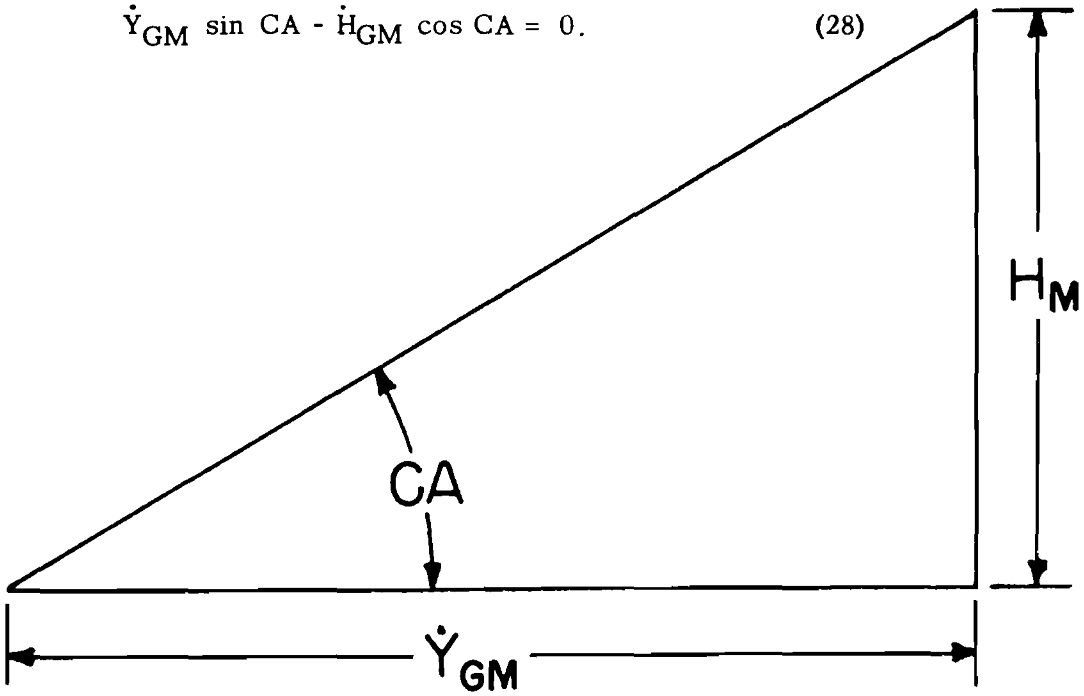

Figure 32. Trigonometric solution of the climb angle.

68. FUNCTIONAL OPERATION OF CLIMB ANGLE SERVO

Since the component of missile velocity, $\dot{H}_M$, along the earth vertical axis, H, is already parallel to the gyro vertical axis, it does not pass through the missile rate converter, but is applied directly to a sin-cos CA card, as an input to the CA servo. The quantity $\dot{Y}_{GM}$ is also applied to a sin-cos CA card, as an input to the CA servo. Taken from these sin-cos CA cards are the terms $+\dot{Y}_{GM}$ sin CA and $-\dot{H}_M$ cos CA. These terms are added by the CA amplifier. When $\dot{Y}_{GM}$ sin CA minus $\dot{H}_M$ cos CA equals zero, the input to the amplifier is zero and CA is correct. If the CA input to the amplifier, is positioned for a value of CA which is too large, the term $\dot{Y}_{GM}$ sin CA is larger than the term $-\dot{H}_M$ cos CA. This positive signal is inverted by the amplifier. A negative voltage at the output of the CA amplifier causes the CA servo to decrease the value of CA until the input to the summing amplifier is again zero. The d-c output of the CA amplifier is used to control the output of a 400-cycle modulator. The modulator output is then amplified to drive the CA servo.

69. TURN ANGLE SERVO

The function of the turn angle servo is to determine the missile's turn angle. As was pointed out in the discussion of the climb angle servo the steering errors are determined in the earth coordinate system. It is required that the steering errors be related to the missile in flight. To accomplish this the steering errors are rotated through the climb and turn angles. This requires that the missile turn angle be determined.

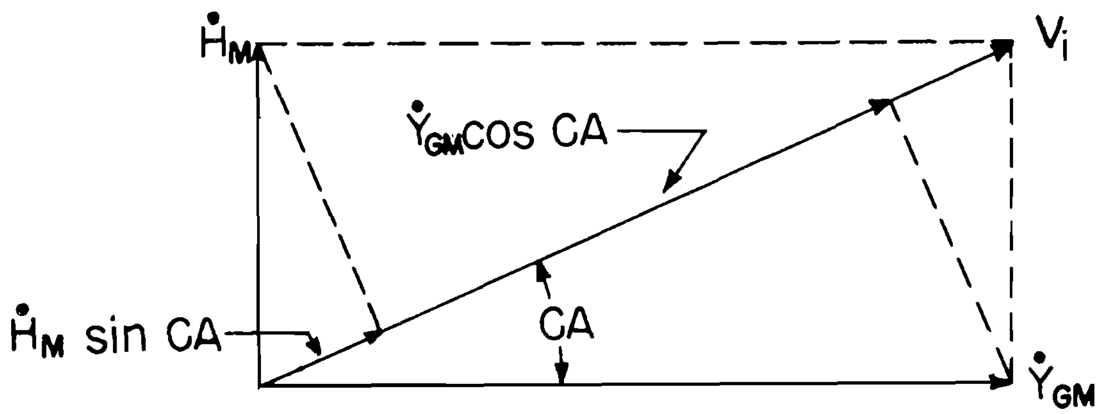

Figure 33. Solution for V_i.

70. MATHEMATICAL ANALYSIS OF TURN ANGLE SERVO

To calculate the turn angle of the missile it is necessary to use components of missile velocity which lie in the missile velocity slant plane. These are V_i, which lies along the L_i line and is calculated in terms of $\dot{Y}_{GM}$ and $\dot{H}_M$, and $\dot{X}_{GM}$, which is obtained directly from the missile rate converter. Both $\dot{Y}_{GM}$ and $\dot{H}_M$ have components which lie along the L_i line (fig 33). The component of $\dot{Y}_{GM}$ is $\dot{Y}_{GM}$ cos CA. The component of $\dot{H}_M$ is $\dot{H}_M$ sin CA. The component V_i is equal to the sum of these two components. Thus the equation for V_i is:

$$V_i = \dot{Y}_{GM} \cos CA + \dot{H}_M \sin CA.$$

The formula which the computer uses to solve for the turn angle may be calculated by using the trigonometric solution (fig 34). The turn angle is calculated from the relation:

$$\tan TA = \frac{\dot{X}_{GM}}{V_i} \qquad (29)$$

and since

$$\tan TA = \frac{\sin TA}{\cos TA}$$

the equation may be written thus by substituting in the equation for V_i:

$$\frac{\sin TA}{\cos TA} = \frac{\dot{X}_{GM}}{V_i} = \frac{\dot{X}_{GM}}{\dot{Y}_{GM} \cos CA + \dot{H}_M \sin CA}. \qquad (30)$$

When rearranged into a form the computer can use, the formula is:

$$\dot{Y}_{GM} \cos CA \sin TA + \dot{H}_M \sin CA \sin TA - \dot{X}_{GM} \cos TA = 0. \qquad (31)$$

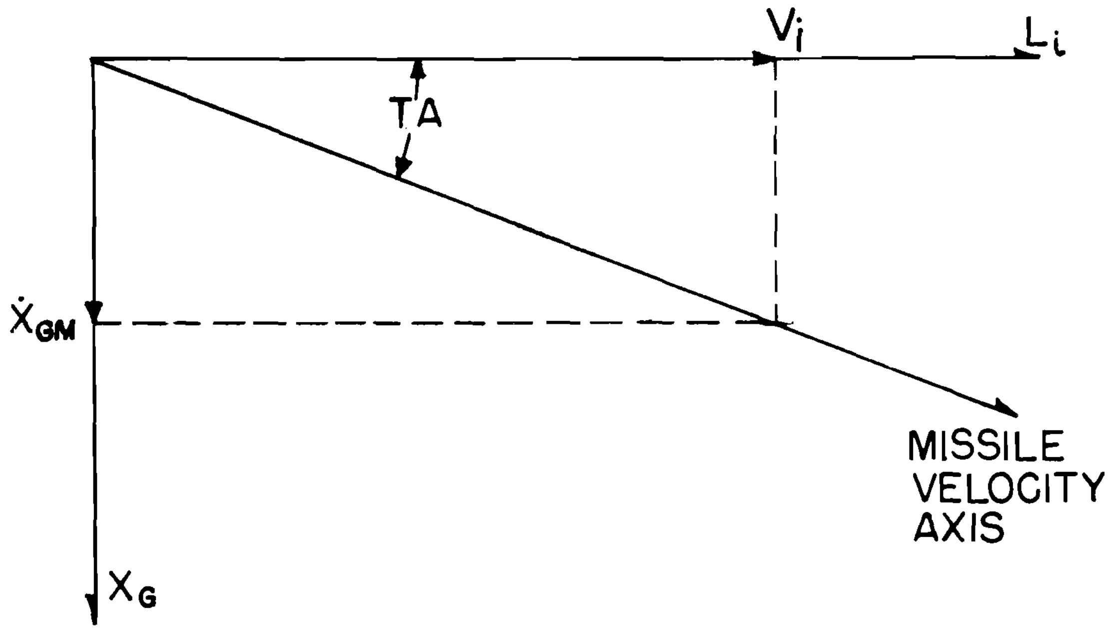

Figure 34. Determination of the missile turn angle.

71. FUNCTIONAL OPERATION OF TURN ANGLE SERVO

Potentiometer brushes positioned by the CA servo develop the terms $\dot{H}_M$ sin CA and $\dot{Y}_{GM}$ cos CA which are applied to sin-cos TA cards. These cards develop outputs, $\dot{Y}_{GM}$ cos CA sin TA and $\dot{H}_M$ sin CA sin TA, which are summed in the input network of the TA amplifier with $-\dot{X}_{GM}$ cos TA. This input is from another TA potentiometer which is fed by $\pm\dot{X}_{GM}$ from the missile rate converter. When:

$$\dot{Y}_{GM} \cos CA \sin TA + \dot{H}_M \sin CA \sin TA - \dot{X}_{GM} \cos TA = 0$$

there is no output from the TA amplifier, and the position of the TA servomotor output shaft represents the correct turn angle. The modulator, LPSA, and servomotor operate in the same manner as those in the CA servo loop. If the TA servo is positioned for a value of TA which is too large, then the term $\dot{Y}_{GM}$ cos CA sin TA + $\dot{H}_M$ sin CA sin TA will be larger than the term $\dot{X}_{GM}$ cos TA. The servomotor will be caused to turn to decrease the TA and the equation will equate to zero.

72. RADAR-TO-RADAR PARALLAX UNIT

In the computation of steering commands to be sent to the missile, the compu ter must determine the rectangular coordinates, X, Y, and H from the missile to the target. The origins of the rectangular coordinates which represent present position of the target and missile are the TTR and MTR. In figure 35, the target-tracking radar, the missile-tracking radar, the missile, and the target are pictured on the east axis. Note the available data for determining the distance between the missile and target. Voltages representing X_T, Y_T, and H_T, and X_M, Y_M, and H_M are obtained from the target and missile coordinate converters. If the computation is to be accurate, the distance between the two radars must be included in the calculation. This is done by voltages representing X_R, Y_R, and H_R, the rectangular coordinates of the missile-tracking radar using the target-tracking radar as the origin. These voltages come from handset potentiometers.

73. CLOSING SPEED SOLVER

The purpose of the closing speed solver is to calculate the desired or ideal closing velocity between the missile and the target. This velocity represents the theoretical closing velocity which would be present were the missile and target on exact collision courses. It is possible to determine the actual closing velocity by adding the known velocities of the missile and target. By comparison of the actual and desired closing velocities, steering errors are determined. If the two velocities are equal, the missile is on the correct course. If they are not equal, a steering error exists. This error is converted into commands to steer the missile onto the correct trajectory.

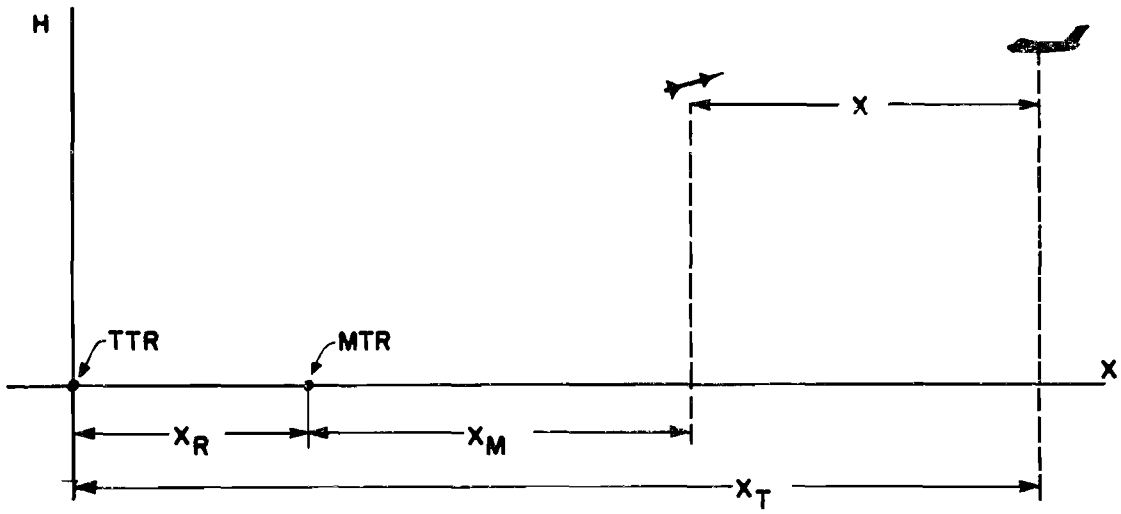

Figure 35. Determination of distance between missile and target.

74. MATHEMATICAL ANALYSIS OF CLOSING SPEED SOLVER

To compute the ideal closing velocity, the closing speed solver requires target and missile position and time-to-intercept data. Present position data are supplied in rectangular coordinates from the target and missile coordinate converters, and time is applied as a mechanical shaft position from the time-to-intercept servo. In addition, parallax information is supplied from the radar-to-radar parallax unit. Figure 35 shows a representation of missile, target, and MTR positions relative to the TTR. For simplicity, only the X-coordinate is shown. The X rectangular coordinate of the distance between missile and target is the desired quantity. The target is shown approaching in the first quadrant. The following equation is derived from the figure.

$$X = X_T - X_R - X_M \tag{32}$$

The quantity labeled X is always measured from the missile to the target. The other coordinates, Y and H, are obtained similarly:

$$Y = Y_T - Y_R - Y_M \tag{33}$$

$$H = H_T - H_R - H_M. \tag{34}$$

To obtain the ideal closing velocity from the rectangular coordinates of missile and target positions, the known rectangular coordinates are divided by the remaining time to intercept, t. The outputs of the closing speed solver are obtained by solving the right side of the equations:

$$\frac{X}{t} = \frac{X_T - X_M - X_R}{t} \tag{35}$$

$$\frac{Y}{t} = \frac{Y_T - Y_M - Y_R}{t} \tag{36}$$

$$\frac{H}{t} = \frac{H_T - H_M - H_R}{t}. \tag{37}$$

75. FUNCTIONAL OPERATION OF CLOSING SPEED SOLVER

Assume that the conditions shown in figure 35 are still current. Figure 36 shows a functional schematic of the closing speed solver. Since it is desired that the output of the amplifier have a positive polarity, and since inversion of voltage occurs in the amplifier, the negative equation (35) is used as the input. That is:

$$-\frac{X}{t} = \frac{-X_T + X_M + X_R}{t}.$$

Thus $-X_T$ is applied to terminal 4, $+X_M$ is applied to terminal 5, and $+X_R$ is applied to terminal 6. The sum of these for the situation shown in figure 35 is negative, since X_T is the larger term, and the output of the amplifier is a positive voltage. The sum of these voltages equals X. To divide by time to intercept to obtain $\frac{X}{t}$, a time potentiometer is incorporated in the feedback loop of the DC amplifier. The brush arm of the potentiometer is connected to and driven by the time-to-intercept servo. The output of the amplifier circuit then is a voltage representing $\frac{X}{t}$.

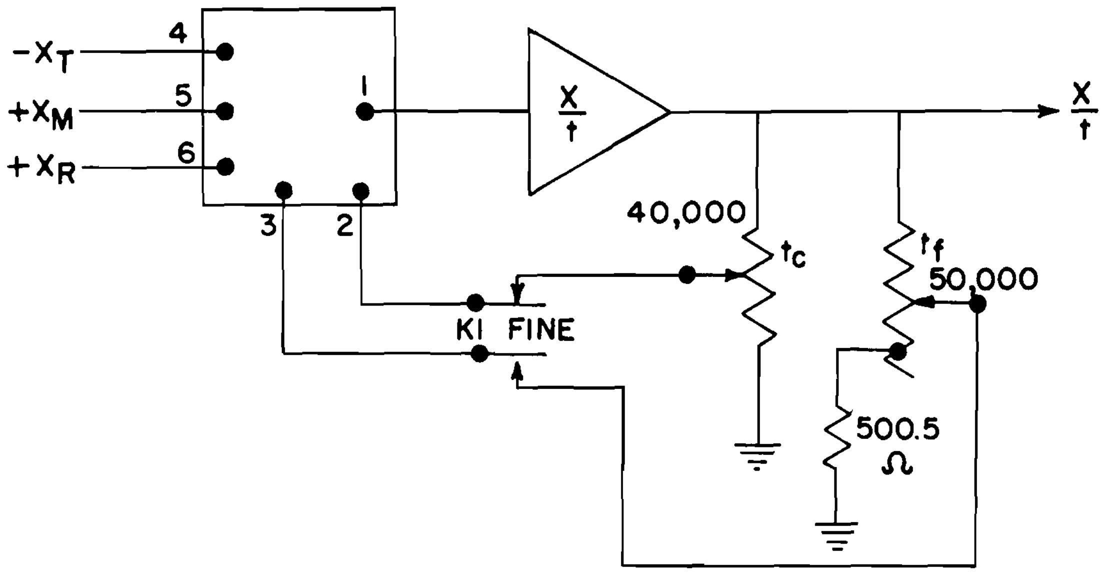

Figure 36. Closing speed solver, simplified schematic.

76. STEERING ERROR SOI VER

The purpose of the steering error solver is to compare the ideal closing velocity components ($\frac{X}{t}$, $\frac{Y}{t}$, $\frac{H}{t}$) from the closing speed solver with the actual velocity components ($\dot{X}_T - \dot{X}_M$, $\dot{Y}_T - \dot{Y}_M$, and $\dot{H}_T - \dot{H}_M$) from the target steering and missile differentiators. The inputs to the steering error solver are as follows: $\frac{X}{t}$, $\frac{Y}{t}$, and $\frac{H}{t}$ from the closing speed solver; $\dot{X}_T$, $\dot{Y}_T$, and $\dot{H}_T$ from the target steering differentiator; and $\dot{X}_M$, $\dot{Y}_M$, and $\dot{H}_M$ from the missile differentiator. The outputs are voltages which represent steering errors in terms of velocity components S_X, S_Y, and S_H. The steering error solver generally is composed of DC amplifiers which act to sum the voltage inputs.

77. MATHEMATICAL ANALYSIS OF STEERING ERROR SOLVER

a. Computation of closing velocity. To consider the mathematical operation of the steering error solver, one should first examine the computation of the actual closing velocity of the missile and target. This is done by subtracting the missile velocity as obtained by the missile differentiator from the target velocity as obtained from the target steering differentiator. Figure 37 shows the interception of a target by a Nike missile. Vectors representing the actual velocity components of missile and target are labeled $\dot{X}_M$ and $\dot{X}_T$, respectively. The resultant of the subtraction of these two velocity components is the actual closing velocity component $\dot{X}$. The equation for $\dot{X}$ is:

$$\dot{X} = \dot{X}_T - \dot{X}_M. \qquad (38)$$

b. Direction and signs. Note that for the case of figure 37, $\dot{X}_T$ is a negative quantity because the target is proceeding in a westerly direction. Conversely, $\dot{X}_M$ is a positive quantity because the missile is traveling eastward. The vector which represents $\dot{X}$ is pointing westward and that which represents $\frac{X}{t}$ is pointing eastward.

c. Steering errors. The next step to consider is the comparison of $\dot{X}$ and $\frac{X}{t}$, the actual and ideal closing velocity components. These quantities are added in the steering error solver and the resultant is called S_X. Stated in an equation:

$$S_X = \dot{X} + \frac{X}{t}. \qquad (39)$$

In figure 42, $\dot{X}$ and $\frac{X}{t}$ are shown exactly equal and opposite, indicating that the missile is on the correct intercept trajectory. Since the vector representing $\dot{X}$ and $\frac{X}{t}$ are equal and opposite in this case, S_X is zero. If under other conditions S_X is not zero, it is evident that the missile is not on the correct trajectory and that a steering error exists. The Y-coordinate is treated similarly:

$$\dot{Y} = \dot{Y}_T - \dot{Y}_M \tag{40}$$

and

$$S_Y = \dot{Y} + \frac{Y}{t}. \tag{41}$$

$\dot{Y}$ is the actual closing velocity component and S_Y is the steering error component.

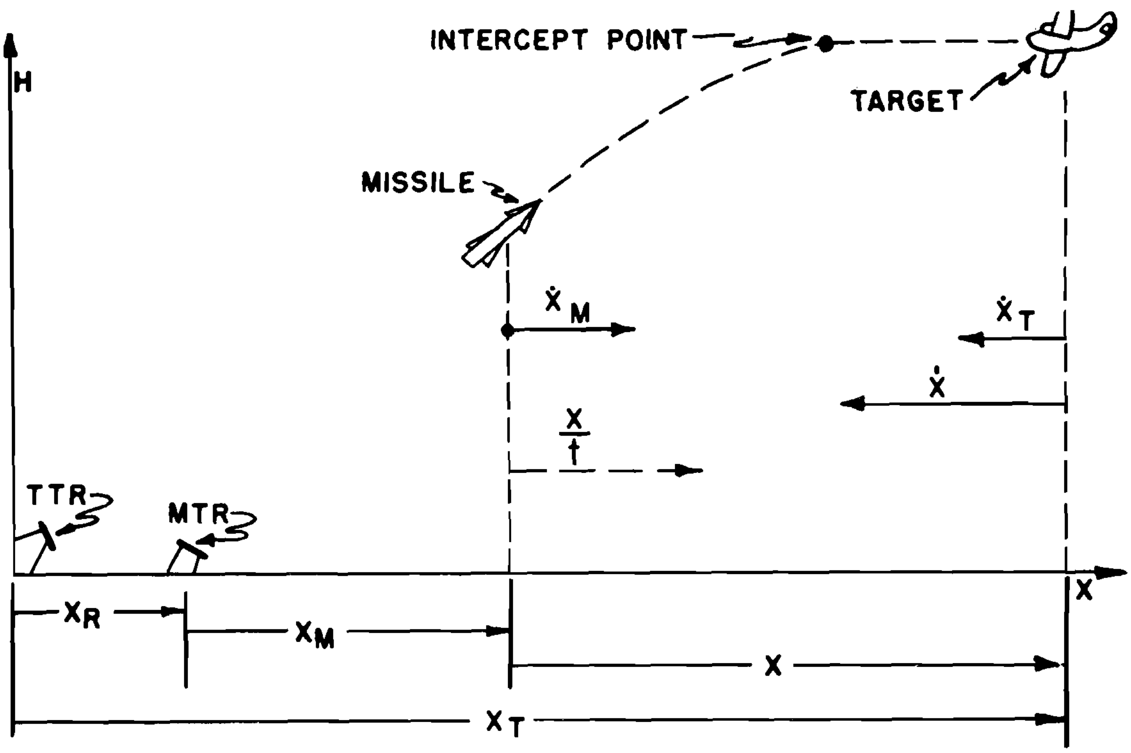

Figure 37. Comparison of ideal and actual closing velocities.

d. Vertical steering error and gravity. The equation representing the steering error in the vertical or H direction must be modified to allow compensation for the effect of gravity upon the missile. This equation is:

$$S_H = \dot{H} + \frac{H}{t} + \left(\frac{1}{4}+\frac{1}{6}\right) gt. \tag{42}$$

The $\left(\frac{1}{4}+\frac{1}{6}\right)$ gt provides the compensation necessary for the effect of gravity. In developing the steering circuits for the computer, a special missile trajectory had to be selected, and compensation for gravity effects had to be included to define the selected trajectory. Three possible trajectories were considered: a zero-g lift trajectory, a 1g lift trajectory, and a ½g lift trajectory.

78. COMPARISON OF TRAJECTORIES

a. Zero-g lift trajectory. To accomplish the zero-g lift trajectory, the OT (on trajectory) signal has to be received after the missile has dived so that it is aimed toward an imaginary intercept point in space (IP_f) above the actual intercept point (IP), a vertical distance equivalent to $\frac{1}{2} gt^2$ (fig 38). Once the missile arrives on trajectory, it drops toward the IP as time to intercept decreases to zero. The gravitational force acting on the missile causes the missile to nave a 1g downward acceleration. Since the distance $\frac{1}{2} gt^2$ decreases with time, it is evident that, with decreasing time to intercept, the IP_f is continually dropping toward the IP, and at time zero, the IP_f and the IP are in the same position. The advantage of this trajectory is that after OT is received, there is no requirement for a lift order to be constantly applied to keep the missile on trajectory, thus allowing minimum drag on the missile. The disadvantage of this trajectory is that the missile must climb to high altitudes for long range targets, thus reducing maneuverability, and increasing time to intercept.

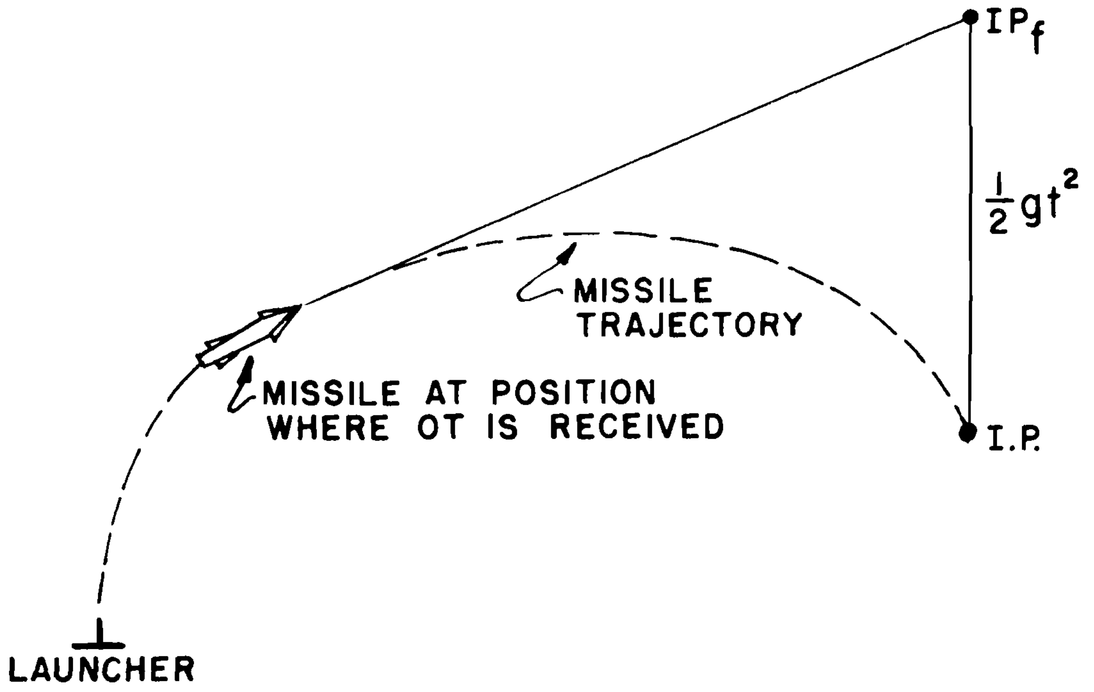

Figure 38. A zero-g lift trajectory.

b. One-g lift trajectory. To accomplish the 1g lift trajectory, the OT signal has to be received after the missile has dived so that it is aimed toward the IP (fig 39). Under ideal conditions the missile is then steered along a straight-line path to the IP. This requires that a climb order be constantly applied after

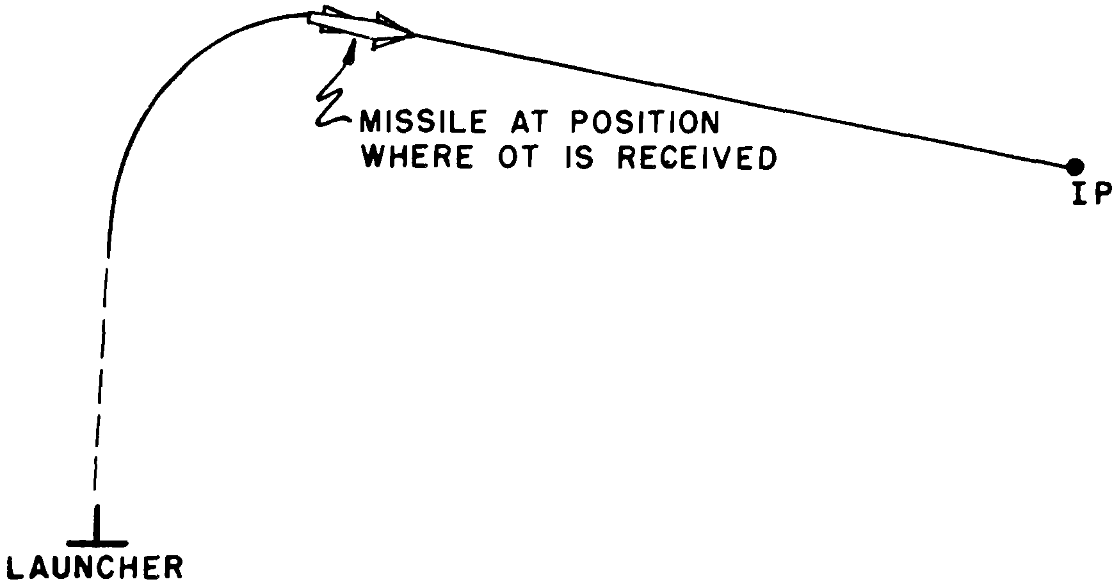

Figure 39. A 1g lift trajectory.

OT is received to oppose the downward pull of the gravitational force acting on the missile. The advantage of this trajectory is that it gives a shorter time to intercept; the disadvantage is that a 1g lift order must be constantly applied to the missile to keep it on trajectory with minimum pitch oscillations. The constantly applied lift order increases drag on the missile and results in reduced speed and maneuverability and shortened maximum effective range.

c. Half-g lift trajectory. The $\frac{1}{2}g$ lift trajectory represents a compromise between the two other trajectories. It employs the advantages of one type of trajectory to minimize the disadvantages of the other. To accomplish the $\frac{1}{2}g$ lift trajectory, the OT signal has to be received after the missile had dived so that it is aimed toward an imaginary intercept point, IP_f, a distance equivalent to $\frac{1}{4}gt^2$ above the IP (fig 40). From OT until intercept, a climb order must be constantly applied to oppose half of the gravitational force pulling downward on the missile. By countering one-half of the gravitational force acting on the missile, the missile is allowed to drop the distance $\frac{1}{4}gt^2$ between the IP_f and the IP as time to intercept decreases to zero. The advantage of this trajectory is that the missile flies low enough so that maneuverability is not sacrificed, and yet not so low that drag on the missile fins is excessive. This is the trajectory used for the Nike missile. The $\frac{1}{4}gt$ in formula (41) is used as a bias voltage to define the position of the IP_f above the IP, and is needed to develop the OT signal. Thus, when $\frac{H}{t} + \dot{H} + \frac{1}{4}gt$ is equal to zero, the missile will be on the correct intercept trajectory. The $\frac{1}{4}$ gt is added after OT to develop the $\frac{1}{2}g$ lift order sent to the missile. This steers the missile smoothly on a dropping course to the IP, eliminating large pitch oscillations.

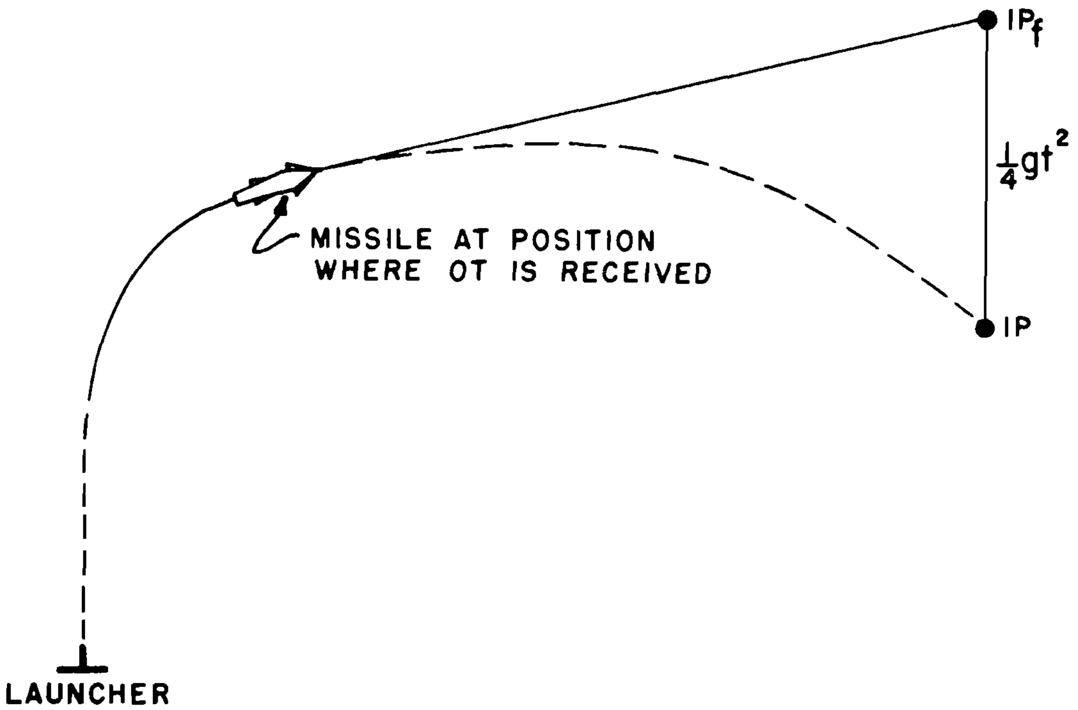

Figure 40. A ½g lift trajectory.

79. STEERING ERROR CONVERTER

The purpose of the steering error converter is to convert the steering error components derived from the steering error solver from earth to missile axis.

80. CONVERSION OF STEERING ERRORS FROM EARTH TO GYRO COORDINATES

Since the vertical steering error, S_H, is equal to S_{GH}, only the mathematics of the conversion of S_X and S_Y to S_{GX} and S_{GY} need be discussed. Figure 41 shows the basic form of the coordinate conversion problem. S_Y is a positive steering error component along the earth Y axis. The gyro axes, Y_G and X_G, and the gyro azimuth angle, A_G, are shown. This angle is measured between the earth Y axis and the Y_G axis. The angle measured between the earth X axis and the X_G axis is also equal to A_G because the X_G axis is perpendicular to the Y_G axis and the X axis is perpendicular to the Y axis. Project the vectors representing S_X and S_Y onto the X_G and Y_G axes. These projections produce four components, two derived from S_X and two from S_Y. Those derived from S_X are $S_X \cos A_G$ and $S_X \sin A_G$, while the components from S_Y are $S_Y \cos A_G$ and $S_Y \sin A_G$. Consider the transformation of S_X. The component lying along the X_G axis is positive going and equals $S_X \cos A_G$. The component lying along the Y_G axis is positive going and

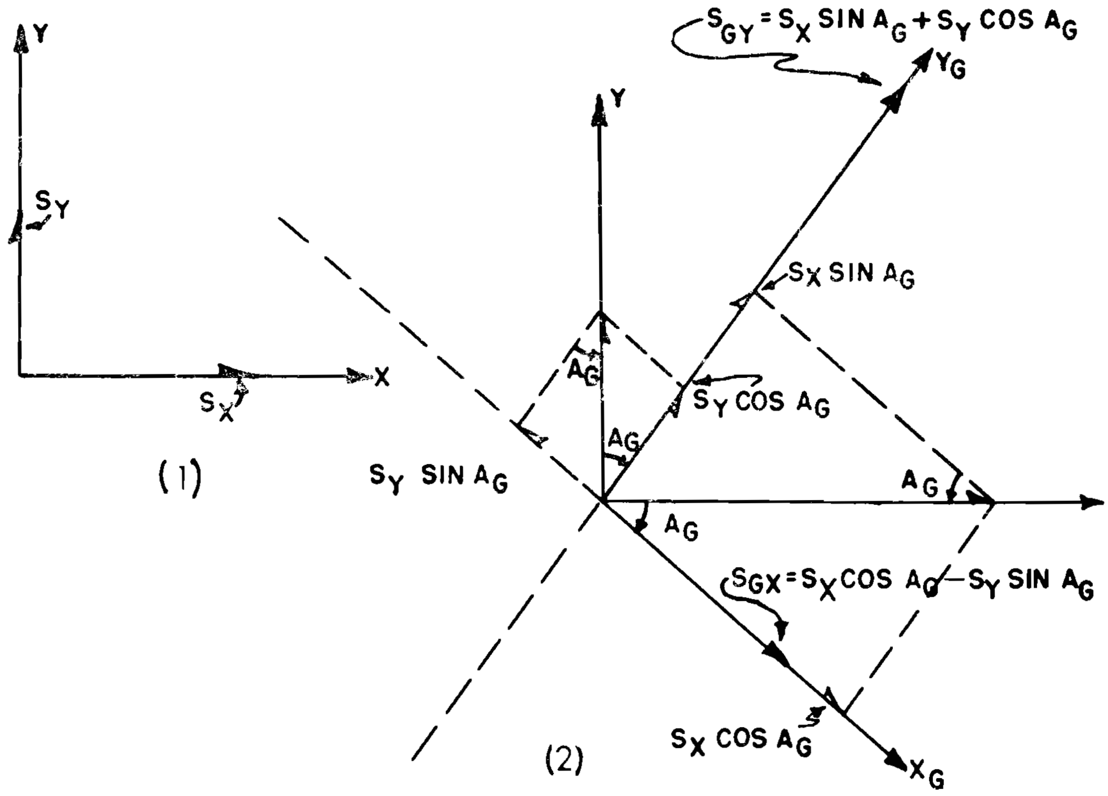

Figure 41. Conversion of steering errors from earth to gyro coordinate axes.

equals $S_X \sin A_G$. The latter expression is obtained directly when one considers that by geometry the angle opposite $S_X \sin A_G$ is equal to A_G. In similar manner, S_Y is converted into its two components. $S_Y \sin A_G$ lies along the X_G axis and is negative going. $S_Y \cos A_G$ lies along the Y_G axis and is positive going. The steering errors along the X_G and Y_G axes must now be summed to give S_{GX} and S_{GY}. Expressed as equations:

$$S_{GX} = S_X \cos A_G - S_Y \sin A_G \qquad (43)$$

and,

$$S_{GY} = S_X \sin A_G + S_Y \cos A_G. \qquad (44)$$

$S_Y \sin A_G$ is negative and must be subtracted from the positive term, $S_X \cos A_G$.

81. CONVERSION OF STEERING ERRORS FROM GYRO TO MISSILE COORDINATES

a. The planes and axes of reference. TM 9-5000-8, page 108, shows the horizontal plane, the gyro reference plane, and the missile velocity slant plane. The horizontal plane contains the gyro reference (Y_G) and gyro spin (X_G) axes. Perpendicular to the intersection of the X_G and Y_G axes is the gyro vertical axis, H_G. The H_G axis is contained in the gyro reference plane. The missile velocity axis represents the direction of motion of the missile and is contained in the missile velocity slant plane. The intersection of the missile velocity slant plane and the gyro reference plane is known as L_i. The climb axis is contained in the gyro reference plane and is perpendicular to and intersects L_i at the origin. The turn axis is contained in the missile velocity slant plane and is perpendicular to and intersects the missile velocity axis at the origin. The climb angle, CA, is measured between the horizontal and missile velocity slant planes. From geometry, the angle between the H_G axis and the missile climb axis is also equal to CA. The missile turn angle, TA, is the angle between L_i and the missile velocity axis measured in the missile velocity slant plane. Again by geometry, the angle between X_G and the missile turn axis is equal to TA.

b. Mathematical analysis. Converting from gyro to missile axes is only slightly more complicated than converting from earth to gyro coordinates. Not only is the missile coordinate system rotated with respect to the gyro coordinate system, but it is also tilted. The angle of rotation is the turn angle, TA, and the angle of tilt is the climb angle, CA. The components of steering error S_{GY} and S_H, which are contained in the gyro reference plane, are illustrated in figure 42. The axes in the gyro reference plane are L_i and the climb axis. S_H and S_{GY} are converted into components S_C and S_i. The process for obtaining S_C and S_i is similar to that for obtaining S_{GX} and S_{GY} in the conversion from earth to gyro coordinates. S_{GY} and S_H are projected onto the missile climb axes and L_i. These projections produce four components: two along the missile climb axis; and two along L_i. Those derived from S_{GY} are S_{GY} cos CA and S_{GY} sin CA; those derived from S_H are S_H cos CA and S_H sin CA. S_{GY} cos CA and S_H sin CA are both positive going along L_i and are summed to obtain S_i. The vectors S_{GY} sin CA and S_H cos CA on the missile climb axis oppose each other and are summed to obtain S_C. Stated in equation:

$$S_i = S_{GY} \cos CA + S_H \sin CA \qquad (45)$$

and

$$S_C = S_H \cos CA - S_{GY} \sin CA. \qquad (46)$$

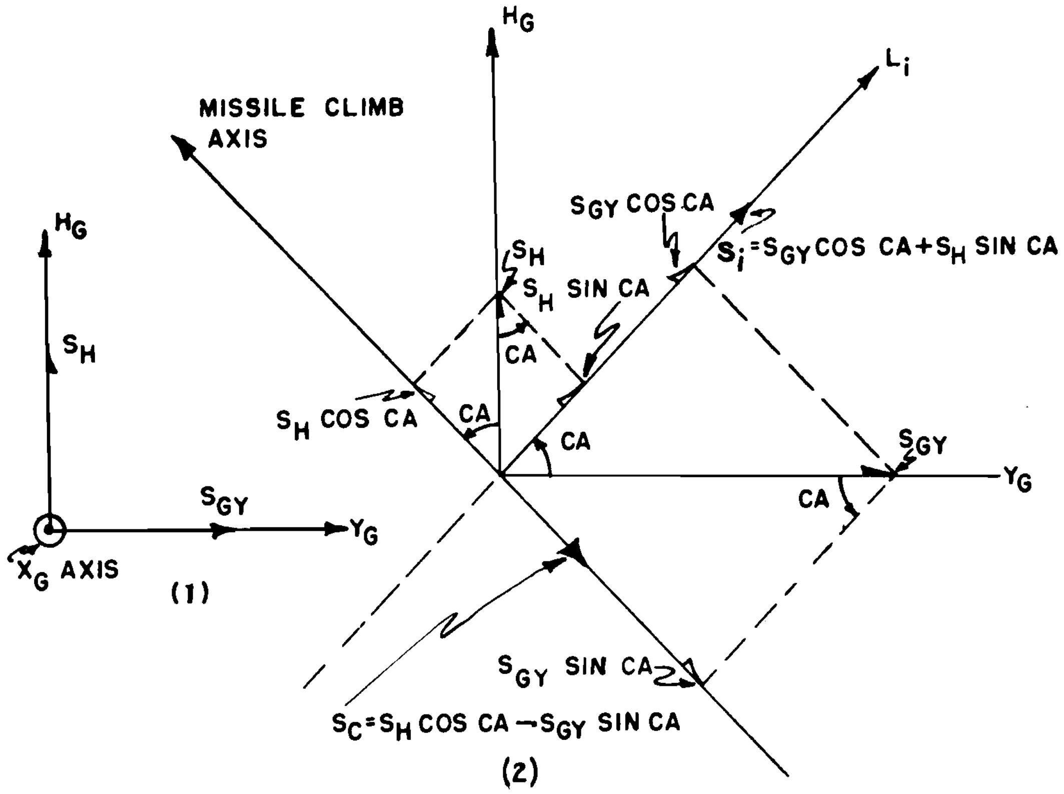

Figure 42. Coordinate system for determining steering errors in gyro reference plane.

c. Steering error components. The components of steering error in the missile velocity slant plane are shown in figure 43. Vectors representing S_{GX} and S_i are shown on the perpendicular axes, X_G and L_i, respectively. In figure 43, the missile velocity axis and missile turn axis have been indicated. The turn angle is observed as the angle between L_i and the missile velocity axis and also between X_G axis and the missile turn axis. As in the preceding conversion, S_i and S_{GX} are projected onto two missile axes, the missile velocity and the missile turn axes. These projections produce four components. Those obtained from S_i are S_i cos TA and S_i sin TA. From S_{GX} are obtained S_{GX} cos TA and S_{GX} sin TA. The resultant vector along the missile turn axis is called S_T while that along the missile velocity axis is known as S_V. The vectors representing S_i sin TA and S_{GX} cos TA are summed to obtain S_T while those representing S_i cos TA and S_{GX} sin TA are summed for S_V. Stated in equations:

$$S_T = S_{GX} \cos TA - S_i \sin TA \quad (47)$$

and

$$S_V = S_{GX} \sin TA + S_i \cos TA \quad (48)$$

It is already known (equation 44), that

$$S_i = S_{GY} \cos CA + S_H \sin CA \tag{49}$$

therefore

$$S_T = S_{GX} \cos TA - (S_{GY} \cos CA + S_H \sin CA) \sin TA \tag{50}$$

or

$$S_T = S_{GX} \cos TA - S_{GY} \cos CA \sin TA - S_H \sin CA \sin TA \tag{51}$$

In like manner

$$S_V = S_{GX} \sin TA + (S_{GY} \cos CA + S_H \sin CA) \cos TA \tag{52}$$

or

$$S_V = S_{GX} \sin TA + S_{GY} \cos CA \cos TA + S_H \sin CA \cos TA. \tag{53}$$

In the example shown in figure 43, note that S_T is a negative component and that S_V is a positive component of steering error.

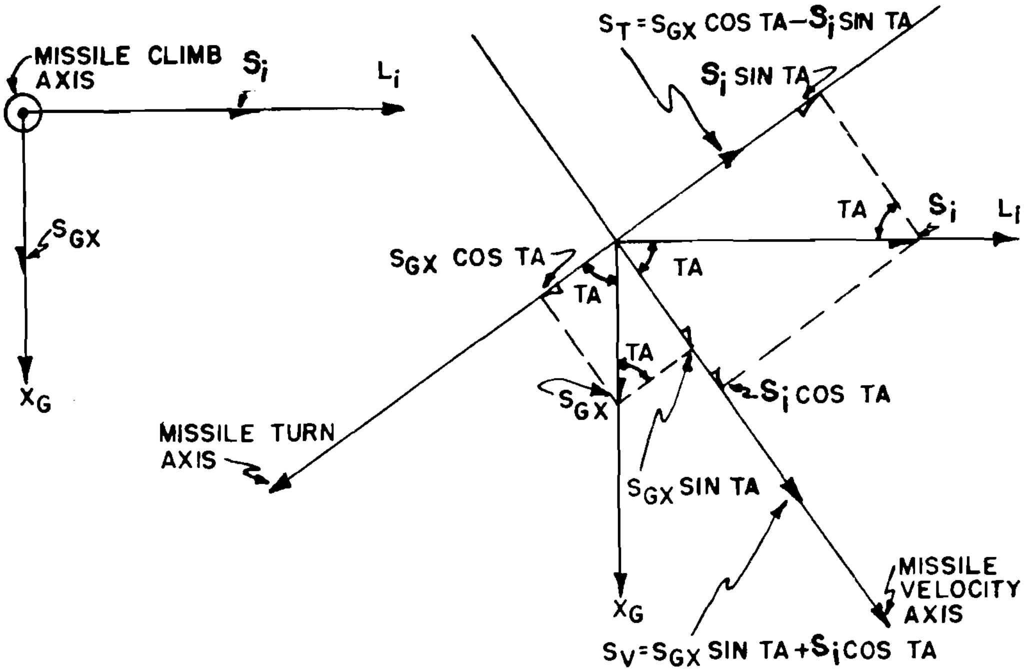

Figure 43. Steering error components in the missile velocity slant plane.

82. FIN ORDER SOLVER

The purpose of the fin order solver is to determine the steering orders to be sent to the missile to cause it to change its direction and thus intercept the target at the proper point. In computing the required orders, the fin order solver calculates the components of the orders required by each fin pair. This is necessary because the P-fin and Y-fin planes are located 45° from the missile velocity slant plane and the gyro reference plane. If a turn to the right or left is desired, the order must be applied in part to both fin pairs. It must be remembered that there is no way of regulating the speed of the missile directly, so it is necessary to apply transverse accelerations to the missile to change its heading. These accelerations cause the velocity components along the climb and turn axes to change. The inputs to the fin order solver are the steering errors along the climb and turn axes, S_C and S_T. The outputs, G_Y and G_P, are voltage analogs of acceleration and as such are sent to the missile-tracking radar where they are combined to cause proper orders to be sent to the missile.

83. MATHEMATICAL ANALYSIS OF FIN ORDER SOLVER

a. Conversion of steering error components. The velocity steering errors, S_T and S_C, are in the missile velocity slant plane and gyro reference plane respectively. Before they can be used to determine the orders to be applied to the missile fins, they must be converted into components of steering errors which are perpendicular to the fin planes (fig 44). Figure 44(1) is a rear view of the missile and shows the missile climb and turn axes and the positions of the P- and Y-fin pairs. Figure 44(2) shows the steering errors S_T and S_C drawn as vectors along the missile turn and climb axes. Figure 44(3) shows the conversion of S_T and S_C to S_{PF} and S_{YF}. This conversion is accomplished in the same manner as the conversions from earth to gyro coordinates and from gyro to missile coordinates. Projections are dropped from S_T and S_C to axes perpendicular to the Y-fin and P-fin planes. S_{YF}, the steering error in the Y-direction, lies along an axis perpendicular to the Y-fins. This orientation is necessary since S_{YF} must lie in the axis which is in the direction in which the Y-fins can turn the nose of the missile. The Y-fins cause the missile to climb to the right or dive to the left. In a like manner, S_{PF}, the steering error in the P-direction, must lie along an axis perpendicular to the P-fins. The P-fins cause the missile to climb to the left or dive to the right. Stated mathematically,

$$S_{YF} = S_C \sin 45° + S_T \cos 45° \quad (54)$$

$$S_{PF} = S_C \cos 45° - S_T \sin 45°. \quad (55)$$

The sine and cosine of 45° are both equal to 0.707. Therefore,

$$S_{YF} = 0.707\ (S_C + S_T) \tag{56}$$

$$S_{PF} = 0.707\ (S_C - S_T). \tag{57}$$

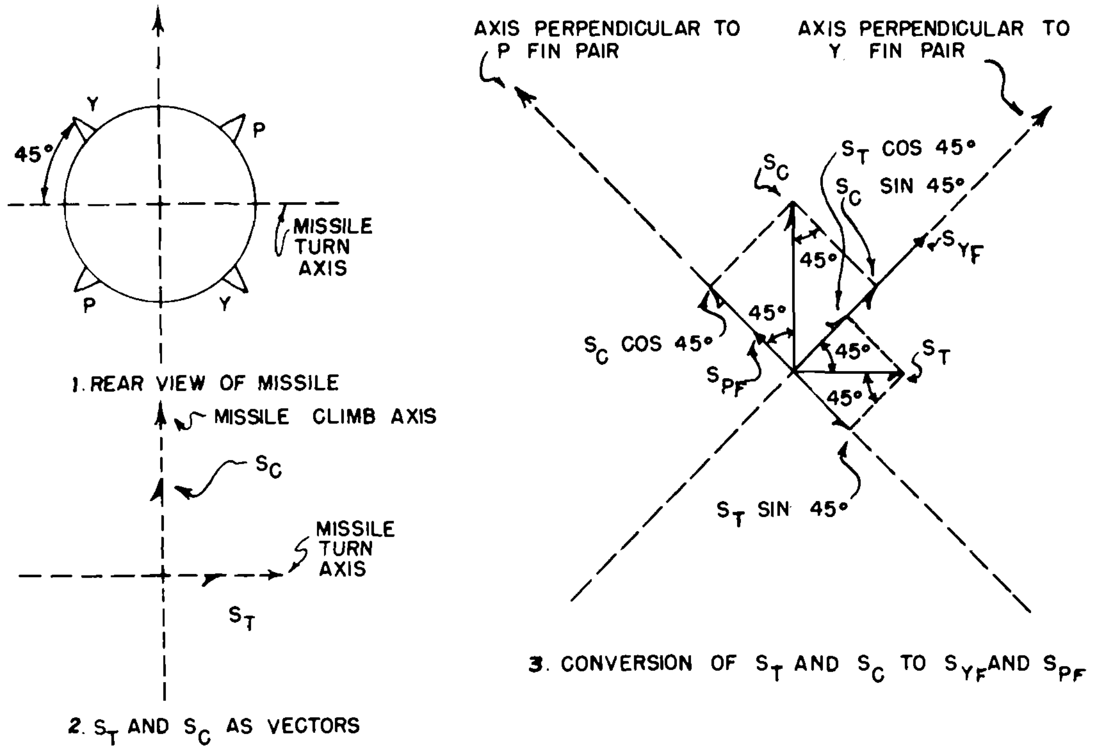

Figure 44. Conversion of velocity steering errors.

b. Steering orders sent to missile. The computer must now determine the steering orders to be sent to the missile using S_{YF} and S_{PF} as known quantities. The steering errors have dimensions of velocity while the steering orders are in terms of acceleration. The steering orders must impart an average velocity to the missile which will just cancel the steering error in the remaining time to intercept. Consider the formula:

$$V = \frac{1}{2}\ at, \tag{58}$$

where a represents constant acceleration, t represents the time during which the object is accelerating, and V is the average velocity over the entire time interval from zero to t seconds. To calculate the acceleration required by the missile, the steering error perpendicular to one of the fin pairs is substituted in the above formula:

$$S_{PF} = \frac{1}{2} \text{ at.} \qquad (59)$$

Solving this equation for a, the acceleration order which must be applied to the missile P-fin pair is:

$$a = \frac{2\ S_{PF}}{t}. \qquad (60)$$

The computer sends orders to the missile in terms of g's. One g is a force which will cause an acceleration of approximately 10.7 yards per second per second. The number of g's acceleration to be applied is therefore:

$$a = \frac{2\ S_{PF}g}{10.7t}. \qquad (61)$$

The orders sent to the Y-fin pair are called G_Y while those sent to the P-fin pair are G_P. The order determined above is the acceleration which must be applied to the missile to cancel the steering error in the remaining time to intercept. It is necessary, however, to apply slightly more than the minimum order required to reduce the steering error to zero before intercept occurs. The strength of the order is increased 50 percent. The equation for G_P then becomes:

$$G_P = \frac{2\ S_{PF}}{10.7t} \times \frac{3}{2} = \frac{3\ S_{PF}}{10.7t}, \qquad (62)$$

but $S_{PF} = 0.707\ (S_C - S_T)$ equation (56), therefore:

$$G_P = \frac{3 \times 0.707\ (S_C - S_T)}{10.7t} = \frac{0.198\ (S_C - S_T)}{t}. \qquad (63)$$

In the same manner

$$G_Y = \frac{2\ S_{YF}}{10.7t} \times \frac{3}{2} = \frac{3\ S_{YF}}{10.7t}, \qquad (64)$$

but

$$S_{YF} = 0.707\ (S_C + S_T) \text{ equation (56).}$$

Therefore,

$$G_Y = \frac{3 \times 0.707\ (S_C + S_T)}{10.7t} = \frac{0.198\ (S_C + S_T)}{t} \qquad (65)$$

By increasing the orders shown above by the constant factor of three-halves, the steering error is canceled in only two-thirds of the remaining time to intercept. This action insures removal of the error at considerable distance from the intercept point.

84. TIME-TO-INTERCEPT SERVO

a. General. The time-to-intercept servo is a servomechanism which continuously solves for remaining time to intercept during the computer steering phase. There are two reference inputs to this servomechanism: actual closing velocity, and position difference, X, Y, and H. The output of the servomechanism (controlled variable) is the position of a shaft which is proportional to the remaining time to intercept. The comparator consists of the steering error solver and the steering error converter.

b. Components. The servomechanism is composed of the following major functional units and components:

Steering error solver.

Steering error converter.

Closing speed solver.

t_s input network.

t-amplifier.

Plus and minus t-amplifiers.

Second-per-second bias network.

Low-power servoamplifier.

Modulator.

Motor tachometer.

Output shaft and gearing, cams, and microswitches.

Potentiometers.

During the prelaunch configuration, the t amplifier, low-power servoamplifier, modulator, motor tachometer, output shaft, gearing, cams, microswitches, and potentiometers are part of the time-of-flight predictor. All functional components of the time-to-intercept servo are physically distributed about the computer amplifier and servo cabinets.

c. Outputs. The outputs of the time-to-intercept servo are of two types. One type is the position of the shaft which is analogous to time. This output is designated by the symbol t. The other type is a voltage and is designated by the symbols +t or -t. The +t voltage scale factor is 1 volt equals 1 second. The position of the output shaft, t, determines the position of the brush arm on each of the 21 potentiometers in the time-to-intercept servo assembly. Some of these potentiometers are coarse potentiometers and others are fine potentiometers. In each case the shaft is so positioned that the voltage tapped off any coarse potentiometer is given by the following expression:

$$E_{tapped} = \frac{t}{t_{max}} \times E_{applied}$$

and, since t_{max} is equal to 100 seconds:

$$E_{tapped} = 0.01 \ E_{applied} \times t.$$

For fine potentiometers t_{max} is equal to 25 seconds:

$$E_{tapped} = 0.04 \times e_{applied} \times t.$$

85. MATHEMATICAL ANALYSIS OF TIME-TO-INTERCEPT SERVO

Time remaining to intercept is determined mathematically by using the well-known relationship: Distance is equal to rate multiplied by time. In the problem solved by the Nike I computer, the distance is the missile-to-target distance (position difference) and the rate is the actual closing velocity. Thus, the equation for time to intercept is:

$$\text{time to intercept} = \frac{\text{position difference}}{\text{actual closing velocity}}.$$

86. ONE-DIMENSIONAL TIME-TO-INTERCEPT PROBLEM

As an aid in understanding the reasoning involved in determining time to intercept, consider the following problem which is a special case:

a. Problem. Determine the remaining time to intercept and the location of the intercept point.

b. Given. Target is at point A, missile is at point B (fig 45). The distance A to B is 20,000 yards. Target velocity is 300 yards per second directed from A to B. Missile velocity is 700 yards per second directed from B to A.

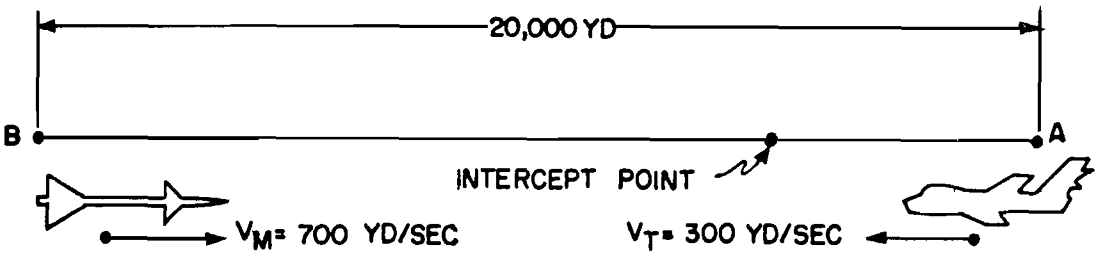

Figure 45. One-dimensional time-to-intercept problem.

c. Solution.

(1) The actual closing velocity is 300 plus 700 = 1,000 yd/sec. The position difference is 20,000 yards, since that is the distance between points A and B. By substitution in the equation:

$$\text{time to intercept} = \frac{\text{position difference}}{\text{actual closing velocity}}$$

$$t = \frac{20,000}{1,000} \text{ : or } t = 20 \text{ seconds.}$$

(2) The target will travel 300 x 20 = 6,000 yards from A to the intercept point. Therefore, the intercept point is 6,000 yards from point A. The missile will travel 700 x 20 or 14,000 yards from point B to the intercept point. Therefore, the intercept point is 14,000 yards from point B.

(3) Assume that the output shaft of the time-to-intercept servo indicates 19 seconds. The target will travel 300 x 19 = 5,700 yards from A, and the missile will travel 700 x 19 = 13,300 yards from point B. The position difference, however, is 20,000 yards; therefore, the target and the missile are still 1,000 yards apart when time becomes zero. Obviously, time remaining to intercept is incorrect. In this special case the missile is flying the correct course, so direction of flight cannot be changed, but time must be changed. In the time servo loop, the incorrect time of 19 seconds divides the position difference of 20,000 yards. The ideal closing velocity is $\frac{20,000}{19}$ = 1,052.6 yd/sec. In the steering error solver the ideal closing velocity, 1,052.6 yd/sec, is compared algebraically with the actual closing velocity, 1,000 yd/sec. The steering velocity error S_V = 52.6 yd/sec. This velocity error causes the time servo to increase time, and as time increases the ideal closing velocity of 1,052.6 yd/sec becomes smaller.

(4) When time is 20 seconds, the ideal closing velocity is equal to the actual closing velocity and $S_V = 0$. The missile and the target will reach the same point in space when the output of the time-to-intercept servo is equal to 20 seconds. If time is less than 20 seconds, they will be separated when time is zero. If time is greater than 20 seconds the missile and target will overshoot and be beyond one another at time zero. The controlled variable is time to intercept. The servomechanism is actuated by an error voltage which continuously attempts to approach a zero value.

(5) The equation for determining time to intercept,

$$t = \frac{\text{position difference along the } M_V \text{ axis (D)}}{-\text{ actual closing velocity along the } M_V \text{ axis (V)}} \qquad (66)$$

is not a form which can be readily solved by the time-to-intercept servo. The error voltage which activates the time-to-intercept servo is S_V. The equation which the servo solves to compute time to intercept correctly is:

$$S_V = 0. \qquad (67)$$

(6) Mathematically, equation (67) is obtained by rewriting equation (66) as follows:

$$t = \frac{D}{-V}, \qquad (68)$$

rearranging terms:

$$-V = \frac{D}{t}, \qquad (69)$$

by further rearrangement:

$$\frac{D}{t} + V = 0. \qquad (70)$$

Since $\frac{D}{t}$ and V are the components of ideal and actual closing velocity along the missile velocity axis, the expression $(\frac{D}{t} + V)$ is equal to S_V. Therefore,

$$S_V = 0. \qquad (71)$$

Whenever S_V is not zero, the solution for time to intercept is not correct and the output (t) of the time-to-intercept servo is changed until S_V does become zero.

87. SIMPLIFIED FUNCTIONAL OPERATION OF TIME-TO-INTERCEPT SERVO

A simplified functional block diagram of the servomechanism which solves for the remaining time to intercept is shown in figure 46. The output quantity (controlled variable) is the remaining time to intercept, t. The two reference inputs are the rectangular earth coordinates of actual closing velocity ($\dot{X}$, $\dot{Y}$, and $\dot{H}$) and the rectangular earth coordinates of position difference (X, Y, and H). The output, t, is fed back mechanically to the closing speed solver (feedback element). The position difference voltages are applied to the closing speed solver. In the closing speed solver the position difference is divided by time so that the outputs of the closing speed solver are the components of ideal closing velocity along the rectangular earth axes ($\frac{X}{t}$, $\frac{Y}{t}$, and $\frac{H}{t}$). These are feedback functions of the controlled variable. Actual closing velocity and ideal closing velocity are applied to the comparator (steering error solver and steering error converter). Within the comparator, the actual and ideal closing velocities are compared. If these two quantities, always opposite in polarity, do not add up to zero, then an error voltage exists. The error voltage is converted within the comparator to an error voltage which represents the steering error along the missile velocity axis. The output of the comparator is S_V, the actuating error, which is applied as an actuating voltage to the controller. S_V is a d-c voltage. Within the controller, it is converted to an a-c driving voltage which is amplified and applied to the servomotor, causing the position of the output shaft to change until the actuating error S_V is reduced to zero. The solution for remaining time to intercept is correct when S_V is zero. When S_V is positive, t increases; when S_V is negative, t decreases.

88. BURST ORDER CIRCUITS

a. General. The final output of the computer is the burst order which is transmitted via the missile-tracking radar to the missile in flight. The purpose of the burst order circuit is to cause the transmission of the burst order to the missile.

b. Components. The components of a burst order circuit are the burst time bias potentiometer, a DC amplifier designated as the C (for comparison) amplifier, a triode called the burst order amplifier, and a number of relays. The burst time bias potentiometer is located behind the right front swinging door in the computer amplifier cabinet.

c. Inputs. The two inputs to the burst order circuit are a d-c voltage tapped off a potentiometer by a brush arm driven by the time-to-intercept servo, and a d-c voltage (burst time bias) tapped off the burst time bias potentiometer. The burst time bias is established by setting a knob, which is a part of the burst time bias potentiometer. This knob is adjusted by the battery commander and may be set to provide values of burst time bias between the limits of zero and 200 milliseconds.

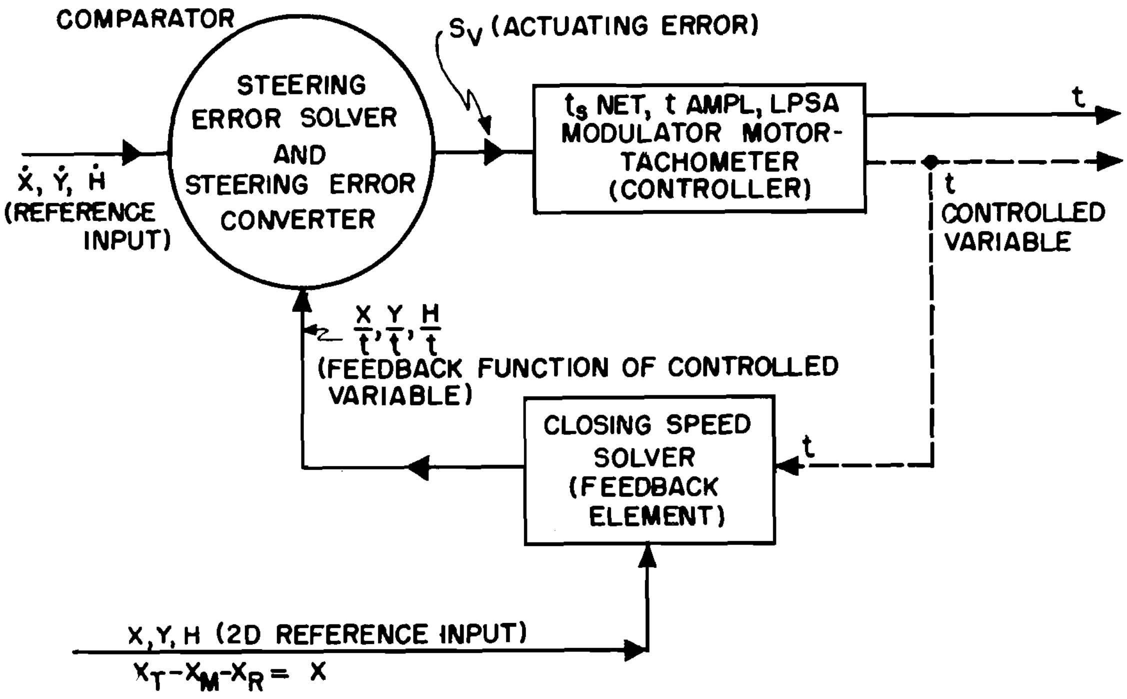

Figure 46. Block diagram, time-to-intercept servomechanism.

d. Output. The output of the burst order circuit is a surge of direct current which causes the transmission of the burst order to the missile via the missile-tracking radar (MTR). Switches on the battery control console permit the battery control officer to delay the burst or to send a command burst signal whenever such action is necessary.

89. SIMPLIFIED FUNCTIONAL OPERATION OF BURST ORDER CIRCUITS

A simplified block diagram of the burst order circuit is shown in figure 47. The burst order circuit produces the burst order when time to intercept is equal to the burst time bias set into the computer by the battery commander. The input voltages to the burst order circuit are a positive voltage that is tapped off a potentiometer in the time-to-intercept servo and a negative voltage tapped off the burst time bias potentiometer. At times to intercept of 0.25 second (250 milliseconds) or greater, the positive voltage input is constant in value and always greater than the burst time bias. The positive and negative input voltages are compared, amplified, and inverted in the C-amplifier. At times to intercept 0.25 second, or greater, the output of the C-amplifier will be approximately -30 volts. The burst order amplifier and the burst indicator amplifier are biased by the output voltage of the C-amplifier, and are always cut off when the output voltage is -30 volts. The burst order relay is in the output circuit of the burst order amplifier and will be energized only when sufficient current flows through the relay coil. Therefore, at times to intercept of 0.25 second, or greater, no burst order can be sent. As time to intercept becomes less than 0.25 second, the positive voltage decreases. When time to intercept has decreased to a value approximately 24 microseconds greater than the burst time bias setting, the limiting action in the C-amplifier circuit ceases and the output voltage changes from -30 volts to a less negative value. When time to intercept is equal to burst time bias, the output of the C-amplifier will be approaching zero. Zero bias on the burst order amplifier and the burst indicator amplifier will allow the amplifiers to pass enough current to energize the burst order relay and the burst indicator relay. This action causes the burst order to be sent to the missile and provides a visual burst indication in the battery control and radar control trailers.

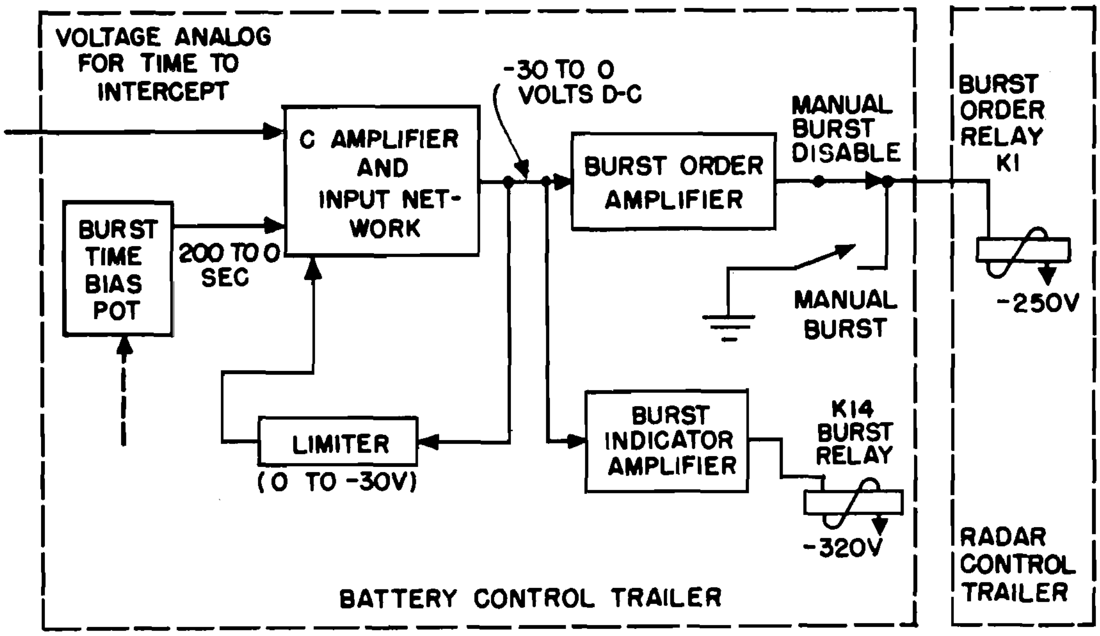

Figure 47. Burst order circuit, simplified block diagram.

90. ADJUSTMENT OF BURST TIME BIAS

The battery commander is responsible for setting the burst time bias to obtain the most effective burst. He may direct any subordinate to set the burst time bias, and he may call upon his fire control specialists for advice as to the proper setting for any particular engagement. The chain of events which results in missile burst is started a few milliseconds before the missile reaches the desired intercept point because of several time delays. The most important of these delays for an incoming course are:

a. A delay of 5 microseconds in information delivered to the computer by the missile-tracking radar.

b. A delay of 2 microseconds in information passing through the computer geometric resolution circuits.

c. A delay of 25 microseconds in the computer issuance of the burst order.

d. A delay of 62 microseconds in the missile circuits. This time is variable for each missile, but will be reported from the launcher area.

e. A delay of 1.5 microseconds in a warhead missile from detonator signal to full explosive intensity. This delay is 5 microseconds for a spotting charge missile to build up a light intensity great enough to show on photographs.

f. A delay of 10 microseconds, for a warhead round only, in order to put the lethal center of the exploded warhead at time zero. At a closing velocity of 1,400 yards per second, the error in burst timing is approximately 3 microseconds or 5 yards. Total delay for a warhead missile, therefore, is approximately 105.5 microseconds (varies according to missile delay) and the total delay for missile carrying a spotting charge is approximately 99 microseconds plus or minus the delay variation of the missile circuit.

91. ORDER LIMITING

The order-limiting circuit is required because the maximum acceleration order that may be transmitted to the missile must be reduced as the height of the missile above sea level increases. At moderate altitudes above sea level, the missile can execute orders up to 5g acceleration per fin-pair. As the altitude increases and the density of the air decreases, larger and larger fin deflections are required to produce the same turning moment on the missile. As a result, the missile develops instabilities when it is ordered to execute large (5g) accelerations at high altitudes. Therefore, it is necessary to make sure that, as the missile gains altitude above sea level, the limiting of the fin orders is decreased below the 5g limit established for fin accelerations at moderate altitudes. The decrease in the limiting value as altitude above sea level increases is accomplished by the order-limiting circuit.

92. THE ORDER-LIMITING CIRCUIT

The order-limiting circuit consists of the positive and negative order-limiting amplifier circuits, the order-limiting limiter, the G_Y limiter, the G_P limiter, and the order-limit threshold control.

93. FUNCTIONAL OPERATION OF THE ORDER-LIMITING CIRCUIT

The fin orders result from voltages produced as outputs by the G_Y and G_P amplifiers. The output voltages are limited by the action of the G_Y and G_P limiters, which are of the conventional type. However, these limiter circuits differ from the other diode limiters used in the computer, in that they are biased by a variable voltage instead of a fixed voltage. The bias voltage is called the order-limit voltage, and is supplied to the G_Y and G_P limiters as positive and negative order limiting. The limiter circuit is arranged so that it will limit the output voltages of the G_Y and G_P amplifiers when the positive or negative magnitude of the output voltages exceeds the magnitude of the order-limit voltages. The order-limit voltages are supplied by a circuit that accepts missile height data from the missile coordinate converter, and develops a positive and a negative voltage that vary with missile altitude.

94. ORDER-LIMITING REQUIREMENTS

At this writing, the requirements of the order-limiting circuit are:

a. To limit the G_Y and G_P outputs to a maximum of 5g (100 volts) when missile altitude above sea level is 30,000 feet or less.

b. To reduce the limiting value by 1g (20 volts) for each 8,000-foot increase in altitude between 30,000 and 50,000 feet above sea level. (For example, if the missile is 38,000 feet above sea level, the outputs of the G_Y and G_P limiters will be limited when the output exceeds 4g.)

c. To reduce the limiting value by 1g (20 volts) for each 13,300-foot increase in altitude above 50,000 feet above sea level. (For example, if the missile is 50,000 feet above sea level, the G_Y and G_P orders will be limited when the orders exceed 2.5g; then if the missile goes from 50,000 feet to 63,300 feet, the 2.5g limiting value will be reduced to 1.5g.) Since work is in progress to reduce or eliminate, if possible, the missile instabilities resulting from large acceleration orders at high altitudes, the circuit used to generate the order-limit function has been designed to be as flexible as possible. The altitude above sea level at which the limit begins to be reduced (30,000 feet at the present time) is established by the position of the OL

THRESHOLD potentiometer. It is geared to a dial which indicates the order-limit threshold in feet. The threshold is the height of the missile above the radar site at which the order limit starts reducing the maximum fin orders. This potentiometer also provides a method of introducing the height above sea level of the radar site. The missile-height data obtained from the missile-coordinate converter is height of the missile above the missile-tracking radar. The computer does not know the height of the missile above sea level. The difference between radar altitude and the absolute altitude of missile can be corrected by means of the OL THRESHOLD potentiometer. For example, suppose that the reduction of order limit from ±5g should begin at 30,000 feet above sea level, and that the radar site is 4,000 feet above sea level. In this case the OL THRESHOLD potentiometer should be set at 26,000 feet. The range of the OL THRESHOLD potentiometer extends from 0 to 60,000 feet. This range is large enough so that, in case the missile instabilities are eliminated entirely, it will be possible to set the threshold to a value that is higher than any altitude that the missile might reach in actual flight.

95. ORDER SHAPING

The S_T and S_C signals applied to the fin order solver are error voltages developed in the over-all missile control loop. The function of this loop is to position the missile in flight so that its path will intersect the flight path of the target at a point at which both the target and the missile arrive at the same time. Basically, the control loop operates on the proportional mode of control. This means that the acceleration orders for the missile are made proportional to the error voltages. The response of a proportional controller can be improved by making it error-rate sensitive. Instead of controlling the output member (in this case the missile) in accordance with only the direction and magnitude of the error, the controller takes into account also the rate at which the error is changing. By observing the rate of change of the error, the controller can anticipate future acceleration requirements and make the response more accurate. The technique of adjusting the acceleration of the output member of a control, in accordance with the error rate, is called "lead equalization" or "derivative control." The term "lead equalization" comes from the fact that by using this adjustment, a given output of the controller will occur sooner than it will with strict proportional control. The term "derivative control" arises from the fact that error-rates are determined mathematically by taking the derivative of the output with respect to time.

96. EXAMPLE OF STRICT PROPORTIONAL CONTROL

The rudder of a ship if steered according to the strict proportional mode of control, is deflected in proportion to the heading error existing at the time.

Suppose that the ship is proceeding along the correct course with the rudder undeflected when, due to a current, it starts turning to starboard. As the heading error increases, the rudder turns slowly to port. The starboard drift of the ship is first slowed and then brought to a halt when there is already a considerable heading error. By this time, the rudder has been turned sharply to port and will cause the ship to start turning to port toward the correct heading. The ship will gain angular momentum in the process. As it approaches the correct heading, the rudder turns back toward the undeflected position, but the angular momentum built up during the heading correction will cause the ship to continue to turn to port, and the correct heading is overshot. Then the process repeats in the reverse direction: As the rudder turns slowly to starboard, the ship will overshoot the correct heading several times before the correct heading is reestablished. Obviously, this is not the best method for steering the ship.

97. EXAMPLE OF PROPORTIONAL PLUS DERIVATIVE CONTROL

The steering in the example above can be improved by making it error-rate sensitive. Again the ship proceeds along the correct course with undeflected rudder when the current causes it to turn starboard. This time, an increase in the rate of heading error is noted immediately, even before time has passed for the heading error to accrue to an appreciable value. Responding to the error rate, the rudder deflects to port sooner than it would if strict proportional control were used. The starboard momentum of the ship is thus overcome, and the starboard rotation is quickly stopped. As the starboard rotation of the ship becomes very slow, the error rate becomes small and the additional rudder deflection dictated by derivative control is negligible when compared to the rudder deflection called for by strict proportional control. The rudder is then kept deflected to port in proportion to the accrued starboard heading error, but the more prompt movement of the rudder has kept the total change in the starboard direction to a much smaller value. Proportional control now causes the ship to turn to port. As the ship builds up angular momentum in the port direction, the increasing rate of decrease of the error is noticed by the error-rate sensitive device. This causes the rudder to turn back to zero deflection faster. As the ship approaches the correct position, the rudder is deflected very little by proportional control since the heading error is now small. However, the error-rate sensitive device now detects a decreasing rate of decrease in error and deflects the rudder toward starboard before it reaches the correct heading. The port rotation of the ship can thus be brought to a halt with little, if any, overshoot. The rudder has turned first to port and then to starboard as in the strict proportional control illustrated, but the rudder moved sooner than in the purely proportional case. The lead in rudder response considerably reduced the first buildup of the error and the amount of overshoot.

98. CONFLICTING REQUIREMENTS IN ORDER SHAPING

The preceding discussion illustrates that lead equalization has the advantage of more rapid and more accurate response to changing control requirements. In the example of the ship, the sudden change in the control requirement was due to a sudden current. In the case of the missile control, changes in the control requirement will usually be caused by maneuvers of the target. Any acceleration of the target, such as a turn or a dive, makes it necessary to change the course of the missile. The change in the intercept course will be reflected in a more or less sudden increase in the steering error components. Lead equalization makes the missile control loop response very sensitive to such rapid changes in the error voltages. Sensitivity is desirable from the point of view of pursuit, but it has a serious disadvantage. Because the radar tracking of the target and missile is not perfectly smooth, spurious fluctuations will occur in the error signals applied to the fin order solver. These result in the sending of spurious orders to the missile. The noise orders will cause the fins of the missile to flap erratically. Since the flapping motion of the fins will center about the desired fin position, it will usually be averaged out, due to the inertia of the missile and the course will not be affected. However, the jerky motions of the control fins greatly increase the air resistance and slow the missile down. For this reason, it is desirable to make the fin order solver as insensitive to noise as possible. It thus appears that rapidity of response to target accelerations and insensitivity to noise are two conflicting design requirements.

99. COMPROMISE SOLUTION IN ORDER SHAPING

A compromise is achieved by inserting the lead equalization in steps. In the early stages of the pursuit, when the missile is still far from the target, ample time is available for adjusting the course of the missile. Prompt response to target acceleration is unessential and lead equalization is not required. As the time remaining before intercept decreases, speed of response becomes more and more important, and lead equalization is switched in when the time to intercept (t) equals 24 seconds provided that RADAR CLEARED and ON TRAJECTORY have occurred. (It is not desirable to use lead equalization when the initial-turn section has control of the missile.) In the final phase of the pursuit, rapidity of missile response is of utmost importance. So, for the last 10 seconds before intercept, the lead equalization is doubled with respect to the amount put in at 24 seconds time to intercept. Loss of missile speed due to fin flapping is the lesser of two evils during the final phase.

CHAPTER 3

COMPUTER OPERATION

Section I. OVER-ALL OPERATION OF THE COMPUTER

100. GENERAL

a. Operation. The operation of the Nike I computer during an engagement is separated into three phases: the prelaunch phase, the initial-turn phase, and the steering phase. During the prelaunch phase, the computer solves tactical problems prior to launching the missile. The initial-turn phase starts 7 seconds after the FIRE button has been pressed; by this time the missile has gone through the boost period and has been roll-stabilized. The steering phase starts a few seconds after, when missile has been directed on a ½ g trajectory toward the predicted intercept point. During the initial-turn phase and the steering phase, the computer transmits steering orders to the missile via the beam of the missile-tracking radar, directing the missile to intercept the target. During all three phases the computer supplies data continuously to horizontal and altitude automatic plotting boards, and to an event recorder. The plotting boards display tactical information that makes it possible for personnel of the battery control console to observe the progress of the engagement at a glance. The event recorder is a photographic oscillograph. It provides a permanent record of the battery activity and equipment operation during engagements. In addition to the data transmitted to the automatic plotting boards and the event recorder, the computer displays the instantaneous values of target, missile, or closing velocities in rectangular coordinates (X, Y, and H) at meters on the computer control panel (fig II-35) and the magnitude of fin orders transmitted to the missile-tracking radar on meters at the tactical control console (fig II-28). The computer provides a dial indication at the tactical control console of the gyro azimuth preset angle, and lamps to indicate when the missile is near gimbal lock, and whether the computer is in a test or action condition.

b. Configurations. The computer is subdivided on a functional basis into three configurations: the prelaunch configuration, the initial-turn configuration, and the steering configuration. Each configuration functions according to an ordered cycle of operation automatically controlled by relays, switches, and timers dispersed throughout the computer.

101. PRELAUNCH SECTION CONTROL

a. Sequence of events. As far as the computer is concerned, the engagement starts when the TARGET TRACKED signal is given. Prior to this, the target has been detected by the acquisition radar system, identified as a foe, and designated by the battery control officer as the next target to be engaged. Information defining the approximate position of the target has been transferred from the acquisition radar to the target-tracking radar, and used there to slew the target-tracking radar to the azimuth and range of the designated target. After the target-tracking radar has acquired the designated target and is tracking in the desired mode, the target radar azimuth operator at the target-tracking console presses the TRACKED pushbutton. This initiates the sequence of events in the computer. A graphical representation of this sequence of events, starting with the TARGET TRACKED signal, is shown in figure II-94

b. Target tracked. When the computer receives the TARGET TRACKED signal, it enters the prelaunch phase, and the prelaunch section of the computer (fig II-89 of 1y) is switched into operation. The computer accepts target position coordinates from the target-tracking radar and differentiates them in order to derive target velocity. The differentiating networks have a settling characteristic of 4 seconds; therefore, 4 seconds is allowed for the computer to settle and begin supplying reliable data.

c. Section operation. The prelaunch section of the computer receives the present position of the target as well as information regarding the position of the missile designated to be fired. On the basis of this information, and taking into account the ballistic characteristics of the missile, the prelaunch section predicts the position of the intercept point and the time of flight of the missile on the assumption of an immediate launching. The prelaunch section also determines the azimuth angle of the predicted intercept point measured at the launcher site. This angle is called the gyro azimuth (A_G) and is used to preset the roll-amount gyro in the missile. The roll-amount gyro is positioned in azimuth so that the plane of rotation of the gyro wheel (the gyro plane) will contain the predicted intercept point.

d. Information distribution. The prelaunch section of the computer sends information to the horizontal plotting board, the altitude plotting board, and the event recorder. The horizontal plotting board receives the X and Y coordinates of the target present position and of the predicted intercept point and displays azimuth and ground range to the target and to the predicted intercept point. The altitude plotting board receives the altitude of the predicted intercept point and the predicted time of flight and plots one against the other. The event recorder receives the X-, Y-, and H-component velocities of the target, which are recorded together with other significant data on battery activity before launch.

e. Plotting board information. By observing the altitude plotting board, the battery control officer can determine when the intercept point first comes within range. When this occurs, the pen plotting intercept altitude starts moving from its extreme right position. Once the intercept point is within range, the horizontal plotting board is observed to check that the predicted intercept point is not over a restricted area. After that, the battery control officer can fire at his discretion by pressing the FIRE switch.

f. Actions initiated by FIRE switch. The pressing of the FIRE switch initiates three actions. First, a brake is applied to the gyro-azimuth servo in the computer, fixing the azimuth of the intercept point predicted at the instant at which the FIRE key is pressed as a *reference azimuth*. Second, a clutch in the dead-time unit in the computer is engaged, permitting the unit to operate. This unit operates as a clock which indicates the remaining dead time. The dead time is the 7-second interval between the pressing of the FIRE switch and missile roll stabilization. After the FIRE switch has been pressed, the remaining dead time is reduced to zero at the rate of 1 second per second. Third, a 1.75-second timer is started at the launching area. This timer plus the 0.25-second squib delay provides a total delay of 2 seconds between the pressing of the FIRE switch and the actual launching of the missile. The prelaunch section of the computer continues to predict intercept point location and time of missile flight during the 7 seconds dead time following the FIRE signal. The only difference is that the gyro azimuth solution is frozen and that the dead time, which was constant before, is now running down.

g. Gyro settling. The 2-second time interval after the FIRE signal allows the presetting servo for the roll-amount gyro in the missile to catch up with the latest A_G data and settle at precisely the azimuth at which the gyro azimuth servo in the computer has been frozen. At the end of this 2-second interval, the roll-amount gyro is disconnected from the presetting servo and uncaged; the booster squib is ignited and the missile lifts off.

h. Missile away. The missile ascends along an almost vertical path, which is tilted slightly toward the booster disposal area. The missile-tracking radar follows the missile in its upward flight, causing the voltage representing H_M (missile altitude) in the computer to increase. This increase is detected by the missile-away circuit, which operates and triggers a 4.5-second timer called the missile-away + 4-second delay timer. There is an interval of approximately 1 second between the actual missile takeoff and the operation of the missile-away circuit.

i. Actions initiated by the missile-away circuit. In addition to starting the missile-away + 4-second delay timer, missile-away signal initiates three other events. First, the missile differentiators providing missile velocity data are enabled. Second, the pen on the horizontal plotting board, which hitherto has plotted the position of the predicted intercept point, is switched so it will now plot the position of the missile. The other pen of the horizontal plotting board continues to plot the target position. Third, the right pen of the altitude plotting board, which has been plotting the altitudes of the predicted intercept point versus time of missile flight, is switched to plot the altitude of the target versus time to intercept. The left pen of the altitude plotting board, which has not been used to this point, starts to plot missile altitude versus time to intercept.

102. INITIAL TURN SECTION BECOMES OPERATIVE

a. General. At missile away, the initial-turn section of the computer becomes operative and starts solving for the critical-turn angle and the difference-turn angle. This operation of the initial-turn section is preparatory as the output of the initial-turn section will not be used until 4.5 seconds later, when roll stabilization is completed. It is necessary, however, to start the computations in advance, so that the initial-turn section will have reliable solutions ready when it takes over the function of steering the missile. During the 4.5 seconds timed by the missile-away + 4-second delay timer, both the prelaunch section and the initial-turn section of the computer are operative. Portions of the steering section of the computer are also operative during this time. The climb-angle servo and the turn-angle servo are solving for the missile flight angles. Other components of the steering section are solving for the steering error vector.

b. Roll stabilization. During the period timed by the missile-away + 4-second delay timer, approximately 3 seconds after launch, the booster separates from the missile. After that, a hydraulic servo in the missile will control the ailerons on the main fins so that the missile rolls into the belly down position and thus has the proper sense of up and down and right and left. This roll stabilization will be completed before the 4.5 seconds are up. This marks the end of the 7-second period of dead time. The missile is now ready to be steered by the computer. When the missile-away + 4-second delay timer clocks down, the prelaunch section of the computer ceases to operate and the initial-turn section of the computer, operating in conjunction with portions of the steering section, assumes control of the missile.

103. INITIAL-TURN AND STEERING SECTIONS CONTROL

a. General. At the end of roll-stabilization time, the missile is still in a steep climb. Because of the initial-tilt angle of the missile, directed toward the booster disposal area, and possible additional tilting of the missile during the boosting period, the flight angles of the missile at this time are rather uncertain. The climb angle is near 90° but could deviate as much as 15°. The turn angle is near zero but it too could deviate as much as 15°. To get the missile on the intercept course, it is now necessary to quickly reduce the climb angle to that called for by the ½g trajectory and to reduce the turn angle to zero. During this maneuver, it is necessary to insure that the missile does not get too close to the missile-tracking radar, for the azimuth tracking rate is inversely proportional to ground range. It thus could exceed the maximum angular tracking rate of the radar if the ground range becomes too small. This initial computer-controlled maneuver of the missile is controlled by the initial-turn section of the computer.

b. Initial turn operation. The 7g dive order circuit orders the missile to dive as quickly as possible, that is, with an acceleration of 7g's, to reduce the climb angle to the correct value for the ½ g trajectory toward the predicted intercept point. At the same time the initial-turn section issues turn orders that cause the missile to fly with zero turn angle, i.e., in the gyro plane. Moreover, the initial-turn section checks the position of the missile against position of the missile-tracking radar and insures that the missile does not get too close to the radar. If the direct course toward the predicted intercept point would lead the missile through the critical zone around the missile-tracking radar, the initial-turn section would steer the missile around this zone. When the missile has passed the missile-tracking radar, so there is no longer any danger of the radar losing the missile due to excessive angular rates, this fact is detected by the radar cleared circuit of the initial-turn section, and this circuit puts out a radar cleared signal. When the missile has dived to the point where the steering error component in the climb direction has been reduced to zero, this fact is detected by the on-trajectory circuit, and this circuit issuing an on-trajectory signal. Also when t equals 10 seconds, the on-trajectory signal is given. The 7g dive order is then removed, and further steering orders in the vertical trajectory plane (i.e., climb or dive orders) are issued by the steering section of the computer. The steering section then guides the missile in the vertical plane so that it follows a ½g trajectory to the intercept point. The steering section assumes control of the maneuvers of the missile in the turn direction, i.e., in the slant plane, only after both the on-trajectory and radar cleared signals have been received. Depending upon the direction of the intercept point, the emplacement of the missile-tracking radar, and other factors, the radar cleared condition may occur either before or after the on-trajectory condition. If the on-trajectory condition occurs first, there is a brief time interval, i.e., the time interval between on-trajectory and radar cleared when the computer flight is controlled jointly by the initial-turn section and the steering section. The initial-turn section controls the missile flight in the turn direction, while the steering section controls the missile flight in the climb-dive direction.

104. STEERING SECTION CONTROL

a. Steering operation. When the on-trajectory and radar cleared signals have both been received, the steering section of the computer has complete control of the missile flight path. From target and missile velocities the computer determines how fast the target and missile are coming together. This is called the actual closing velocity. From target present position, missile present position, and the time-to-intercept solution, the computer determines how fast the target and missile should be coming together, that is, by dividing the distance between the two by the time solution, it determines the required closing velocity. This is called the ideal closing velocity. The computer then compares the actual velocity with the ideal velocity and, from this comparison, determines if a steering error exists. From the rectangular components of this steering error, the computer determines what orders must be sent to the missile to equate the actual to the ideal and at the same time recomputes the time solution to equate the ideal to the actual.

b. Order shaping. In the steering section of the computer, the steering orders sent to the missile fins are modified by order shaping. Order shaping uses a technique called lead equalization to make the missile respond more rapidly to steering errors. Here, lead equalization is mentioned only for the sake of completeness in the sequence of operation. When the time to intercept is large and there is ample time left for correcting steering errors, it is better for the missile to respond slowly since its course will not be affected by sporadic signals. Accordingly, no lead equalization is used as long as the time to intercept is greater than 24 seconds. At a time to intercept of 24 seconds, lead equalization is switched into the fin order solver. At a time to intercept of 10 seconds, when errors must be corrected very quickly, the lead equalization switched in at 24 seconds is doubled.

c. Time card switching. The time-to-intercept servo contains two sets of time-to-intercept potentiometers. The coarse time potentiometers cover the full time range from zero to 100 seconds time to intercept. The fine time potentiometers cover the partial time range from zero to 25 seconds time to intercept with greater accuracy. At a time to intercept of 24 seconds, several time-to-intercept outputs are switched from coarse potentiometers to fine potentiometers.

105. BURST CIRCUIT ENABLED

When the time to intercept has run down to 250 milliseconds, the burst circuit is enabled. Simultaneously, the time correction voltage S_V is disconnected from the input network of the time-to-intercept servo. Instead, it is supplied with a constant error voltage that makes it run down the remaining 0.25 second at a second-per-second rate, i.e., like an ordinary timing circuit. Moreover, at the same time, the closing speed solver, which has been computing the ideal closing velocity, stops dividing by t. Thus, from this point on, the outputs of the closing speed solver are no longer proportional to the closing velocity components, but are proportional to the target-to-missile separations, X, Y, and H. This is done to avoid overloading the amplifiers, which would result from dividing by a quantity which is approaching zero. The ideal closing velocity is no longer required, since the computer can no longer appreciably change the course of the missile, but the miss distances are of interest now as they are required for the apparent miss-distance circuits.

106. BURST ORDER ISSUED

a. Burst order. When time to intercept becomes equal to a certain preset time value, called the burst time bias (BTB), the burst circuit issues the burst order. Depending upon a number of factors, the burst time bias is selected at approximately 105.5 milliseconds. The burst order is transmitted via the missile-tracking radar to the missile. The missile then explodes when the time to intercept is zero, or a few milliseconds before that. Additional burst time bias may be set in if it is desired to burst the missile at some small distance from the target. The advantage of exploding the missile at some small distance from the target is that the fragments are given time to disperse before they engulf the target.

b. Apparent miss distance. At the instant the time-to-intercept servo goes through zero time, holding circuits are operated on the three outputs of the closing speed solver. These outputs are now proportional to the miss distances, X, Y, and H. The holding circuits store the values of X, Y, and H at $t = 0$ for a short period of time and apply these values to the event recorder. This circuit is called the apparent miss-distance circuit. If the apparent miss distances thus indicated are small, there is no guarantee that a hit has been scored, but merely that the computer has solved the problem as accurately as is possible with the information supplied to it. However, if the apparent miss distances are large, this may serve as an indication that something has gone wrong; either the missile has failed to respond properly to the guidance commands, or it has been out-maneuvered by the target.

Section II. SEQUENCE OF EVENTS OF COMPUTER

107. INTRODUCTION

A complete summary of the computer switching includes a listing of the signal, its origin, and the effects in the computer (fig II-98). This switching is a resume of the computer control circuits and falls into two categories: computer conditioning and sequential operation.

108. COMPUTER CONDITIONING

This switching is under control of the COMPUTER CONDITION switch on the computer control panel. The positions of this switch, with the corresponding computer configurations, are as follows:

Position	Configuration
STANDBY	Prelaunch section configuration. The A_G servo is positioned in the direction of expected attack by synthetic voltages from GYRO AZIMUTH switch on computer control panel. The time-of-flight predictor positions the t-servo to the 44.57-second standby position. The missile and target differentiators are disabled, with $\dot{H}_M$ having an end-of-boost velocity input. The +S-voltage supply is disconnected from the missile and target range data potentiometers. The 120-volt, 400-cycle voltage is applied to all computer circuits.
ACTION	When the COMPUTER CONDITION switch is first thrown to ACTION, computer conditions are essentially the same as in STANDBY, that is: The A_G servo is positioned in the direction of expected attack by the GYRO AZIMUTH switch and the time-of-flight predictor positions the t-servo to the standby valve. All differentiators are disabled, with $\dot{H}_M$ having a synthetic end-of-boost vertical velocity. The climb-angle and turn-angle servos are set by the synthetic end-of-boost vertical velocity to 1,600 mils and zero mils respectively. The computer is in the prelaunch configuration. The +S-voltage supply is connected to the missile and target range data potentiometers. Circuits that will subsequently do sequential switching are enabled. The second-per-second bias network for the t servo is enabled.

Position	Configuration
ACTION (cont)	Launcher parallax potentiometers are connected to the input of the intercept-point solver. Target position data is disconnected from the intercept-point solver. Steering order outputs of the computer are disconnected from the missile radar. The burst order circuit is disconnected. The $-H_M$ amplifier is conditioned for the missile-away circuit. The 120-volt, 400-cycle voltage is applied to all servos, plotting board circuits, and computer circuits.

109. SEQUENTIAL SWITCHING

The following switching takes place in sequence, when the computer is in the ACTION condition, under control of the signals listed:

Signal	Origin	Effects in Computer
Target tracked	TRACKED button, target-tracking radar console	1. Target differentiators are enabled. 2. The 4-second electronic settling timer is started. 3. The dead-time unit motor is started. 4. The synthetic voltage input from GYRO AZIMUTH voltage divider is disconnected; target position data is connected to intercept point solver. 5. A_G, B, and t servos now start prelaunch computations. 6. The horizontal board pen plots target position, if plotting board condition switch is at PLOT.
Computer settled	Relay in computer which operates when 4-second settling timer has timed out, and t and A_G servos have ceased slewing	1. The horizontal board pen plots intercept point position, if plotting board condition switch is at PLOT. 2. The altitude board pen plots intercept point position, if plotting board condition switch is at PLOT.

Signal	Origin	Effects in Computer
Missile tracked	AGC monitor or TRACKED button, missile-tracking console	1. The intercept-point solver amplifier inputs are switched from launcher-to-target radar parallax to missile position data. 2. The missile tracked signal is sent to event recorder.
Ready-to-fire	Relay in battery control console, which operates when missile is tracked, computer is settled, and target identified as foe	1. The ready-to-fire lamp is illuminated at the battery control console.
Fire	FIRE switch, battery control console	1. The dead-time unit clutch is engaged and the dead-time potentiometer shaft starts. 2. The A_G servomotor is deenergized and the brake is applied. 3. The fire signal is sent to the event recorder and to the plotting boards. 4. The 1.75-second electronic timer is started in the launcher control trailer.
Fire + 2 seconds (lift-off)	Timer in launcher control trailer (1.75-second)	1. Servos in the missile are allowed to catch up to last data from A_G servo in computer. 2. Launch order is given and missile booster squib is ignited. Shortly thereafter, missile lift-off occurs.
Missile away	Missile-away detector in computer ($-H_M$ amplifier)	1. Missile differentiators are enabled. 2. The missile away + 4-second delay timer is started.

Signal	Origin	Effects in Computer
Missile away (cont)	Missile-away detector in computer ($-H_M$ amplifier)	3. The LAUNCH indicator light is illuminated on battery control console. 4. The altitude and horizontal plotting boards are switched to plot the missile and target positions. 5. Final inputs are applied to the initial-turn section. 6. The missile-away signal is sent to the event recorder.
Fire + 7 seconds	Cam-operated switch in dead-time unit	1. The dead-time unit resets.
Missile away + 4 seconds (roll stabilization)	Relay in computer which operates when the missile away + 4-second delay time has timed out	1. The prelaunch section, computation for time-of flight, is removed from t servo. 2. The second-per-second bias network is connected to the t servo. 3. The fin order solver is connected through to missile-tracking radar. 4. The 7g dive order is transmitted to the missile. 5. Missile position data is removed and launcher-to-target radar parallax connected to intercept point solver. 6. Initial turn section amplifiers are switched to HOLD. After a short delay, initial-turn section orders are transmitted to the missile.
Radar cleared	Relay in computer which operates when Y_G first goes positive	1. If a skirting turn was required, it is removed and a zero turn order is substituted.

Signal	Origin	Effects in Computer
On-trajectory	Relay in computer which operates when climb error S_C first becomes positive	1. The 7g dive is removed and the steering error along climb axis, S_C, is applied. 2. The 1/6gt bias is applied to steering error computation. The missile now flies ½ g lift trajectory. 3. The steering error along missile velocity vector, S_V, is connected to the t servo.
Reception of both radar-cleared and on-trajectory signals	Relay in computer that deenergizes	1. The initial-turn section is disabled. 2. All steering orders from the steering section are applied.
t = 24 seconds	Cam-operated microswitch in t servo. Operates at t = 24 seconds	1. The t servo is switched from coarse to fine potentiometer cards. 2. Half order shaping is applied, giving 1-second lead equalization.
t = 10 seconds	Cam-operated microswitch in t servo. Operates at t = 10 seconds	1. Full order shaping is applied, giving 2-second lead equalization.
Burst enable	Cam-operated microswitch in t servo. Operates at t = 250 milli-seconds	1. The burst order circuit is enabled. 2. S_V is disconnected from the t servo which then runs down at a second-per-second rate. 3. Three channels in the event recorder which plot target velocity components are switched to plot apparent miss distance at burst.

Signal	Origin	Effects in Computer
Burst	Relay in comparator amplifier output operates when t voltage = burst time bias	1. The burst order is sent to the missile radar. 2. The BURST light is illuminated on the battery control console, missile control console, and radar control console. 3. The burst signal is sent to the event recorder.
Apparent miss distance at burst holding	Cam-operated microswitch in t servo operates at t = 0 seconds	1. The apparent miss distance at burst circuit is switched to HOLD. 2. Apparent miss distance at burst and burst order circuit lockup when t-servo starts slewing.
Computer reset	Loss of missile beacon signal	1. The computer is returned to prelaunch section configuration. Time servo will go: a. To the 44.57-second standby position if target-tracked signal is removed. b. To solution for prelaunch time computation for time of flight if target-tracked signal is on.
Burst release	Time slew control circuit monitoring time servomotor drive voltage	1. Locks on burst enable and apparent miss distance at burst circuits are released, restoring them to prelaunch configuration.

The following signals do not normally occur in sequence, but will occur under certain conditions:

Signal	Origin	Effects in Computer
Missile reject	Cam-operated monitor switch in dead-time unit. Operates 5 seconds after FIRE signal	1. The missile reject signal is transmitted to other locations if the missile-away signal has not been received.

Signal	Origin	Effects in Computer
Gimbal limit	Cam-operated switch in turn angle servo. Operates if turn angle reaches plus or minus 1, 260 angular mils. (Aprx 70°)	1. A 2.41g turn order is sent to the missile, directing the turn angle to decrease. 2. The gimbal limit order is sent to the event recorder. 3. The GIMBAL LIMIT light is illuminated on the battery control console.

110. COMPLETION

This completes the sequence of events in the computer. The computer is now ready for the next problem.

CHAPTER 4

BUILT-IN TEST EQUIPMENT

111. INTRODUCTION

a. Theory. The computer, in keeping with the theory of the Nike I system, has built-in test facilities for checking purposes to insure proper operation. These test facilities are the tracking test, static test, and dynamic test modes. Each of these modes is primarily concerned with testing certain sections of the computer. Certain tests are also performed in the action condition. When all are used, practically any trouble can be detected and can be easily isolated. These tests are performed by operating personnel. The correct reading to be obtained from the tests are found in the logbook that comes with each set of equipment. The logbook contains a 6-month supply of sheets for making the daily, weekly, and monthly tests.

b. Test modes. The modes of the computer operation are under control of the 5-position COMPUTER CONDITION switch, located on the computer control panel. The five positions on this switch are: ACTION, STANDBY, TRACKING, STEERING, and PRELAUNCH AND INITIAL TURN. The tracking test mode is obtained when the COMPUTER CONDITION switch is placed in the TRACKING position. The tests which can be performed in the tracking mode are as follows: tracking test, orientation check, amplifier balance (zero check), and amplifier overload. The static test mode is obtained by placing the COMPUTER CONDITION switch in the STEERING position and in the PRELAUNCH AND INITIAL TURN position. In the static test mode, a number of relays are energized in either switch position to condition the computer for testing the steering, prelaunch, and initial turn sections with voltages obtained from the precision built-in voltage dividers. With the COMPUTER CONDITION switch in the ACTION position, the computer can be conditioned for the dynamic mode by operating the DYNAMIC switch. This mode is incorporated to simulate an engagement and to observe the operation of the computer control circuit, the differentiators, and the burst circuit. During an actual engagement, the COMPUTER CONDITION switch must be in the ACTION position. In the STANDBY condition, the GYRO AZIMUTH switch located on the computer control panel is used for setting an artificial gyro azimuth angle into the prelaunch section of the computer. No relays are energized by the COMPUTER CONDITION switch when in the STANDBY position, but all computer and horizontal plotting board servos receive a 120-volt, 400-cycle excitation. The not-standby relay will be deenergized, connecting voltages from the gyro azimuth test voltage divider to the intercept point solver circuit ($-X_I$, $-Y_I$, and H_I) networks. The voltages, as obtained from this voltage divider by the GYRO AZIMUTH switch, will position the A_G servo at 800 angular mil intervals. If the azimuth angle of an expected engagement is known, this switch is positioned so that the A_G servo has a solution for the gyro azimuth angle in the direction of the expected engagement. For the STANDBY condition, the voltages from the gyro azimuth test voltage divider position the B-servo to -565 mils and the time-of-flight predictor to approximately 45 seconds. This will minimize the slewing time for the servos of the prelaunch section at the beginning of an engagement.

112. TRACKING TEST MODE

With the COMPUTER CONDITION switch in the TRACKING position, the following tests can be conducted: tracking test, orientation check, amplifier unbalance (zero check), and amplifier overload. The orientation check and amplifier balance (zero check) tests are conducted under the TRACKING position because it is sure that they will be turned off automatically when the computer is placed in an ACTION condition.

a. Tracking test. The tracking test is used to indicate any tracking error when the missile-tracking radar and the target-tracking radar are automatically tracking the same target. The computer prelaunch configuration is set up somewhat the same as in the ACTION condition, except there is no prediction in the system. This test differs from the static tests because no built-in voltage dividers are used. Sections of the COMPUTER CONDITION switch energize appropriate relays and condition the computer for this test. Additional sections of the COMPUTER CONDITION switch perform the following functions: connect the VELOCITY AND POSITION DIFFERENCE meters (X, Y, and H), located on the computer control panel, to the output of the $\frac{X}{t}$, $\frac{Y}{t}$, and $\frac{H}{t}$ amplifiers; complete the energizing circuit to the steer relays on the position difference relay panel. The 120-volt, 400-cycle excitation voltage to all the computer servos is removed. The target position in spherical coordinates (elevation, azimuth, and slant range) is sent to the computer from the missile-tracking radar and the target-tracking radar by way of their range, azimuth, and elevation data units. The target coordinate converter resolves the target position, with respect to the target-tracking radar, into rectangular coordinates X_T, Y_T, and H_T. The missile coordinate converter resolves the target position, with respect to the missile-tracking radar, into rectangular coordinates X_M, Y_M, and H_M. The $+H_T$, $+Y_T$, and $+X_T$ signals are sent to the $-H_I$, $-Y_I$, and $-X_I$ input network of the intercept point solver through contacts of the energized tracking test relay and contacts of the energized not-standby relay. Since the A_G-servo, B-servo, and time-of-flight predictor are disabled (due to no 120-volt, 400-cycle excitation voltage) no useful outputs are obtained from these circuits. The radar-to-radar parallax, $+X_R$, $+Y_R$, and $+H_R$, along with the $-X_T$, $-Y_T$, and H_T target position as derived in the target coordinate converter, and the $+X_M$, $+Y_M$, and $+H_M$ target position as derived from the missile coordinate converter, are fed to the $\frac{X}{t}$, $\frac{Y}{t}$, and $\frac{H}{t}$ closing speed solver input networks. The X, Y, and H position difference is furnished by the $\frac{X}{t}$, $\frac{Y}{t}$, and $\frac{H}{t}$ amplifiers, respectively. These amplifiers normally deliver output voltages which correspond to velocities and at a scale factor of 25 millivolts per yard per second. If the time-to-intercept servo is positioned at less than 0.25 second, the $\frac{X}{t}$, $\frac{Y}{t}$, and $\frac{H}{t}$ amplifiers will have a constant gain corresponding to a division by 0.25 second. The steer relays are energized by the COMPUTER CONDITIONING switch and the fine relays are energized when the time to intercept is less than 24 seconds. Contacts of these relays connect the t potentiometer cards in the amplifier feedback circuits. With the relays energized and with the t-servo positioned at less than 0.25 second, the $\frac{X}{t}$, $\frac{Y}{t}$, and $\frac{H}{t}$ amplifiers will have a fixed gain of 100, and the output voltages will be at a scale factor of 100 millivolts per yard (0.1 volt/yard). With the VELOCITY AND POSITION DIFFERENCE switch in the RADAR DATA DIFFERENCE YARDS (missile from radar) position, the X, Y, and H VELOCITY AND POSITION DIFFERENCE meters on the computer control panel are connected to the output of the $\frac{X}{t}$, $\frac{Y}{t}$, and $\frac{H}{t}$ amplifiers. The meter full-scale deflection is plus or minus 1,000 yards, and the sensitivity can be increased by 10 by pressing the YDS/10 pushbutton. A 1-yard position difference can thus be read on the meter. The tracking test is a quick test on the system used to see if the missile-tracking radar and the target-tracking radar are actually furnishing the same position information when they are tracking the same target.

b. Orientation check. The orientation check is made to make sure that the azimuth data and elevation data units of the missile-tracking radar and the target-tracking radar are providing accurate data. For this test the ORIENT CHECK switch on the amplifier cabinet assembly must be placed in the ON position, and the radar-to-radar parallax must be reduced to zero. The missile-tracking radar antenna and target-tracking radar antenna are alined so that they point in exactly the same direction. With the ORIENT CHECK switch in the ON position, the orientation check relay in the test relay panel is energized. The tracking test relays will remain energized for the test. The contacts of the orientation check relay will perform the following functions: The $-D_M$ signal is removed and the $-D_T$ signal is connected to the elevation data unit of the missile-tracking radar; the $+D_M$ signal is removed and the $+D_T$ signal is connected to the elevation data unit of the missile-tracking radar; additional contacts remove the $-R_M$ and $+R_M$ voltages from the azimuth data unit of the missile-tracking radar and in their place connect the $-R_T$ and $+R_T$ voltages. The identical $+D_T$ and $-D_T$ voltages are fed to the missile elevation data unit and the target elevation data unit. The same $+R_T$ and $-R_T$ voltages are sent to the missile azimuth data unit and the target azimuth data unit. The magnitude of the X, Y, and H coordinate voltages at the output of the missile coordinate converter and the target coordinate converter should be the same, provided no errors exist in the circuits. With the target-tracking antenna pointed in the same direction as the missile-tracking antenna, the polarity of the output voltages from the target coordinate converter and the missile coordinate converter will be of opposite polarity. The X, Y, and H position differences will appear on the three VELOCITY AND POSITION DIFFERENCE meters. The YDS/10 pushbutton should be depressed to obtain maximum sensitivity. The orientation check will help determine if errors exist in the elevation and azimuth data units. The missile-tracking radar elevation and azimuth data units can then be adjusted, if necessary, to minimize the voltages appearing on the VELOCITY AND POSITION DIFFERENCE meters. At the completion of the test, the radar-to-radar parallax should be reinserted into the computer. There is an interlock system provided with the test so that the ORIENT CHECK switch will be opened when the computer doors are closed.

c. Amplifier balance (zero check) test. The amplifier balance test is performed on all the DC amplifiers used in the computer. The purpose of the test is to measure the DC amplifier output voltage when no signal voltage is applied to the input grid. When the zero-check switch is operated, a relay in the test relay panel is energized. This relay removes the d-c voltages from the data potentiometers, the plus 250-volt supply for the servo potentiometer cards, and the d-c voltages for all the static test voltage divider networks. Therefore, by operating the ZERO CHECK switch, the input voltages to all of the DC amplifiers in the system are removed, and the outputs of all the amplifiers should be zero. The automatic zero-set circuits operate to keep the input grid at ground potential. If there is some leakage in the grid, or in the grid current in the tube, or leakage across the tube socket, the grid cannot be kept at ground potential by the automatic zero-set circuit, and there may be a value of output voltage. The amplifier balance test is performed on the DC amplifier to see that the automatic zero set units are zero setting the amplifiers, and there is no serious leakage into the input grid. In the computer there is provision for rapid switching by means of which the outputs of all the DC amplifiers can be measured in sequence. The ZERO CHECK meter is in the amplifier output circuit when the switches are in the INTERNAL position. The amplifier outputs are measured in sequence by pressing on the amplifier selector switch and rotating it. The meter sensitivity is increased by operating the SENSITIVITY switch. There is an interlock system provided with the ZERO CHECK switches. When the amplifier balance test is performed, the main doors for the left and right frames must be open. At the completion of the test, the ZERO CHECK is automatically removed when the doors are closed.

d. Amplifier overload test. The amplifier overload test is conducted on the computer DC amplifiers by connecting a plus 250-volt supply or a minus 250-volt supply to the amplifier input grid. This connection should cause them to swing into overload, provided there are no diode or thyrite limiters in parallel with the amplifier. This test is conducted with the COMPUTER CONDITION switch in the TRACKING position and with the ZERO CHECK switch operated. A test panel is mounted on the left swinging frame of the computer amplifier cabinet. This same test panel is located on the right swinging frame. The overload test is made by connecting to the grid terminal of the appropriate input network a shielded lead to the +OVERLOAD or to the -OVERLOAD test jack. The built-in ZERO CHECK circuits can be used to measure the amplifier output voltage by rotating the amplifier selector switches to the correct amplifier position. The INTERNAL-EXTERNAL switch is placed to the EXTERNAL position, and the amplifier outputs are brought out to a test access jack. An external meter is connected to this jack to measure the output voltage. The amplifier output voltages recorded in tabular form for the minus 250-volt input and the plus 250-volt input will be found on the monthly log sheet. Note that the amplifiers with limiters should not overload but should dip to a specified voltage. In addition, as each amplifier is overloaded, the corresponding AMPLIFIER UNBALANCE lamp on the computer control panel will flash, except those equipped with limiters. When the LAMP TRANSFER on the corresponding automatic zero set switch is depressed, the individual amplifier overload is indicated.

113. STATIC MODE

In the static test mode of operation, the PRELAUNCH AND INITIAL TURN and STEERING static tests are conducted to check the operation of the three sections of the computer. Eight different problems are introduced to the computer section for each test. The voltages for these tests are derived from built-in voltage dividers.

a. Prelaunch static test. The prelaunch static test is conducted under the PRELAUNCH AND INITIAL TURN position of the COMPUTER CONDITION switch. This static test is used to check the performance of the prelaunch section of the computer. The PRELAUNCH AND INITIAL TURN position of the COMPUTER CONDITION switch completes the energizing circuit for the prelaunch test relays located in the static test relay panel. In addition, a switch section permits the interchanging of pen functions on the horizontal plotting board when the PEN INTERCHANGE pushbutton is depressed. For this test the radar-to-launcher parallax dial readings should be set to zero. Contacts of the prelaunch test relay connect synthetic voltages to the input of the target prelaunch differentiators X_P, $\dot{Y}_P$, and $\dot{H}_P$. For performing checks on the prelaunch section of the computer, only target position and velocity data are used. Only two values of target velocity and two values of target position are necessary to check the prelaunch ballistics and the prelaunch computations. For the target position, the value of X_T, Y_T, and H_T coordinate positions are plus or minus 10,000 yards and plus or minus 40,000 yards. The input position data are at a scale factor of 1 millivolt per yard. The synthetic position data from the test voltage dividers are fed to the input networks of the intercept point solver amplifiers $-X_I$, $-Y_I$, and $-H_I$ through the main X_T, Y_T, and H_T input terminals. The test values for the synthetic target velocity are plus or minus 100 yards per second and plus or minus 400 yards per second. These synthetic velocities are fed through the TEST input to the prelaunch differentiator networks $\dot{X}_P$, $\dot{Y}_P$, and $\dot{H}_P$. During the prelaunch test the differentiator circuits are switched and they become summing amplifier circuits with unity gain. Therefore, the input coordinate test voltages are at a scale factor of 12.5 millivolts per yard per second and the output voltages are at the same scale factor. The input voltages for the eight test positions give a fixed intercept point position for each test. The time-of-flight predictor servo, the ballistic servo, and the gyro azimuth servo will solve the specified problem giving definite dial readings for each of the three servos. Tabulations for each test problem are shown on the log sheets for the normal voltage values at the input network terminals and the allowable tolerances for each reading. These deviations from the normal readings result from tolerances in the computer components due to resistance tolerances, network tolerances, automatic zero set error of the DC amplifier, potentiometer brush tolerances, heating, etc. When a test problem is introduced in the prelaunch section, observe the three servo dials and check their respective readings. If they are not within the specified tolerance limits, further testing should be done to isolate any trouble by measuring the voltages at the terminals of the DC amplifier input networks.

b. Initial-turn static test. The initial-turn static test is conducted under the PRELAUNCH AND INITIAL TURN position of the COMPUTER CONDITION switch. Synthetic input voltages for the test are derived from the initial-turn test voltage divider. Eight problems are introduced by the PRELAUNCH AND INITIAL TURN STATIC TEST switch and the voltages are selected to check the operation of the computing circuits and of the logic circuits (yes-no relay circuits). The relay circuits to be checked are the radar cleared relay, critical turn angle negative relay, skirting turn angle negative and the turn angle zero relay. The problems are such that they check all possible combinations of the logic circuits as well as introduce input data for checking computation accuracy. When a test problem is introduced, the initial-turn section will solve the specified problem, and the output will appear as acceleration orders on the FIN ORDER meter located on the tactical control panel. For all eight problems, the climb angle servo is set at 1,600 angular mils, and the VARIABLE RANGE PARALLAX potentiometer is set to the 5,000- to 6,000-yard position. For each of the problems, the turn angle servo is adjusted manually to obtain the output orders listed. Problem 1 is used to check the radar-cleared circuit operation and the sensitivity of the TA card appearing at the input of the $-S_T$ amplifier. For problem 1, the radar-cleared relay is energized, removing the initial-turn section outputs from the $-S_T$ amplifier. The TA servo potentiometer card remains connected to the input of the

$-S_T$ amplifier. Since the 7g dive order is applied, the G_Y and G_P FIN ORDER meters read -5g when the TA servo is set on 0 angular mils. The gain of the TA card is checked by displacing the TA servo by 100 angular mils. The G_P meter should read zero, and the G_Y meter should still read -5g. Problems 2 through 6, inclusive, are used to introduce different input values and polarities of the CTA with a constant DTA to check and to see that the correct relay combinations are operative. Then the TA servo is positioned until the correct orders appear on the G_Y and G_P FIN ORDER meters. Test problem 7 is used to check the holding features of the CTA and DTA circuits. These circuits are required to hold the input information that was present at the end of roll stabilization. The same input voltages are used in problem 7 that are used in problem 6. However, in problem 7 the roll stabilization -1 relay and the roll stabilization -2 relay are energized, removing the input to the CTA and DTA circuits. If the amplifiers are actually holding, the amplifier outputs will not change from the values they had on problem 6, so the holding features are checked. Problem 8 is introduced to confuse the system. The roll stabilization -1 relay is deenergized, but the roll stabilization -2 relay remains energized, and new input test voltages are applied to the CTA amplifier. This situation will never arise in actual operation, but this problem is used to check the relay operation.

c. Steering static test. The steering static test is performed under the STEERING position of the COMPUTER CONDITION switch. The purpose of this test is to check the operation of the steering section of the computer. The STEERING position of the COMPUTER CONDITION switch completes the energizing circuit to the steering test relays on the steering computer relay panel and the steering test relay on the test relay panel. Synthetic voltages from the steering test voltage divider are used to perform eight individual tests. The eight test problems are selected by the STEERING STATIC TEST switch located on the computer control panel. Synthetic position difference voltages are inserted into the input networks of the $+\frac{X}{t}$, $+\frac{Y}{t}$, and $+\frac{H}{t}$ networks and synthetic velocity voltages into the target steering differentiator ($\dot{X}_T$, $\dot{Y}_T$, and $\dot{H}_T$) networks and the missile differentiator networks ($\dot{X}_M$, $\dot{Y}_M$, and $\dot{H}_M$). The $\dot{X}_M$ $\dot{Y}_M$, and $\dot{H}_M$ and the $\dot{X}_T$, $\dot{Y}_T$, and $\dot{H}_T$ networks have a test input which converts the differentiating amplifiers to summing amplifiers when the relay internal to the differentiator networks are deenergized. Under the test condition the input voltages to the $\dot{X}_M$, $\dot{Y}_M$, $\dot{H}_M$, $\dot{X}_T$, $\dot{Y}_T$, and $\dot{H}_T$ amplifiers represent a velocity and are derived from the built-in test voltage dividers, giving synthetic velocity voltages at the output of the differentiating amplifiers. The position difference test voltages are not applied on the main X_T, Y_T, and H_T inputs to the $+\frac{X}{t}$, $+\frac{Y}{t}$, and $+\frac{H}{t}$ input networks, but are applied through the parallax inputs. The reason for this is that the driven shield data cables are carried from the data units located on the radar antenna mounts directly to the $+\frac{H}{t}$, $+\frac{Y}{t}$, and $+\frac{H}{t}$ input networks. It is not desirable to break these cables with a relay contact or switch because it might introduce leakage currents which would cause position difference errors. Since the main input circuits are so critical, test voltages representing missile-to-target position differences are put into the radar-to-radar parallax inputs of the $+\frac{X}{t}$, $+\frac{Y}{t}$, and $+\frac{H}{t}$ amplifiers. Contacts of the steering test relay remove the radar-to-radar parallax input and in their place connect built-in voltage dividers. The plus 106.67-volt supply or the +S -voltage supply is removed from the missile and target range data units, so that no data comes in on the position leads. The voltage on the position leads is zero and the position difference voltage is applied on the parallax input. In the steering static test, the eight test problems are introduced into the steering section by the STEERING STATIC TEST switch. In addition, for certain problems the GYRO AZIMUTH test switch is used to insert synthetic voltages into the $-X_I$, $-Y_I$, and $-H_I$ networks. This will give different values for the A_G angle and will help in testing the steering section of the computer. In the STEERING TEST, eight test problems are introduced controlled by the STEERING selector switch and the GYRO AZIMUTH selector switch. The eight test problems are set up to give approximately the following times to intercept: 5, 10, 20, 40, and 80 seconds with various altitudes of target and missile determined by the position differences and the velocity differences that are obtained from the voltage divider. The target and missile are headed, not at each other, but offset by varying amounts. Selection of the test problems is done so that certain acceleration orders will be called for and can be read on the FIN ORDER meters, G_Y and G_P, located on the tactical control panel. In making the steering tests, the steering test switch is rotated through its eight positions. Problem 6 and problem 7 are used in conjunction with the GYRO AZIMUTH selector switch to make a complete check on the resolutions in the climb angle servo and the turn angle servo. The problems are set up so that the missile is pointed toward an intercept point in the gyro plane. Then, the climb angle is a particular angle and the turn angle is zero. If now the gyro azimuth servo is rotated 90°, the climb angle is changed from its former value to 90° and the turn angle, which was zero, now becomes 90° minus the former climb angle. Problem 6 is set for a 20° climb angle with the gyro azimuth lined up with the missile velocity vector. The gyro azimuth is then rotated in 800-angular-mil increments by operating the gyro azimuth selector switch. This causes the climb angle servo to go from 20°, up through the zenith and over 20° in the opposite direction. It will cause the turn angle servo to go to the full excursion, +70° and -70°. This is a fairly complete check at a number of points of the climb and turn angle resolution. Also, since the acceleration orders are developed in missile coordinates, the missile coordinates change for the different positions, giving different FIN ORDER meter readings. Since the missile coordinates are different for each problem, the fin order circuits are also checked by this method.

114. COMPUTER DYNAMIC TESTING

a. Dynamic test. The dynamic test is used to supplement existing static tests which do not indicate proper functioning of certain computer circuits. This 2-course test simulates a normal missile flight as closely as possible. It is performed weekly and checks the signaling circuits, the 4-second delay timer, the missile-away circuit, the 4.5-second delay timer, the dead-time unit, the time servo second-per-second circuit, the on-trajectory circuit, the order-limiting circuit, and the burst-order circuit. In the dynamic test, the computer is supplied with fixed target position data and with changing missile position data to simulate an engagement. With the target-tracking radar positioned to fixed values of azimuth, elevation, and range, the missile-tracking radar is initially positioned similarly but at a lesser range. At the appropriate time the missile flight is simulated by insertion of a proper missile range rate. The first course represents the first stages of a missile flight up to ON TRAJECTORY. The second course represents the flight of the missile from ON TRAJECTORY to BURST.

b. Computer data unit check. The computer data unit check makes certain that the data potentiometers are giving smooth, continuous information. With the COMPUTER CONDITION switch in the TRACKING position, aided rates are placed in the azimuth, elevation, and range units, respectively, of the target-tracking and missile-tracking radars. With the rate applied, the plotting boards are used as a visual check of the smoothness of the data from the potentiometer being checked.

c. Differentiator scale factor check. The differentiator scale factor check dynamically checks the operation of the computer differentiators. The normal static tests do not check these units, since the differentiators require a changing input for proper operation. In this check, the system is energized and so much of the sequence of events as is required to enable the differentiators is completed. The target-tracking and missile-tracking radar azimuth and elevation are then set to give equal values of X, Y, and H and range rates are introduced. The effect is that the X, Y, and H rates for a particular radar, as indicated on the VELOCITY AND POSITION DIFFERENCE meters, should be equal.

CHAPTER 5

GYRO AZIMUTH TRANSMISSION SYSTEM

115. GENERAL

The physical orientation of the roll amount gyro in the missile is shown in figure 48. The spin axis of the gyroscope is seen to be horizontal, or parallel to the ground. The balance wheel of the gyro is directed along the longitudinal axis of the missile. The plane of this wheel is called the neutral plane. This neutral plane is always directed along the longitudinal axis of the missile. When the missile lies on the launcher rail ready to be erected, the neutral plane coincides with the direction in which the missile is headed while sitting in the launcher. As the rail is raised, the missile assumes a vertical position. The movement of the missile while being raised does not alter the orientation of the neutral plane. The neutral plane is still directed toward the front of the launcher. If a target is moving, the intercept point will also move. The computer in the battery control trailer maintains a running account of the change in position of the intercept point, and through the gyro azimuth transmission system, is continually reorienting the reference plane. The A_G transmission system transmits this change of position of intercept point from the computer down to the roll amount gyro in the missile. The roll amount gyro is positioned continually from TARGET TRACKED until it is uncaged after FIRE.

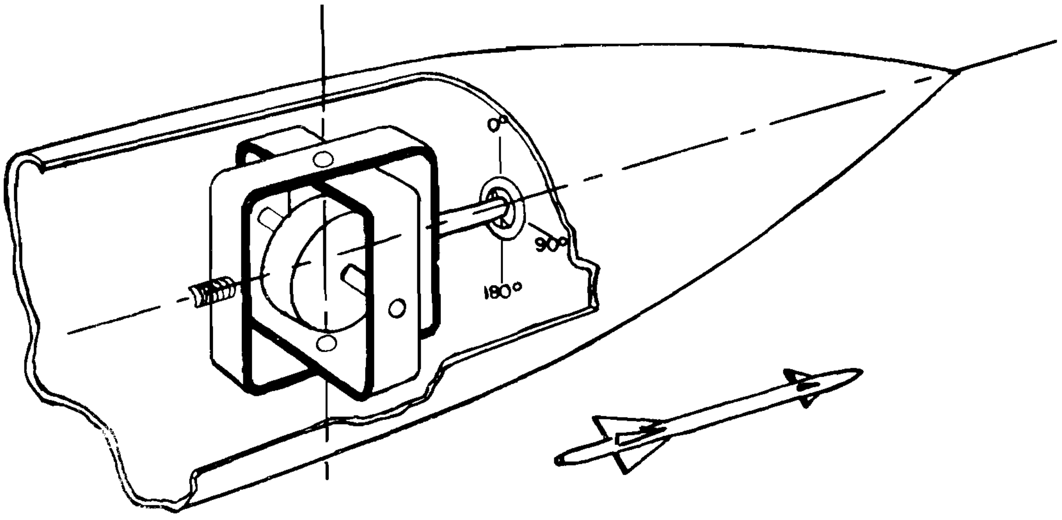

Figure 48. Physical orientation of the roll amount gyro.

116. OVER-ALL PURPOSE

The over-all purpose of the gyro azimuth transmission system is to preset, or orient, the roll amount gyro in the missile toward the ground projection of the intercept point. The gyro is set at the azimuth of the reference plane because, when it is so set, a maximum amount of maneuverability by the missile may be expected before a gimbal limit can occur.

117. BATTERY BLOCK DIAGRAM

In figure II-97 the A_G information originating in the battery control trailer is seen to be transmitted 1,500 to 16,000 feet to the launcher control trailer. Over this distance the A_G information is carried through a special 4-conductor spiral cable. At the launcher control trailer the information is routed to one of the four sections which control launchers through shielded pairs of wires in ordinary cable. This distance may vary between 1,000 and 1,500 feet. At the launcher section the information is again channeled to the particular launcher in which a missile is ready to be fired. The distance between section and launcher is a maximum of 125 feet.

118. COMPONENTS AT THE BATTERY CONTROL TRAILER

a. General. Figure II-97 illustrates the block diagram of components of the A_G transmission system which are located in the battery control trailer. Of the units shown, the 400-cycle oscillator, the resolver amplifier, the line amplifier, and the line transformers are located behind the swinging frame of the right amplifier bay. The A_G resolver is located in the servo and timer assembly in the servo cabinet. The input to the circuits shown is a mechanical input from the gyro azimuth servo system to the rotor of the A_G resolver. The outputs are two 400-cycle voltages: one is called the A_G reference; the other, the A_G signal. These outputs are fed into the interarea cable for transmission to the launcher control trailer.

b. 400-cycle oscillator. Direct current transmission of the A_G information is not practicable for long distances, since voltage drops, due to line resistance, would introduce large errors. Therefore, the A_G angle is transmitted as the phase difference between two 400-cycle sine-wave voltages. Voltage drops in the line will not affect phase, and the line constants will cause only a slight phase error which can be compensated for at the time of installation. A source of very stable alternating voltage is obviously required. The 400-cycle oscillator provides this source. The 20-volt output from the oscillator is fed into the resolver amplifier.

c. Resolver amplifier. The resolver amplifier serves to isolate the oscillator from the remainder of the circuit and to match the high output impedance of the oscillator to the low impedances of the line transformer and of the A_G resolver. There is a single 400-cycle output, 20 volts rms in amplitude. Though this unit is capable of maintaining 50 volts across the resolver winding, it is conservatively operated at 20 volts. One channel conducts the resolver amplifier output to the launcher control trailer through a line transformer and the interarea cable. This is called the A_G reference voltage and may be considered to be of zero phase. A second channel carries the resolver output into the A_G resolver.

d. A_G resolver. The resolver is used to convert a mechanical shaft rotation to a phase shift in the 400-cycle lines. This resolver is an integral part of the A_G servo system of the computer. The A_G angle as solved by the A_G servo system in the prelaunch section of the computer is received by the resolver as a mechanical position of its rotor. The stator of the resolver is excited by the 400-cycle voltage from the resolver amplifier. The output voltage of the resolver, when fed into the input network of the line amplifier, differs from the input voltage only in phase. The phase difference is determined by the position of the resolver rotor. The exact method by which the resolver and line amplifier input network accomplish this phase shift is discussed in TM 9-5000-31. It is sufficient to state at this time that the internal windings of the resolver do not provide the phase shift directly, but must be used in conjunction with the R-C circuit which follows in the line amplifier. Transformer action in the resolver itself may result in some slight variation in the amplitude of the output voltage as the rotor turns. This variation is negligible. The output voltage from the resolver is called the A_G signal and is sent to the A_G line amplifier.

e. A_G line amplifier. The output of the A_G resolver is first circuited across an R-C network at the input to the line amplifier. This network, when connected to the A_G resolver, completes the phase shift in the A_G signal voltage. The rest of the line amplifier is designed to provide the high impedance load needed by the input network and to prevent interaction between the line transformer which follows and the resolver. It is essentially a cathode follower with a gain of close to 1. The 14-volt output is sent to the line transformer.

f. A_G line transformer. The line transformer is used in both the A_G reference and the A_G signal circuits to provide a better impedance match to the spiral 4-conductor cable between the guidance area and the launcher area. It is essentially a repeating coil with either a step-up or a step-down ratio, depending on whether it is used at the transmitting or receiving ends of the line. The high voltage side of the transformer has two taps so that either a 20-to-3 or a 14-to-3 ratio is obtained. Leaving the battery control trailer are two voltages, each about 3 volts rms, 400-cycle; the phases of these voltages differ by a number of electrical degrees corresponding to the degrees of gyro azimuth. The phase of the A_G signal voltage always leads.

g. Missiles prepared circuit. The missiles-prepared circuit is a d-c phantom circuit superimposed on the gyro azimuth transmission system. A d-c current proportional to the number of missiles ready for use is sent from the launcher area to the battery control trailer. This type of circuit was chosen to reduce the number of wire pairs necessary in the interarea cable.

119. COMPONENTS AT THE LAUNCHER CONTROL TRAILER

a. General operation. The A_G reference and A_G signal voltages are each fed into a type KS-9923 transformer. These transformers are connected in a 1-to-1 ratio, so the output voltage has the same amplitude as the input voltage. The transformers serve to isolate input and output cables and provide points for the connection of the missiles prepared circuit. At the launcher control trailer the A_G reference and A_G signal are circuited to the selected launcher section by a deck of the SECTION SELECTOR switch.

b. Phase reversal switch. A phase reversal switch is provided in the A_G signal circuit to correct any connection between the battery control trailer and the launcher section which may have resulted in a reversed plate relation between the A_G reference and A_G signal voltages. If the connections for either voltage were accidentally reversed, an improper indication of the gyro azimuth would be made on the launcher section control panel. The error is readily corrected by operation of the phase reversal switch, an easier operation than the long and tedious examination of the several cables and connectors.

c. Missiles-prepared circuit. It is in the launcher control trailer that the missiles-prepared circuit is phantomed onto the A_G transmission system for transmission of information to the battery control console concerning the number of missiles prepared.

120. COMPONENTS AT THE LAUNCHER SECTION

a. General. The A_G reference and the A_G signal voltages are sent from the launcher control trailer to the selected section only by individually shielded wire pairs in an ordinary cable. Figure II-98 is a block diagram of the A_G transmission circuits for one of the four launcher sections. Each launcher section contains identical components.

b. Line transformers. The A_G reference and A_G signal voltages pass through line transformers (identical to those in the battery control trailer) for proper impedance matching of the cables. In addition, these transformers are connected in a 3-to-20 step-up ratio, providing approximately 20 volts rms for use in the section.

c. Phase adjust control. Immediately following the line transformer in the A_G signal circuit, a phase control adjustment is provided to compensate for any phase differences which may have crept into the voltages between the battery control trailer and the launcher section. This control can vary the phase of the A_G signal between ±10°. The phase control must feed into the R-C network provided at the line amplifier before the phase adjustment is effective. The line constants of both the interarea and launcher area cables will vary under diverse weather conditions and methods of laying. As a result, either the A_G reference or the A_G signal voltage may have acquired a spurious phase change. It has been calculated that the maximum spurious phase change will be about ±4°. The phase adjust control can adequately handle the undesired phase shift which may occur. This is merely a device for returning to the A_G reference and the A_G signal voltages the phase relationship of A_G. This control is set up and adjusted at the time of installation.

d. Modes of operation. Two modes of operation are available at the launcher section. The automatic mode of operation is normally used in the zone of the interior. To obtain automatic operation one must place the GYRO PRESET and the ALERT SELECTOR switches on the launcher section control panel in the AUTO position. This energizes the servo and gyro relay K1AP. The A_G reference and A_G signal voltages are routed through contacts of this relay to the remainder of the A_G transmission circuits at the launcher section.

e. Launcher orient circuit. It is presently conceived that the physical position of the launcher in the launcher area will be determined by the primary direction of fire and the local terrain. Each launcher as it is positioned will assume a certain azimuth, that azimuth being the angle between grid north and the direction of the launcher rail from rear to front. This azimuth is called the launcher heading, or azimuth, and carries the symbol A_L. The neutral plane of the roll amount gyro in the missile will also assume the azimuth A_L. In figure 49 the launcher heading and gyro azimuth are illustrated as measured at the launcher from grid north. The wheel of the roll amount gyro must be rotated through the counterclockwise angle M in order to be directed toward the ground projection of the intercept point. If the A_G angle were used directly in orienting the roll amount gyro, the gyro wheel would aline itself along the dotted line L. It is readily seen that there is a great difference between the azimuth to the intercept point and the line L. The angle M is common to both the launcher heading and the angle between A_G and line L. From the figure it may be seen that the angle which can be used directly in positioning the roll amount gyro is $A_G - A_L$. The angle A_L is determined at the launcher after it is emplaced by using an

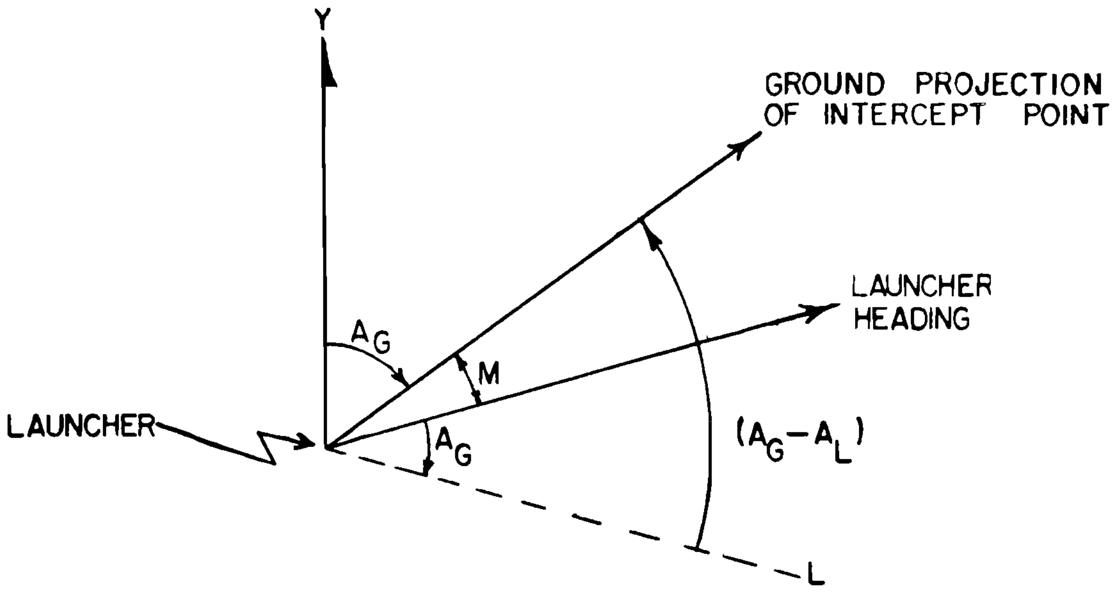

Figure 49. Launcher heading.

aiming circle and backsighting on the target-tracking radar. After occupation of position the heading of a particular launcher designated for action is selected at the launcher section. The A_G signal from the phase adjust control is channeled through contacts of the K1AP relay to the A_G line amplifier. The input network to the line amplifier has an R-C circuit which completes the phase change dictated by the phase adjust control. The phase adjust control and the input network act in a manner identical to the resolver and line amplifier in the battery control trailer. The resolver amplifier following the line amplifier is used to match the output impedance of the line amplifier to the input impedance of one of the launcher orient resolvers which follow. Four individual launcher orient resolvers are located in each launcher section control panel to provide correction for each of the 4 launchers of the section. The rotors of these resolvers are hand-set to correspond to the heading of the launcher with which it is associated. By use of the LAUNCHER DESIGNATE switch at the launcher section control panel, the A_G signal voltage is channeled through the launcher orient resolver of the launcher which has been designated for firing. Since the rotor of the resolver has been set to subtract the launcher heading A_L, the output of the launcher orient resolver will contain a phase angle equal to A_G - A_L. The phase shift is not completed until the voltage has passed through the input network of the A_G converter servoamplifier. This servoamplifier has an R-C input network similar to that of the line amplifier. The launcher orient resolver must be connected to the converter servoamplifier in order to accomplish the desired phase shift. This connection is similar to that of the resolver and line amplifier used in the battery control trailer.

f. A_G converter servo. The A_G reference voltage passes through the line transformer and contacts of the K1AP relay into the input of a resolver amplifier. The resolver amplifier is used to match the impedance of the transformer to that of the resolver which follows and to isolate these portions of the circuit from each other. The converter servo is composed of the A_G data converter, the A_G converter servoamplifier, and the A_G converter modulator. The A_G data converter has within it three smaller components: the data converter resolver, the data converter motor, and the data converter potentiometer. The converter servoamplifier is the comparator of this servo system, and the modulator and motor together are the controller. The data converter resolver is the feedback element of the servo. The feedback is a mechanical positioning of the resolver rotor, altering one input signal. The input signals are the A_G reference voltage and the A_G signal voltage as altered to A_G - A_L. The output is a d-c voltage analogous to A_G - A_L. This voltage is developed by the data converter potentiometer and sent to the gyro preset servo. The A_G reference voltage, at 0° phase, is brought into the data converter resolver for excitation of the stator. This 0° phase is shifted according to the position of the resolver rotor. The output from the resolver rotor feeds into the input network of the converter servoamplifier. This input network is an R-C circuit which completes the phase change dictated by the position of the resolver rotor. The converter servoamplifier is sensitive only to a phase difference between its two input voltages. The phase of the A_G reference voltage, as altered by the A_G data converter resolver, is compared with the phase of the voltage A_G - A_L from the launcher orient resolver. The output voltage varies 0.8 volt per degree of phase difference, saturating at 2 volts. This saturation represents an error of 2½°, whether or not there is a greater error. If there is a phase difference between the two inputs to the servoamplifier, there will be a d-c output voltage to the converter modulator. The converter modulator is similar to other modulators of the computer already studied. The output of the modulator is a 400-cycle voltage, the phase and amplitude of which are controlled by the polarity and magnitude of the d-c error input voltage. This 400-cycle error voltage is used to drive the data converter motor. This motor is a 2-phase motor which receives excitation from a local generator. The motor is geared to the data converter resolver rotor and to the data converter potentiometer brush arm. The change of rotor position will change the phase of the A_G reference voltage serving as one input to the servo system. As the A_G reference voltage changes in phase, the phase error between it and that from the launcher orient resolver becomes smaller. The motor will stop when the modulator output is zero. The converter modulator output will be zero when the converter servoamplifier output is zero. The servoamplifier output will be zero when its two input voltages are in phase. From this, it follows that the data converter shaft will stop when its resolver is positioned to an angle equal to A_G - A_L. The potentiometer brush arms on the same shaft will be turned to the same angle. D-C excitation of this potentiometer causes it to act somewhat like a synchro transmitter which feeds a similar synchro repeater in the gyro preset servo. What actually has been accomplished in the converter servo is to change the phase of the A_G reference voltage from zero to $A_G - A_L$ and then to obtain a d-c voltage analogous to that same angle.

g. Gyro preset servo. The gyro preset servo is composed of the gyro preset servoamplifier in the launcher section, and the roll amount gyro, roll gyro potentiometer, gyro preset motor, and associated gearing in the missile. The input to this servo is the d-c voltage obtained from the data converter potentiometer. This signal and the gyro preset servoamplifier are connected to the selected missile only by the LAUNCHER DESIGNATE switch. When the roll gyro potentiometer arms are not turned through the same angle as those of the data converter potentiometer, a d-c error voltage passes from the roll gyro potentiometer to the gyro preset servoamplifier. This servoamplifier gives an output which is returned to the gyro preset motor in the missile through contacts of the LAUNCHER DESIGNATE switch. The circuit is so arranged that for an angular error in one direction, the current flows through the motor in such a direction as to cause the motor to reduce the error. For the opposite angular error, current flow is in the opposite direction. This motor positions the roll gyro potentiometer arms until no d-c error voltage is sent to the gyro preset servoamplifier. When this occurs, the shaft of the preset servo has reached the angle $A_G - A_L$. The roll amount gyro in the missile is also geared to the gyro preset motor, so it also has been turned through the angle $A_G - A_L$. A null meter is provided on the section control panel to indicate when the gyro preset operation has been accomplished.

h. Manual operation. In the event that the interarea cable has been destroyed or severed, the A_G transmission system at the launcher section must be operated in the manual mode. To obtain the manual mode, the ALERT SELECTOR and GYRO PRESET switches on the launcher section control panel must be placed in the MANUAL position. This deenergizes the K1AP relay. The 400-cycle oscillator present in the launcher section now provides a stable 400-cycle sine wave output voltage to the resolver amplifier. This resolver amplifier matches the impedance of the oscillator to that of the manual preset resolver. It also provides isolation between these two elements. The gyro azimuth is transmitted from the battery control trailer to the launcher area by wire or radio communication means. An operator continually sets the manual preset resolver according to these instruc- manual preset resolver. The output of the resolver amplifier is sent to two different channels. In the A_G signal channel, this voltage, after passing through the manual preset resolver, passes into the R-C input network of the line amplifier. This circuit completes the phase change dictated by the position of the rotor of the manual preset resolver. The output of the resolver amplifier is also sent in to the A_G reference channel through the A_G reference resolver amplifier. From here on, the operation of the components in the launching section is the same as in automatic operation.

APPENDIX I

SYMBOLS AND ABBREVIATIONS

Section I. SYMBOLS

Symbol	Definition
A	Azimuth
A_G	Gyro azimuth
A_L	Launcher azimuth
A_M	Missile azimuth
A_T	Target azimuth
B	Ballistic elevation angle
D	Slant range
D_B	Slant range for ballistic circle or radius of constant time circle
D_M	Missile slant range
D_T	Target slant range
E	Angular height
E_M	Missile angular height
E_T	Target angular height
g	Gravity
G_P	Order transmitted to the missile pitch fin
G_Y	Order transmitted to the missile yaw fin
H-axis	Altitude axis

Symbol	Definition
$\dot{H}$	Vertical component of actual closing velocity between the target and the missile
H-distance	Vertical distance between missile and target
H_B	Altitude of the center of the constant time circle
H_G	Gyro coordinate vertical axis
$\dot{H}_{GM}$	Component of missile velocity along the gyro H-axis
H_I	Altitude of the predicted intercept point above the designated launcher
H_L	Vertical component of launcher parallax
H_M	Vertical distance between missile and MTR
$\dot{H}_M$	Vertical component of missile velocity
$\dot{H}_P$	Vertical component of target velocity during the prelaunch phase
H_R	Vertical component of radar parallax from MTR to TTR
H_{SL}	H-stylus left
H_{SR}	H-stylus right
$\frac{H}{t}$	Vertical component of ideal closing velocity between missile and target
H_T	Vertical distance between target and TTR
$\dot{H}_T$	Vertical component of target velocity during the steering phase
L_i	Line of intersection; the line of intersection between the gyro reference plane and the missile velocity slant plane
R_B	Range to center of constant time circle

Symbol	Definition
R_G	Ground range, TTR to launcher area center
R_I	Ground range to predicted intercept point from designated launcher
R_M	Missile ground range from missile-tracking radar
R_T	Target ground range from target-tracking radar
S_C	Steering error component along missile climb axis
S_{GH}	Steering error component along the gyro H-axis
S_{GY}	Steering error component along the gyro Y-axis
S_{GX}	Steering error component along the gyro X-axis
S_i	Steering error component that lies along the line of intersection (the terms P_i and S_{GPS} have also been used to symbolize this vector)
S_{PF}	Steering error component perpendicular to the pitch fin
S_T	Steering error component along the missile turn axis
S_V	Steering error component along the missile velocity axis
S_X	Steering error component along the earth X-axis
S_Y	Steering error component along the earth Y-axis
S_{YF}	Steering error component perpendicular to the yaw fin
t	Time, time of flight, time to intercept
t_d	Dead time, the period (7 seconds) timed by the dead-time unit
t_{SLR}	Time stylus left and right
V_i	Component of missile velocity along the line of intersection (the terms Q_i and S_{GPS} have also been used to symbolize this vector)

Symbol	Definition
V_M	Missile velocity
X-axis	Earth east-west axis
X	East-west component of actual closing velocity between the target and the missile
X-distance	East-west distance between target and missile
X_G	Gyro coordinate east-west axis
X_{GM}	Distance to missile measured along gyro X-axis
$\dot{X}_{GM}$	Missile velocity component along gyro X-axis
X_I	East-west distance to predicted intercept point from designated launcher
X_L	East-west component of launcher parallax
X_M	East-west distance between missile and MTR
$\dot{X}_M$	East-west component of missile velocity
$\dot{X}_P$	East-west component of target velocity during the prelaunch phase
X_R	East-west component of radar parallax
X_{SL}	X-stylus left
X_{SR}	X-stylus right
$\frac{X}{t}$	East-west component of ideal closing velocity between the missile and the target
X_T	East-west distance between target and TTR
$\dot{X}_T$	East-west component of target velocity during steering phase
Y	Yaw

Symbol	Definition
Y-axis	Earth north-south axis
$\dot{Y}$	North-south component of actual closing velocity between the target and the missile
Y-distance	North-south distance between target and missile
Y_G	Gyro coordinate north-south axis
Y_{GM}	Distance to missile measured along gyro Y-axis
$\dot{Y}_{GM}$	Missile velocity component along gyro Y-axis
Y_I	North-south distance to predicted intercept point from designated launcher
Y_L	North-south component of launcher parallax
Y_M	North-south distance between missile and MTR
$\dot{Y}_M$	North-south component of missile velocity
$\dot{Y}_P$	North-south component of target velocity during prelaunch phase
Y_R	North-south component of radar parallax
Y_{SL}	Y-stylus left
Y_{SR}	Y-stylus right
$\frac{Y}{t}$	North-south component of ideal closing velocity between the missile and the target
Y_T	North-south distance between target and TTR
Y_T	North-south component of target velocity during steering phase

Section II. ABBREVIATIONS

Abbreviation	Definition
Acq	Acquisition radar
Az	Azimuth
BCA	Battery control area
BCO	Battery control officer
BCT	Battery control trailer
BTB	Burst time bias
CA	Climb angle
DTA	Difference turn angle
El	Elevation
IP	Predicted intercept point
IT	Initial turn
LC	Launching control
LCA	Launching control area
LCT	Launching control trailer
Lchr	Launcher
M&ST	Maintenance and spares trailer
MA	Missile away
MSL	Mean sea level
MTR	Missile-tracking radar
MVSP	Missile velocity slant plane

Abbreviation	Definition
O Lim	Order limiting
RCT	Radar control trailer
Rg	Ground range
RTF	Ready to fire
RS	Roll stabilization
+S	Scale factor voltage
SAM	Surface-to-air missile
TA	Turn angle of the missile
TAZ	Turn angle zero
TDE	Target differentiator enable
T Des	Target designate
TGSA	Target ground speed amplifier
TSL	Time slew
TTR	Target-tracking radar

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