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Autonomous Weapon Guidance

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Precision guided weapons are becoming more and more sophisticated. Terminology such as Global Positioning System, Differential Global Positioning System, Laser Radar, Synthetic Aperture Radar, Inertial Navigation System, and Kalman Filtering have become prevalent as weapons are made smarter and more precise. This study comprised an examination of the midcourse and terminal navigation and guidance schemes implemented in the Armament Directorate's advanced development guidance programs. The implementation of these navigation and guidance schemes allow the weapon to be more precise. This report describes these implementations and provides a background for the engineer or manager on the elements that comprise an autonomously guided weapon.

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PREFACE

This final report is the culmination of an in-house effort directed by Lt Col De Lorenzo, MNG Division Chief, and Mr Rick Wehling, MNG Technical Director, to gain an expertise on the contractor midcourse and terminal guidance implementations for the Division's 6.3 advanced guidance programs. This study involved attending program reviews and studying contractor reports and diagrams in-depth. This report is not intended to be a summary of the contractors' reports, but rather a document that will provide a background for the engineer or manager on the elements that comprise an autonomous guided weapon.

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SECTION I

INTRODUCTION

The precision guided weapons of today are becoming more and more sophisticated. Terminology such as global positioning system (GPS), differential global positioning system (DGPS), ring laser gyro (RLG), laser radar (Ladar), synthetic aperture radar (SAR) seeker, inertial navigation system (INS), and Kalman Filtering have become prevalent as weapons are made smarter and more precise. This technology has made weapon autonomy possible. For example, a fighter aircraft flies a precision guided air-to-ground weapon into the vicinity of a predetermined target, launches the weapon and returns to base. The weapon then flies autonomously, guiding into and destroying the target. Before weapon launch, a transfer alignment from the aircraft to the weapon occurs to initialize the weapon's INS. After launch the weapon flies a midcourse guidance phase using only the INS with GPS position and velocity updates for guidance. If the weapon has a seeker, it is activated at the predetermined terminal area, usually several kilometers from the target location. This begins the terminal guidance phase. In this phase the GPS updates are suspended, and the seeker provides targeting information to the guidance processor via the target detection and classification algorithm contained in the image processor for locating and tracking the target. This mode of tracking the target is continued until within several hundred meters of the target location at which time the seeker updates are suspended and the weapon guides inertially to target impact. If the weapon is seekerless, it continues using the INS with GPS position and velocity updates until the weapon is within several hundred meters of the target location. At this time the GPS position and velocity updates are suspended and the weapon guides inertially to target impact.

The purpose of this document is to describe autonomous weapon guidance so that the engineer may become more familiar with the implementation of guidance systems in air-to-ground missiles. Section II explains inertial navigation, coordinate frames, navigation hardware and navigation errors. A description of WGS 84 is contained in Section III. The Global Positioning System is discussed in Section IV. Section V describes Kalman Filtering, while INS/GPS integration is covered in Section VI. An explanation of seekers is given in Section VII. The types of missile guidance are described in Section VIII. Examples of air-to-ground missile guidance system implementation from real-world test programs, Advanced Technology Ladar System (ATLAS), Autonomous Synthetic Aperture Radar Guidance (ASARG), and Operational Concept Demo (OCD), are contained in Section IX. Finally, a concluding summary is given in Section X.

SECTION II

INERTIAL NAVIGATION

Autonomous guidance requires the knowledge of missile location and speed at all times. An inertial navigation system provides this information. Inertial navigation is the process of providing position and velocity of a vehicle with respect to some reference, typically the earth's surface, based solely on inputs from self-contained acceleration sensing instruments. Gyroscopes provide the acceleration direction, while accelerometers provide the magnitude of the acceleration. The sensed acceleration vector data, corrected for gravity, is integrated to obtain vehicle velocity. The velocity data is then integrated to obtain vehicle position.

An inertial navigation system can be either gimbaled or strapdown. Most missile applications today use strapdown systems since they are smaller in size and more reliable than gimbaled systems. In a strapdown system, the accelerometers and gyros are directly mounted on the airframe. Hence, these sensors measure components of vehicle motion in airframe or "body" coordinates. An inertial navigation computer is used to calculate the horizontal and vertical components of acceleration, velocity, and position in the "navigation" coordinates based upon outputs from the body mounted sensors.

As stated above, inertial position and velocity are computed with respect to some reference frame, or coordinate system. These coordinate systems are right-hand Cartesian coordinate frames consisting of mutually perpendicular x, y, and z axes. However, their origins and relative orientations of the axes differ. The earth frame, the navigation frame, and the body frame are typical frames used in navigation systems. The earth centered earth fixed (ECEF) frame has its origin at the earth's center-of-mass and its z-axis aligned with the earth's spin axis. This frame is fixed to the earth and, therefore, rotates with the earth. The navigation frame has its origin on, or close to the surface of the earth, usually at the location of an inertial navigation system. A typical orientation of the navigation frame axes has the x-y plane tangent to the surface of the earth with the positive x-axis pointing true north, the positive y-axis pointing east, and the positive z-axis pointing down. This is the north-east-down (NED) convention. Another frequently used navigation frame has an orientation with the x-axis pointing east, y-axis pointing north, and the z-axis pointing up. This is the east-north-vertical (ENV) convention. The body frame is attached to the rigid body whose motion is being analyzed, this body being that of the missile. The origin of the body frame is usually located at the center-of-mass of the body, and the axes of the frame have the same rotational motion as the body. A typical orientation of the axes has the positive x-axis pointing out the nose of the missile, the positive y-axis pointing out the right side of the missile, and the positive z-axis pointing out the bottom of the missile.

Since several coordinate frames are used for navigation, it is necessary to transform a mathematical vector coordinatized in one frame to the equivalent vector coordinatized in another frame. Several methods for performing this transformation exist. Three such transformations, direction cosine matrix, Euler angles, and quaternions, are described.

The direction cosine matrix is derived from the transformation of unit vectors from one coordinate frame into another. A "standard" direction cosine matrix is obtained if the unit vectors of both frames are right-hand-orthogonal. Figure 1 shows two right-hand-orthogonal coordinate frames, frame 1 and frame 2. The angle α is the angular difference between the two frames in the x-y plane. The z-axis of the two frames is always the same. A simple, cookbook formula exists for computing the standard direction cosine matrix for transforming from frame 1 to frame 2, as shown in Equation 1. A "1" goes on the diagonal element of the axis that coincides with both frames, which in this case is the 3-axis, or z-axis. A "0" goes on the off diagonal elements corresponding to the row and column of the diagonal element that contains the number "1". The cosine of the angle separating the two frames goes on the other two diagonal elements. The sine of the angle separating the two frames goes on the remaining off diagonal elements. If the axis of rotation is odd, then a "-sin" goes on the lower row off diagonal element. If the axis of rotation is even, the "-sin" goes on the upper row off diagonal element.

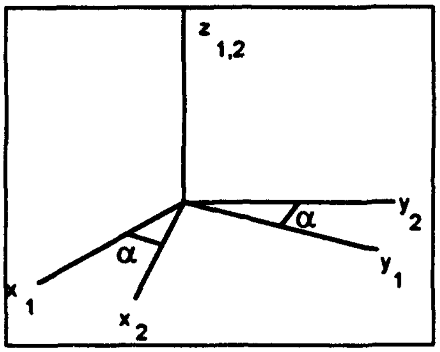

Figure 1

Coordinate Transformation

$$C_2^1 = \begin{bmatrix} \cos\alpha & \sin\alpha & 0 \\ -\sin\alpha & \cos\alpha & 0 \\ 0 & 0 & 1 \end{bmatrix} \qquad (1)$$

where C_2^1 is the direction cosine matrix for a 1-frame to 2-frame rotation

Thus, a vector coordinatized in the 1-frame, $\bar{R}^1$, can be rotated to a vector in the 2-frame, $\bar{R}^2$, through the transformation shown in Equation 2.

$$\bar{R}^2 = C_2^1 \bar{R}^1 \qquad (2)$$

Not all coordinate frame transformations involve the standard direction cosine matrix. Transforming from ECEF to NED results in a non-standard transformation, as shown in Equation 3.

$$C_n^e = \begin{bmatrix} -\sin l \cos L & -\sin L & -\cos l \cos L \\ -\sin l \sin L & \cos L \;\; \lambda & -\cos l \sin L \\ \cos l & 0 & -\sin l \end{bmatrix} \quad (3)$$

where l is latitude and L is longitude.

One must note, however, that the rate of change of a vector is different when viewed from different axes systems. When there are two or more axes systems rotating relative to each other, the need for the Coriolis equation arises. The equation of Coriolis is given in Equation 4 and expresses that the motion of an object as viewed from a reference frame is equal to the motion as seen from the moving frame, plus the motion resulting from the relative angular velocity of the moving frame with respect to the reference frame. This concept is illustrated in Figure 2.

$$\frac{{}^{i}d\bar{R}}{dt} = \frac{{}^{p}d\bar{R}}{dt} + \bar{\omega}_{ip} x \bar{R} \quad (4)$$

$\frac{{}^{i}d\bar{R}}{dt}$ = velocity vector of point in question as seen by frame i

$\frac{{}^{p}d\bar{R}}{dt}$ = velocity of point in question as seen by frame p

$\bar{\omega}_{ip}$ = angular velocity of frame p with respect to frame i.

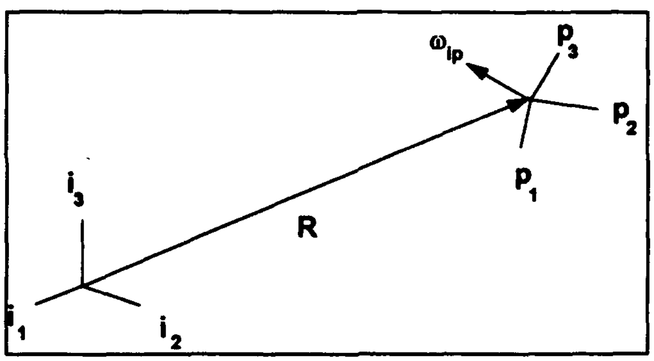

Figure 2

Illustration of Coriolis Equation

Euler angles are another set of parameters which describe the relationship between two coordinate frames. For missile applications, the Euler angles provide a coordinate transformation from the body frame to the navigation frame. These angles are ψ, θ, and ϕ. In other words, Euler angles provide heading, pitch, and roll information of the missile. This collective information is called attitude and is shown in Figure 3.

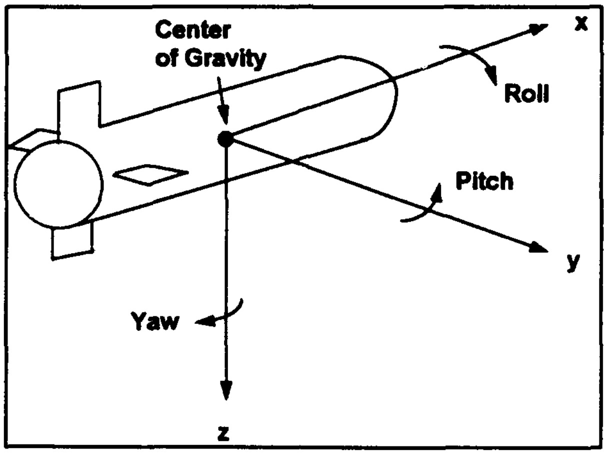

Figure 3

Missile Reference Axes

There are three Euler angle rotations, each of which can be represented by its respective direction cosine matrix. The order of the rotations determines the particular set of Euler angles. If a different order of rotations is used, a different Euler angle set will result. Thus, after a rotation order is chosen, it must be used consistently. A convention used for missile and aircraft applications is to begin with the missile flying straight and level, heading true north, and with constant airspeed and altitude. This configuration makes the Euler angles all zero and the navigation and body frames coincident. Now, two intermediate frames, the 1-frame and the 2-frame, are used to define the three rotations. The first rotation rotates the navigation frame to the 1-frame about the z-axis through the heading angle, y. The second rotation rotates the 1- frame to the 2-frame about the y-axis through the pitch angle, θ. The final rotation rotates the 2-frame to body frame about the x-axis through the roll angle, ϕ. These Euler angle rotations are illustrated in Figure 2. The direction cosine matrix for each respective rotation is shown in Equations 5 - 7.

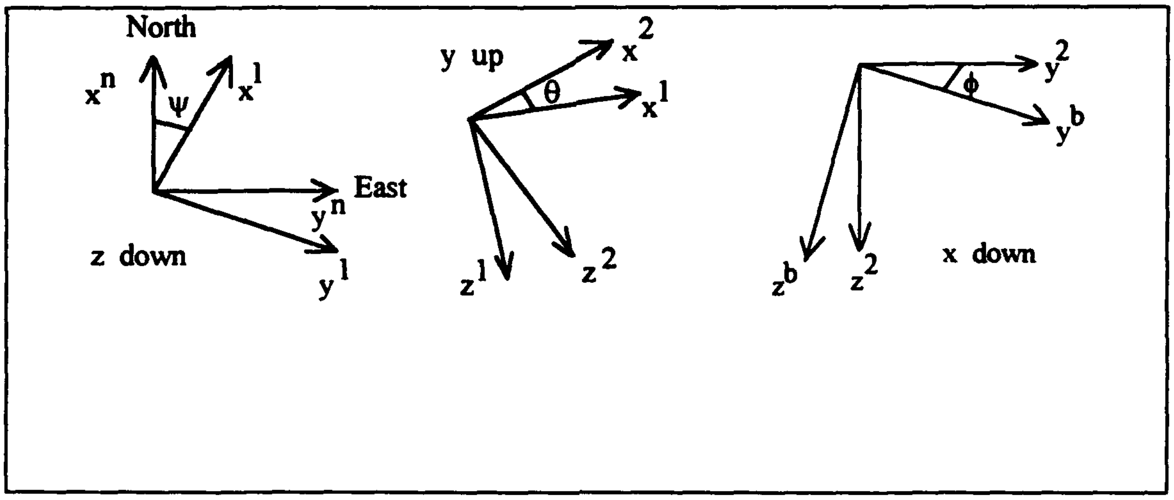

Figure 2

Euler Angle Rotations

$$C_1^n = \begin{bmatrix} \cos\psi & -\sin\psi & 0 \\ \sin\psi & \cos\psi & 0 \\ 0 & 0 & 1 \end{bmatrix} \qquad (5)$$

$$C_2^1 = \begin{bmatrix} \cos\theta & 0 & \sin\theta \\ 0 & 1 & 0 \\ -\sin\theta & 0 & \cos\theta \end{bmatrix} \qquad (6)$$

$$C_b^2 = \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos\phi & -\sin\phi \\ 0 & \sin\phi & \cos\phi \end{bmatrix} \qquad (7)$$

The complete direction cosine matrix transformation from the body frame to navigation frame is the product of the individual direction cosine matrices in reverse order, as shown in Equation 8.

$$C_b^n = C_1^n C_2^1 C_b^2$$

$$= \begin{bmatrix} \cos\psi\cos\theta & \cos\psi\sin\theta\sin\phi - \sin\psi\cos\phi & \cos\psi\sin\theta\cos\phi + \sin\psi\sin\phi \\ \sin\psi\cos\theta & \sin\psi\sin\theta\sin\phi + \cos\psi\cos\phi & \sin\psi\sin\theta\cos\phi - \cos\psi\sin\phi \\ -\sin\theta & \cos\theta\sin\phi & \cos\theta\cos\phi \end{bmatrix} \tag{8}$$

If this direction cosine is computed, the Euler angles can be extracted through the inverse trigonometric functions shown in Equation 8. Note a singularity occurs at $\theta = +/- 90$. Thus, θ is only valid for $-\frac{\pi}{2} \leq \theta \leq \frac{\pi}{2}$.

$$\theta = -\sin^{-1} c_{3,1}, \quad \phi = \sin^{-1}\frac{c_{3,2}}{\cos\theta}, \quad \psi = \sin^{-1}\frac{c_{2,1}}{\cos\theta} \tag{9}$$

To circumvent these problems an additional Euler set with a different rotation order can be used, or quaternions can be employed.

The quaternion is a four parameter coordinate transformation. It is often represented as a four parameter vector, **q**, as shown in Equation 10.

$$\mathbf{q} = \begin{bmatrix} a \\ b \\ c \\ d \end{bmatrix} \tag{10}$$

The four elements of the **q** vector represent components of the sine and cosine of half the Euler angle and are known as Euler's symmetric parameters. It is useful to collect the last three elements of **q** into a three dimensional vector, **r**, and relate to the rotation vector shown in Figure 5. If the quaternion is used to represent the relation between two coordinate frames, the quaternion elements have the definition given in Equation 11 .

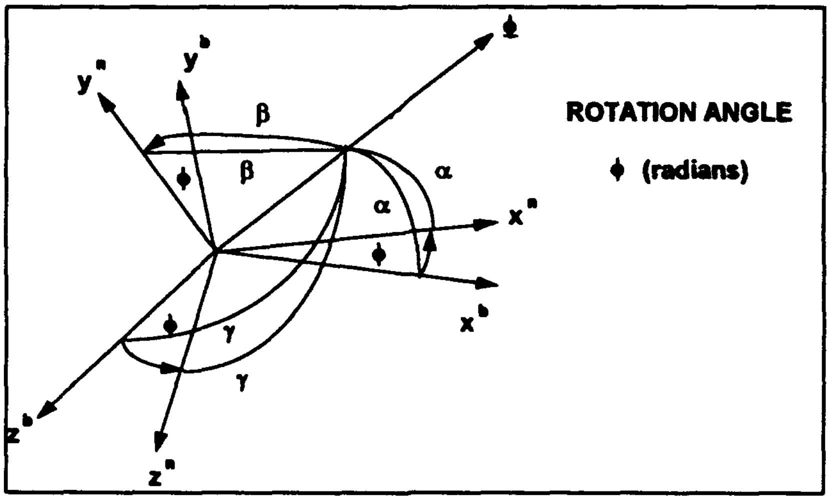

Figure 5

Quaternion Geometry

$$\mathbf{q} = \begin{bmatrix} \cos\frac{\phi}{2} \\ \sin\frac{\phi}{2}\cos\alpha \\ \sin\frac{\phi}{2}\cos\beta \\ \sin\frac{\phi}{2}\cos\gamma \end{bmatrix} \qquad (11)$$

Since this is a four parameter representation of a three degree of freedom rotation, there must be one constraint among the four parameters. This constraint is shown in Equation 12.

$$a^2 + b^2 + c^2 + d^2 = 1 \qquad (12)$$

Once the quaternion rotation parameter is obtained, the actual vector transformation from v^b into v^n can be accomplished in several ways. The first method is a direct transformation as shown in Equation 13.

$$\bar{v}^n = \bar{v}^b + 2a\bar{\rho} \times \bar{v}^b + 2\bar{\rho} \times (\bar{\rho} \times \bar{v}^b) \qquad (13)$$

No singularity problems exist with this method. The next method involves an intermediate step of direction cosine matrix computation.

$$\bar{v}^n = C_b^n \, \bar{v}^b \qquad (14)$$

A direction cosine matrix can be formed directly from the quaternion elements through the use of Equation 15. Thus, given the quaternion, the Euler angles can be derived through the use of Equation 9 if they are needed in a particular application. However, the same singularity problems will exist for $-\frac{\pi}{2} \le \theta \le \frac{\pi}{2}$.

$$C_b^n = \begin{bmatrix} a^2+b^2-c^2-d^2 & 2(bc-ad) & 2(ac+bd) \\ 2(ad+bc) & a^2-b^2+c^2-d^2 & 2(cd-ab) \\ 2(db-ac) & 2(ab+cd) & a^2-b^2-c^2+d^2 \end{bmatrix} \qquad (15)$$

Now that a description of the coordinate frames, coordinate frame transformations, and Euler angle, or attitude, calculations has been given, the next step is to determine the inertial navigation equations. The equations developed are for a geographical navigation frame, one that has its horizontal axes directed along locally North and East axes. A typical geographic implementation of inertial navigation position rate equations calculates latitude and longitude rate directly as functions of computed airframe North and East velocity components. These equations are shown in Equations 16 - 20. These velocity components are then integrated to calculate latitude and longitude.

$$P_N = \frac{V_E}{R_0}\left[1 - \frac{h}{R_0} - e\sin^2 l\right] \qquad (16)$$

$$P_E = -\frac{V_N}{R_0}\left[1 - \frac{h}{R_0}\, e(1 - 3\cos^2 l)\right] \qquad (17)$$

$$\frac{dl}{dt} = -P_E, \quad l = \int_0^t \frac{dl}{dt}\, d\tau + l_0 \qquad (18)$$

$$\frac{dL}{dt} = P_N \sec l, \quad L = \int_0^t \frac{dL}{dt}\, d\tau + L_0 \qquad (19)$$

$$\frac{dh}{dt} = V_Z + u_2, \quad h = \int_0^t \frac{dh}{dt}\, d\tau + h_0 \qquad (20)$$

P_N, P_E = North and East component of airframe position vector angular rate relative to earth

l, L, h = airframe latitude, longitude, and altitude

V_N, V_E, V_Z = Airframe North, East, and vertical velocity relative to earth

R_0 = Mean earth radius at equator

e = earth oblateness coefficient, equal to 1/298

u_2 = Barometric altitude divergence control signal

The velocity components, V_N, V_E, and V_Z would be calculated from the velocity rate equations shown in Equations 21 - 25.

$$W_N = W_E \cos l, \quad W_Z = W_E \sin l, \qquad (21)$$

$$w_N = \rho_N + W_N, \quad w_E = \rho_E, \quad w_Z = \rho_N \tan l + W_Z \qquad (22)$$

$$\frac{dV_N}{dt} = A_N - (W_Z + w_Z)V_E + w_E V_Z, \quad V_N = \int_0^t \frac{dV_N}{dt}\, d\tau + V_{N_0} \qquad (23)$$

$$\frac{dV_E}{dt} = A_E - (W_N + w_N)V_Z + (W_Z + w_Z)V_N, \quad V_E = \int_0^t \frac{dV_E}{dt}\, d\tau + V_{E_0} \qquad (24)$$

$$\frac{dV_Z}{dt} = A_Z - g - w_E V_N + (W_N + w_N)\, V_E + u_1\,, \quad V_Z = \int_0^t \frac{dV_Z}{dt}\, d\tau + V_{Z_0} \tag{25}$$

V_N, V_E, V_Z = Airframe North, East, and vertical velocity relative to earth

A_N, A_E, A_Z = Airframe North, East, and vertical specific force accelerations

w_N, w_E, w_Z = Airframe North, East, and vertical components of nav frame angular rate relative to inertial

W_N,W_Z = North and vertical earth rate components

W_e = Earth inertial rotation rate magnitude

g = local plumb-bob gravity

u_1 = Barometric altitude divergence control signal

l = Latitude

ρ_N, ρ_E = North and East components of airframe position vector angular rate relative to earth

These navigation equations along with the Euler angles are calculated by the navigation computer, usually at a 100 Hz rate. However, over time the accuracy of these calculations degrades due to inherent errors in the accelerometers and gyros. For example, gyros have a bias uncertainty which causes the gyro to give a reading when there is no platform rotation. Noise associated with the bias uncertainty causes the bias to change, or drift over time. This error is known as gyro drift. A tactical grade INS has gyro drifts on the order of 1-10 degrees per hour, while a higher grade INS has gyro drifts on the order of 0.01-0.05 degrees per hour. Random walk is an error caused by a drift that occurs when integrating the noise in the bias uncertainty. A tactical grade IMU has random walk on the order of 0.05-0.125 deg Another error is scale factor error. Rotation of the gyro yields a linear output from -x deg/sec to x deg/sec. A scale factor is calibrated for these ranges. However, the scale factor is not exactly linear which results in errors. Typical INS gyro scale factor error ranges from 3-500 parts per million. Accelerometers have bias and scale factor errors as well. Accelerometer bias exists when there is no platform acceleration, yet there is still a reading from the accelerometer. A tactical grade INS has accelerometer bias on the order of 200 -1000 μG, whereas a higher grade INS has accelerometer bias less than 100 μG. To lessen the effect of these errors on the navigation equations, another source of navigation information can be used to aid the INS calculations. One such navigation aid is the Global Positioning System discussed in Section IV.

SECTION III

WORLD GEODETIC SYSTEM 1984 (WGS 84)

As stated in Section II, inertial navigation is computed with respect to a reference coordinate system. The Department of Defense (DoD) has a common reference system used for supporting DoD operations worldwide. This reference system, WGS 84, was developed by the Defense Mapping Agency (DMA). DMA produces numerous mapping, charting, geodetic, gravimetric, and digital products in support of DoD. WGS 84 is a geocentric system, termed a world geodetic system, that provides the basic reference frame and geometric figure for the earth, models the earth gravimetrically, and provides the means for relating positions on various local geodetic systems to an earth-centered, earth-fixed coordinate system. Three other geocentric system have been developed. These are World Geodetic System 1960 (WGS 60), WGS 66, and WGS 72. Each system has been more accurate than its predecessor. An illustration of the WGS 84 system is shown in Figure 3.

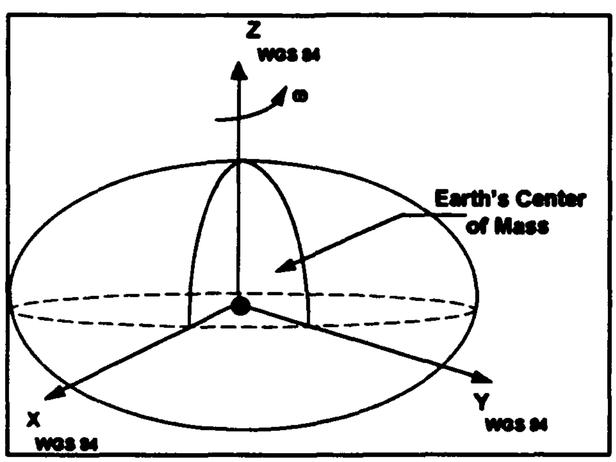

Figure 3

WGS 84 Coordinate System

In addition to the earth's natural suface, there are two other earth figures used in geodetic applications. The first is the geometric surface, the ellipsoid. Four parameters are used to define the WGS 84 Ellipsoid. These parameters are the semimajor axis (a), the earth's gravitational constant (GM), the normalized second degree zonal coefficient ($\overline{c_{2,0}}$), and the angular velocity of the earth (ω). The parameter values are shown in Table 1.

Semimajor axis	a = (6378137 ± 2) meters
Earth's Gravitational constant	$(3986005 \pm 0.6) \times 10^8\ m^3 s^{-2}$
Normalized Second Degree Zonal Coefficient	$(-484.166685 \pm 0.00130) \times 10^{-6}$
Earth Angular Velocity	$\omega = 7292115 \pm 0.1500) \times 10^{-11}$ rad/sec

Table 1

WGS 84 Ellipsoid Parameters

Using the four defining parameters in Table 1, it is possible to derive the other parameters associated with the WGS 84 Ellipsoid. These derivations are contained in Reference 2.

The second earth figure associated with geodetic applications is the geoid. The geoid is defined as that particular equipotential surface of the earth that coincides with mean sea level over the oceans and extends hypothetically over the entire earth. In a mathematical sense, the geoid is also defined as so many meters above or below the ellipsoid. A definition of the WGS 84 Geoid Heights can be found in Reference 2.

The importance of WGS 84 for air-to-surface missile applications is that a common reference system exists for missile inertial navigation and for specifying target locations. In addition, as will be discussed in the next section, the Global Positioning System provides its position measurements in WGS 84 coordinates. Thus, this common reference system facilitates autonomous weapon guidance.

SECTION IV

GLOBAL POSITIONING SYSTEM

The Global Positioning System (GPS) is a satellite-based radionavigation system that continuously provides position and velocity information in WGS 84 coordinates to military and civilian users throughout the world. The users must have receivers capable of processing the signals broadcast by the satellites. The system can be divided into three segments, the space segment, the user segment, and the control segment.

The space segment consists of the GPS satellites in orbit 10,898 nautical miles above the earth in six 55-degree orbit planes. The full GPS constellation consists of 21 satellites plus 3 active on-orbit spares.

The user segment consists of the GPS receivers on the ground, aboard ships, and aboard aircraft. The receiver picks up GPS signals from four or more satellites. Based on signal transmission time from the satellites, the receiver processor can calculate the range to the satellite. Obtaining range information from four satellites allows the receiver to calculate its current position, and based on doppler frequency information, it can determine its velocity. It should be noted that the GPS receiver actually measures pseudorange. Pseudorange is equal to the GPS receiver's observed time of arrival of the GPS signal times the speed of light where the observed time of arrival value includes both the signal propagation delay due to the actual geometric range plus the receiver's clock bias. Clock bias is the amount of adjusting needed to synchronize the receiver's clock with the clocks in the GPS satellites.

The control segment consists of a group of unmanned monitor stations that track each satellite as it orbits the earth. This is necessary since the satellites tend to lose track of their position and time. The information from the monitor stations is used to determine the satellite's orbit and the errors in the satellite's onboard atomic clocks. The resulting corrections of ephemeris coordinates and clock corrections are then broadcast to the satellites once per day. This information is then included in the satellite transmission signal to the users.

Each GPS satellite in the constellation continuously broadcasts its own unique pseudorandom noise (PRN) sequences on two L-band frequencies, L_1 and L_2, using a form of digital phase modulation (PM) radio. For PM radio, the changes in carrier wave phase are used to communicate whether a one or zero is being transmitted, as opposed to carrier wave frequency or amplitude. The L_1 frequency operates at 1575.42 MHz while the L_2 frequency operates at 1227.6 MHz. Two types of PRN codes are used to modulate the carrier waves.

The Precise-code, or P-code, is for military users only. Phase shift keying modulation of the carrier wave by the P-code spreads the power out over a 20.46 MHz bandwidth centered at the 1575.42 MHZ and 1227.6 MHz carrier frequencies. The Coarse Acquisition, or C/A-code, is for civilian users. Phase shift keying modulation of the carrier wave by the C/A-code spreads the power out over a 2.046 MHz bandwidth centered at the 1575.42 MHz L_1 carrier frequency. The P-code sequences are about 6.2 trillion bits long and repeat once every week. The C/A-code is 1023 bits long and repeats every millisecond. The short length of the C/A-code allows for quick acquisition and aids the receiver in handing over to the much longer P-code sequences. The C/A-code has a chipping rate of 1 million bits per second, while P-code has a chipping rate of 10 million bits per second. A chip is defined as the binary transition from 0 to 1, or vice versa. The higher chipping rate of the P-code causes it to be more accurate than C/A-code. It was initially thought that since the P-code chipping rate is ten times faster than the C/A-code chipping rate, P-code would provide ten times more position accuracy. However, in practice this was not the case. P-code provides a position accuracy of 15 meters CEP while C/A-code provides a position accuracy of 40 meters CEP (2:65).

Errors associated with the Space, Control, and User segments result in GPS position and velocity navigation errors. These errors are related to measurements made along the user-to-satellite lines of sight and the relative geometry of the user and tracked satellites. A summary of user-to-satellite lines of sight GPS errors is contained in Table 2.

GPS SEGMENT	ERROR SOURCE	P-CODE PSEUDORANGE ERROR 1σ (m)	C/A-CODE PSEUDORANGE ERROR 1σ (m)
SPACE	Clock & Navigation Subsystem Stability	3.00	3.00
	Predictability of Satellite Perturbations	1.00	1.00
	Other	0.50	0.50
CONTROL	Ephemeris Prediction Model Implementation	4.20	4.20
	Other	0.90	0.90
USER	Ionospheric Delay Compensation	2.30	5.0 - 10.0
	Tropospheric Delay Compensation	2.00	2.00
	Receiver Noise & Resolution	1.50	7.50
	Multipath	1.20	1.20
	Other	0.50	0.50
1σ SYSTEM ERROR (m)	Total	6.60	10.8 - 13.9

Table 2

GPS Error Budget

The Control and Space Segment errors listed in Table 2 are primarily related to the stability and predictability of satellite clock and ephemeris behavior, while the User Segment errors are related to propagation time delays. Propagation time delays occur because the satellite broadcast signals are bent and slowed down as they pass through the earth's ionosphere and troposphere. The time delay due to the ionosphere is inversely proportional to the square of the transmission frequency. Tropospheric errors occur in the near-Earth atmosphere where ionized gases create an atmospheric index of refraction greater than one. These errors are independent of frequency but are dependent on the user altitude and user-to-satellite elevation angle. The GPS receiver contains a tropospheric software model for compensating for tropospheric delays.

Errors relating to the relative geometry of the satellites used by the receiver are called "dilution of precision" or DOP. A good DOP is a low number meaning the satellites exhibit good geometry as seen by the receiver. The optimal geometry is to have one satellite directly overhead with the other three equally spaced 120 degrees apart on the horizon. There are special types of DOPs for each of the position and time solution dimensions. Vertical DOP (VDOP) is used to describe the satellite geometry effect on receiver altitude errors. Horizontal DOP (HDOP) describes the satellite geometry effect on receiver latitude and longitude errors. The time dimension effect of geometry is given by the time DOP (TDOP) value. Position DOP (PDOP) is the combination of HDOP and VDOP, yielding a three dimensional error value. Geometric DOP (GDOP) is the combination of PDOP and TDOP, yielding a four dimensional error value.

As evidenced in the preceeding paragraphs, GPS is subject to several sources of errors that degrade navigational accuracy. Nonetheless, GPS provides an accuracy that is better than most radionavigation systems and is sufficient for most applications. However, there are applications, such as precision guided munitions, where GPS accuracy is not adequate. Therefore, a method of improving GPS accuracy is needed. One method is known as differential GPS. For differential GPS two receivers continuously exchange navigation information with one another in real time. One receiver acts as a base station, and the other receiver is usually mobile and navigates relative to the base station's location. The base station receiver knows its pre-surveyed location. It computes a range solution to the satellite by measuring the signal travel time multiplied by the speed of light. Next, the base station receiver computes a second range solution by subtracting the satellite position, computed from the ephemeris constants, from its pre-surveyed location. It then differences the two range solutions and sends this pseudorange correction in real time to the mobile receiver. Differential GPS improves accuracy because many of the errors common to both receivers, such as clock-bias, satellite ephemeris, and ionospheric and tropospheric delays, are eliminated. Positioning errors as small as three feet can be achieved using differential GPS (6:80).

One of the key factors for enabling the GPS receiver to calculate its position is the satellite navigation message. This message is a 50 bit per second data stream that is superimposed on the C/A-code and P-code pulse trains transmitted by the satellites. This data stream is divided into 30-second frames, each of which contains 1500 bits of information. Each frame is subdivided into five 6-second subframes, each containing 300 bits. Subframe 1 contains the satellite clock bias parameters which precisely give its specific offset from the GPS master clock along with terms that describe the accuracy and health of the satellite's signal. Subframes 2 and 3 provide high precision ephemeris parameters which allow accurate computation of the satellite's location at each point in time. Subframes 1, 2, and 3 are required by the receiver before it starts using the satellite for position fixing. Subframes 4 and 5 contain low-precision clock and ephemeris ("almanac") data for every satellite in the constellation. They also contain information such as health and configuration status for all satellites, user text messages, and parameters describing the offset betwen GPS time and Universal Coordinated Time. It takes 12.5 minutes to collect an entire set of subframe 4/5 data. Each satellite transmits the same subframe 4/5 data as all other satellites.

Selective Availability (SA) and Anti-Spoofing (A-S) are features available on GPS. SA allows the intentional introduction of errors into the GPS signal, along with encrypted correction parameters which allow certain users to remove the effects of these errors. SA degrades the navigational accuracy of civilian GPS users. Military users have access to the encryption keys that allow the encrypted correction parameters to be used so that navigational accuracy is not degraded. Anti-Spoofing is the mechanism used in the military environment to foil an adversary's attempt to use a broadcast beacon that mimics the GPS signal to spoof the receiver into tracking the mimic signals instead of the GPS satellite signals. The A-S feature of GPS allows the P-code portion of the GPS signal to be encrypted, known as Y-codes. Since the spoofer can't autonomously generate the Y-codes needed to mimic the GPS signal, spoofing is prevented.

An area of particular importance to the military user is jamming of the GPS signal. Jamming may occur prior to or after acquisition of the GPS satellites. The receiver acquires and tracks the satellites through the use of tracking loops. A receiver executes two different types of tracking loops, code tracking and carrier tracking. For code tracking, the receiver first acquires the GPS satellites by having a channel generate a copy of the desired PRN code, C/A or P, based on the master time pulse from the quartz clock. The acquisition finishes with the channel slewing its replica PRN code forward and backward in time while simultaneously adjusting the nominal slewing rate faster and slower until the replica PRN code exactly matches up with the desired PRN code. This event is known as correlation. The channel then outputs a value for the total amount of slewing required to maintain correlation between the PRN codes. This amount of slewing value is the channel's pseudorange measurement, which is the time difference between the master time pulse and the incoming PRN code time. The second tracking loop is carrier tracking. The primary purpose of the carrier tracking loop is to compare the incoming carrier signal in phase and frequency with the receiver's internally-generated replica of carrier phase and frequency. Phase lock is achieved when the receiver's estimate of carrier phase and frequency matches the incoming carrier phase and frequency to within one radian. After phase lock the receiver can obtain its instantaneous Doppler shift. This allows the receiver to measure the corresponding delta range so it can estimate accurate values for the receiver's three velocity components. Thus, the typical tracking states for an unaided GPS receiver are acquisition/reacquisition, code tracking, carrier tracking, and data demodulation. In order for the receiver to attain these tracking states, a threshold signal-to-noise density ratio (C/N_0) must be achieved.

The following thresholds reflect typical transition stages in unaided receiver tracking states:

- Data Demodulation - $\geq$ 27 dB-Hz
- Carrier Tracking - $\geq$ 25 dB-Hz
- Code Tracking - $\geq$ 15 dB-Hz
- Acq./Reacquisition - $\geq$ 15 dB-Hz

As C/N_0 increases, so does the quality level of tracking state. The reverse of this progression occurs when the C/N_0 is attenuated due to jamming. A straight-forward relationship between C/N_0 and Jammer-to-Signal (J/S) ratio is shown in Equation 26.

$$C/N_0 + J/S = 70 \qquad (26)$$

As the J/S level increases, the C/N_0 level decreases according to this equation. Given increasing jammer power, the receiver will orderly lose data demodulation, carrier tracking, and finally code tracking.

The jamming immunity or resistance of a GPS receiver is determined by several factors. First, the spreading of the transmitted waveform due to the modulation by the PRN code increases jamming immunity. The more spreading achieved, the more resistant the signal will be to jamming. The P-code spreads the power of the transmitted signal out over a 20.46 MHz bandwidth, while the C/A-code spreads the signal out over a 2.046 MHz bandwidth. Since the P-code bandwidth is ten times that of the C/A-code, the antijamming performance of the P-code signal is 10 dB better than the C/A-code signal. A P-code receiver is able to maintain track on the satellites in the presence of jammer signals 41 dB greater than the satellite signals, while a C/A-code receiver can maintain track on the satellites in the presence of jammer signals 31 dB greater than the satellite signals.

The type of antenna used by the GPS receiver has an effect on jamming immunity. A Controlled Reception Pattern Antenna (CRPA) has gain patterns that are controllable in real time. CRPAs include a specialized antenna electronics unit to create a null and then steer it in a pseudorandom fashion across the nominal gain pattern to minimize the total measured power in either the L_1 or L_2 frequency band. This scheme results in the null being oriented directly towards the jammer because the jammer is a point source of radio frequency energy operating at higher power than the sky's background noise, and orienting the null in its direction reduces the total received power.

Finally, inertial aiding of the GPS receiver can improve jamming immunity. Many GPS receivers allow an external source input such as host vehicle acceleration, velocity, and attitude measurements from an IMU. This information helps the tracking loops avoid the jerk problem that degrades the position, velocity, time accuracy when operating stand-alone in a high-dynamic environment. In other words, the vehicular dynamics are effectively removed as the major component of tracking loop uncertainty, thus allowing the tracking loops bandwidths to be reduced. Reduced code and carrier tracking loop bandwidths directly reduce noise density (N_0) leading to extension of the minimum C/N_0 tracking thresholds, thereby adding jamming immunity.

SECTION V

KALMAN FILTERING

In order to analyze a physical system, such as a missile, an inertial navigation system, or a chemical process, the engineer must develop a mathematical model that describes the behavior of that system. This system dynamics model is developed from the laws of physics and/or empirical testing. No matter how detailed the engineer makes this model, there will be some imperfections in it when compared to the actual physical system. In addition, the physical system may be driven by disturbances which can not be controlled nor modelled by the engineer. Thus, these two forms of uncertainty are classified as noises in the system. Sensors which provide measurements of the states of a dynamic system, such as accelerometers and gyros, contain noise. This noise corrupts the measurement readings, causing measurement accuracy to degrade. A means of accounting for the uncertainty, or noise, in a dynamic model is through the use of a Kalman Filter. A Kalman Filter is an algorithm for computing an optimal estimate of the state of a dynamic system using all available measurements and initial condition information. In other words, the Kalman Filter provides the optimal estimate of the state based on measurements. This optimal estimate minimizes the error statistically. The assumptions made for a Kalman Filter are that the system dynamics model is linear, the process and measurement noises are white and have Gaussian probability densities. For the tactical missile application, a Kalman Filter is used for estimating the errors associated with GPS and the inertial navigation system.

Design of the Kalman Filter begins with the system dynamics model. The system dynamics model and the measurement model can be continuous or discrete, as can the Kalman Filter. Typically, the sensor measurements are discrete, hence a discrete measurement model is used. In addition, the discrete Kalman Filter is used in most applications regardless of whether the system model is continuous or discrete. Therefore, if the dynamics model is continuous, an equivalent discrete-time model is developed and the discrete-time Kalman Filter is generated from this model. A continuous dynamics model is shown in Equation 27 and a discrete finite difference dynamics model is shown in Equation 28.

$$\dot{\mathbf{x}}(t) = \mathbf{F}(t)\mathbf{x}(t) + \mathbf{B}(t)\mathbf{u}(t) + \mathbf{G}(t)\mathbf{w}(t) \qquad (27)$$

$\mathbf{x}(t)$ = State vector

$\mathbf{F}(t)$ = Plant, or system matrix

$\mathbf{B}(t)$ = Input matrix

$\mathbf{u}(t)$ = Inputs

$\mathbf{G}(t)$ = Noise matrix

$\mathbf{w}(t)$ = White noise inputs

$$\mathbf{x}(t_{i+1}) = \underline{\phi}(t_{i+1}, t_i)\mathbf{x}(t_i) + \mathbf{B}(t_i)\mathbf{u}(t_i) + \mathbf{G}(t_i)\mathbf{w}(t_i) \qquad (28)$$

$\underline{\phi}(t_{i+1}, t_i)$ = State transition matrix

$\mathbf{x}(t_i)$ = State vector

$\mathbf{B}(t_i)$ = Input matrix

$\mathbf{u}(t_i)$ = Inputs

$\mathbf{G}(t_i)$ = Noise matrix

$\mathbf{w}(t_i)$ = White noise inputs

The model defining the available measurements is shown in Equation 29.

$$\mathbf{z}(t_i) = \mathbf{H}(t_i)\mathbf{x}(t_i) + \mathbf{v}(t_i) \qquad (29)$$

$\mathbf{z}(t_i)$ = Available measurements

$\mathbf{H}(t_i)$ = Measurement matrix

$\mathbf{x}(t_i)$ = State vector

$\mathbf{v}(t_i)$ = Measurement noise

Given the system model described in Equations 27-29, the Kalman Filter propagates the optimal state estimate beginning at measurement time t_{i-1} until a measurement update occurs at time t_i through the relations shown in Equations 30-31.

$$\hat{\mathbf{x}}(t_i^-) = \underline{\phi}(t_i, t_{i-1})\mathbf{x}(t_{i-1}^+) + \int_{t_{i-1}}^{t_i} \underline{\phi}(t_i, \tau)\mathbf{B}(\tau)\mathbf{u}(\tau)\, d\tau \qquad (30)$$

$$\mathbf{P}(t_i^-) = \underline{\phi}(t_i, t_{i-1})\mathbf{P}(t_{i-1}^+)\underline{\phi}^T(t_i, t_{i-1}) + \int_{t_{i-1}}^{t_i} \underline{\phi}(t_i, \tau)\mathbf{G}(\tau)\mathbf{Q}(\tau)\mathbf{G}^T \underline{\phi}^T(t_i, \tau)\, d\tau \quad (31)$$

In equation 30, $\hat{\mathbf{x}}(t_i^-)$ represents the optimal estimate of the state before a measurement is taken. In equation 31, $\mathbf{P}(t_i^-)$ represents the covariance of the states, or the uncertainty in the states before a measurement is taken. At measurement time t_i, the measurement $\mathbf{z}_i$ becomes available. The estimate is updated through the Kalman gain equation shown in Equation 32. This gain is employed in both the mean and covariance relations shown in Equations 33 and 34.

$$\mathbf{K}(t_i) = \mathbf{P}(t_i^-)\mathbf{H}^T(t_i)[\mathbf{H}(t_i)\mathbf{P}(t_i^-)\mathbf{H}^T(t_i) + \mathbf{R}(t_i)]^{-1} \qquad (32)$$

$$\hat{\mathbf{x}}(t_i^+) = \hat{\mathbf{x}}(t_i^-) + \mathbf{K}(t_i)[\mathbf{z}_i - \mathbf{H}(t_i)\,\hat{\mathbf{x}}(t_i^-)] \qquad (33)$$

$$\mathbf{P}(t_i^+) = \mathbf{P}(t_i^-) - \mathbf{K}(t_i)\mathbf{H}(t_i)\mathbf{P}(t_i^-) \qquad (34)$$

$\hat{\mathbf{x}}(t_i^-)$ = Best prediction of $\mathbf{x}(t_i)$ before measurement at time t_i.

$\mathbf{P}(t_i^-)$ = Uncertainty in states before measurement at time t_i

$\mathbf{r}(t_i) = \mathbf{z}_i - \mathbf{H}(t_i)\,\hat{\mathbf{x}}(t_i^-)$ = Measurement residual
$\mathbf{K}(t_i)$ = Optimal Gain matrix to weight the residual in order to add a correction term to $\hat{\mathbf{x}}(t_i^-)$ to get $\hat{\mathbf{x}}(t_i^+)$

$\mathbf{H}(t_i)\,\hat{\mathbf{x}}(t_i^-)$ = Best prediction of measurement at time t_i before it is actually taken

$\mathbf{z}_i$ = Realized value of measurement at time t_i

$\mathbf{Q}(t)$ = Strength of process noise

$\mathbf{R}(t_i)$ = Strength of measurement noise

The initial conditions are given by Equations 35 and 36.

$$\hat{x}(t_0) = E\{x(t_0)\} = \hat{x}_0 \qquad (35)$$

$$P(t_0) = E\{[x(t_0) - \hat{x}_0][x(t_0) - \hat{x}_0]^T\} = P_0 \qquad (36)$$

The uncertainty in the system dynamics model is represented by the vectors $\hat{x}_0$, P_0, Q, and R. The vector $\hat{x}_0$ represents the initial guess of the values of the system states. In many applications this value is equal to zero. The matrix P_0 represents the covariance of the states. Ideally, the system states will be independent of one another. This makes P_0 a diagonal matrix since all off diagonal terms are zero. The matrix Q represents the strength of the process noise in the system. It can also represent the amount of uncertainty in the system model due to inadequate modelling. The higher the value of Q, the more noise or uncertainty there is in the system model. The vector R represents the amount of noise in the measurements, or sensor. The higher the value of R, the more uncertainty there is in the sensor measurements. In all, the system model structure is determined by $F(t)$ or $\Phi(t,\tau)$, $B(t)$, $G(t)$, and $H(t_i)$ for all times of interest. The uncertainties in the model are specified by $\hat{x}_0$, P_0, $Q(t)$, and $R(t_i)$.

As an example, the Kalman Filter could be used for estimating the errors with the INS. Some of these errors, such as gyro bias and scale factor were discussed in Section II. An INS dynamics error model would be derived and put in the form of Equation 27 or 28. If the INS were unaided, the measurements for the measurement model in Equation 29 would come from the INS. Typically, these measurements are position and velocity. If the INS were aided with a GPS receiver, position and velocity measurements from the INS and GPS would be differenced and used as the measurement inputs. A description of INS/GPS integration is given in Section VI.

SECTION VI

INS/GPS INTEGRATION

An inertial navigation system can provide good inertial navigation information for a certain period of time. An INS provides good high frequency information, but it drifts at a slow rate. Thus, the long term or low frequency content of INS data is poor. All inertial systems have errors that grow slowly with time, and the position errors are unbounded. Most other navigation aids provide good low frequency data but are subject to high frequency noise. Thus, an external source of data, such as GPS or radar position or doppler velocity, could complement the INS by bounding the low frequency errors. A Kalman Filter can be used to integrate the good high frequency data of the INS with the good low frequency data of the external source so that the slowly growing errors inherent in the INS are damped out. For the missile application, the midcourse phase of guidance would integrate the INS with GPS data to produce more accurate calculations of position and velocity.

In an integrated INS/GPS system, each data source provides some benefit to the other. For example, the GPS receiver can provide the INS with accurate real-time estimates on the current behavior of its error statistics. The INS can aid the GPS receiver tracking loops with accurate initial estimates of position and velocity, thereby reducing the time needed to initially acquire the GPS satellites. If the receiver loses lock on the satellites, this same position and velocity aiding from the INS can help the GPS receiver reacquire the satellites in a timely manner.

There are two main methods of integrating the INS with the GPS. The first method is known as tightly coupling. When tightly coupling the INS with the GPS a master navigation Kalman Filter in the INS is used to assimilate INS and GPS data to produce a system estimate of the state vector. The GPS receiver has its own Kalman Filter and outputs its pseudorange and delta range calculations to the master Kalman Filter.

In some implementations the navigation algorithms are omitted from the INS and reside in the GPS receiver, or they are retained in the INS to serve as a backup mode of navigation in case of GPS receiver failure. The GPS Kalman Filter usually allows external velocity inputs from the INS to inertially aid the tracking loops. This allows the tracking channels to hold on to the signals better without risk of losing lock due to unanticipated accelerations. A block diagram illustrating a tightly coupled system is shown in Figure 4.

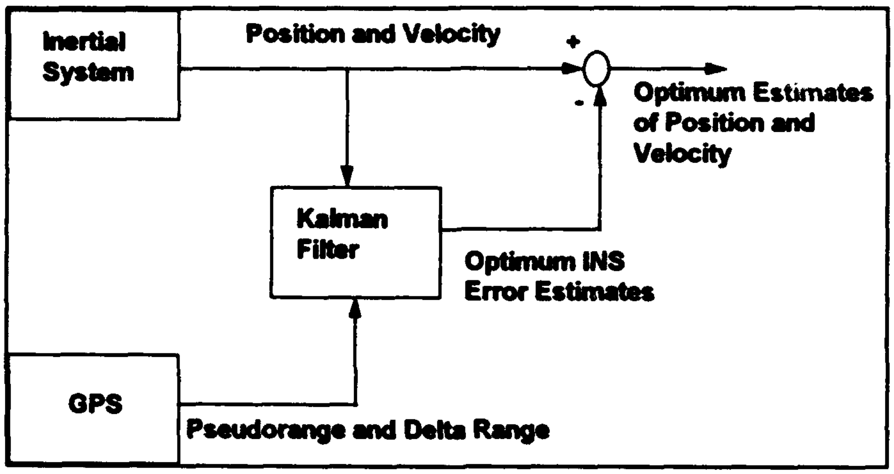

Figure 4

Tightly Coupled INS/GPS

The other type of INS/GPS integration is known as loosely coupled. In this case, the predicted and/or estimated state vector from the internal GPS Kalman Filter is passed to the master Kalman Filter. Essentially, the GPS receiver operates independently of the INS. The navigation state outputs of the GPS receiver are made available to reset the INS navigation algorithms. Typically, velocity aiding is not used for the loosely coupled case. A block diagram of a loosely coupled system is shown in Figure 5.

The advantages of implementing a tightly coupled system are first that updates can be accomplished with one or more satellites through the use of inertial aiding. This improves jamming resistance since three satellites can be jammed, yet a position, velocity, and time update can still be generated with the remaining satellite. Tightly coupling the INS with GPS allows a fast update rate of one to ten seconds, and a velocity accuracy of 0.1 ft/sec.

The advantages of a loosely coupled system are that there is redundancy and simpler integration. The disadvantages of a loosely coupled system are that updates require four satellites, yielding less jamming immunity. Also, an update rate of about thirty seconds is required, and the velocity accuracy is only 0.3 ft/sec.

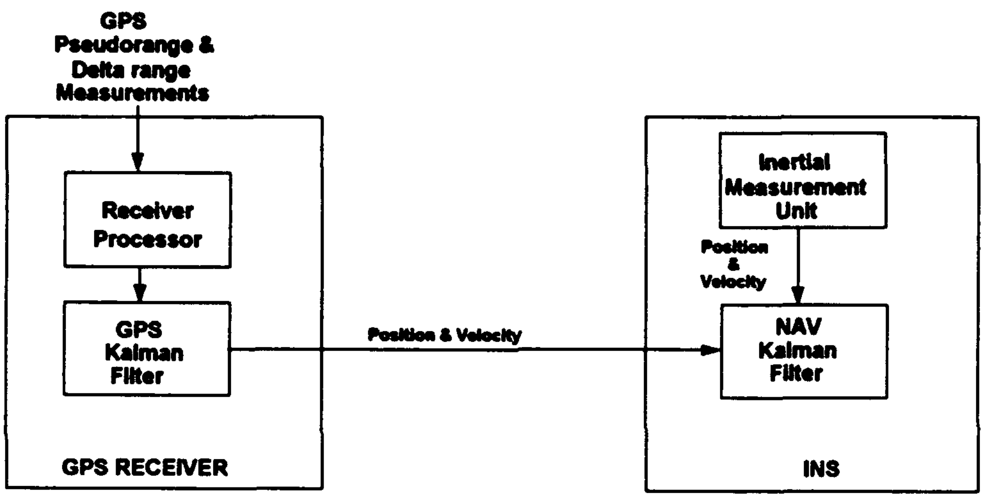

Figure 5

Loosely Coupled INS/GPS

In the terminal phase of guidance, the INS would most likely be integrated with data from a seeker, such as radar range or range rate data. A Kalman Filter would be used in a similar manner to integrate the seeker data with the INS data. A discussion of seekers is presented in the next section.

SECTION VII

SEEKERS

A seeker is a system that processes information from a sensor for use by the missile guidance system. The sensor receives reflected radio frequency (RF) energy or radiant infrared (IR) energy from the target. This energy from the target is used by the guidance system to survey, acquire, and to lock-on and track the target. Typically, seekers are used during terminal guidance to track the missile's target of interest. However, seekers can also be used to aid the missile's INS during midcourse guidance with position and velocity updates. As shown in Figure 6, the seeker is comprised of several major components. An energy-collecting system is the first component. This consists of an antenna if the energy is RF or optics consisting of lenses or mirrors if the energy is in the infrared, visible, or ultraviolet spectral regions. The next component is a stable platform and its associated control system for stabilizing the seeker. Last, the sensor component converts the received energy into a more usable form, and a signal processing system produces a signal to point the antenna or mirror at the target.

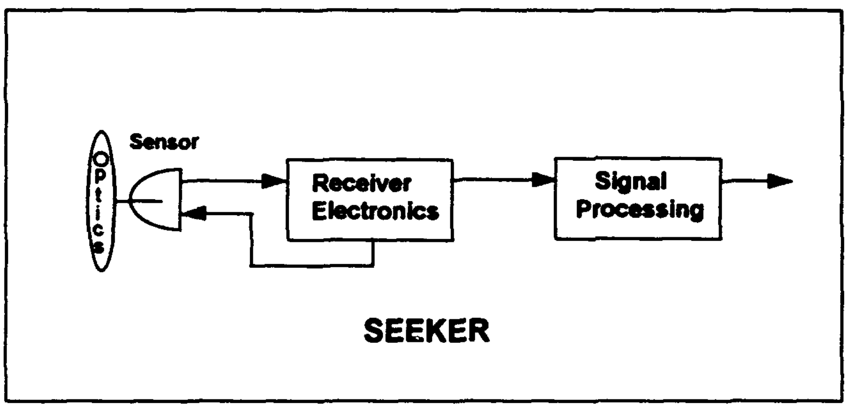

Figure 6

Seeker Block Diagram

Seekers can be active or passive. A passive system detects energy emitted by the target itself for tracking or detects energy originating from natural sources, such as the sun, that is reflected from a target. An active system produces energy which is transmitted to the target where it is reflected back in the direction of the seeker. Thus, an active system must be capable of providing its own energy source. For an active system, the information extracted from the energy returns are range, azimuth angle, and elevation angle of the target along the seeker line-of-sight (LOS). A passive system does not directly provide range information, only angle information. The antenna/optics is usually mounted on a gimbaled platform that provides an inertial reference for making angular measurements. Thus, missile motions due to flight instabilities are removed and the seeker only has to measure the slowly changing angle between the missile and target due to closing motion and intentional maneuvers.

For an active system, the seeker electronics determines range by measuring the time it takes for a radar or ladar pulse to be transmitted and received. Range to the target can then be calculated by Equation 37.

$$R = c\frac{\tau}{2} \qquad (37)$$

where

R = Range
c = Speed of light
t = Two way propagation time

The range performance of the seeker can be determined through use of the ladar range equation for an electro-optical seeker, or through the radar range equation for an RF seeker. The laser radar range equation is developed in Equations 38 - 39. The power received by the laser is shown in Equation 38. This equation assumes that no absorbed power is reradiated and no power scattered out is scattered back in.

$$P_R = \frac{P_T\, G\, A_E\, \sigma T_{os}^2\, e^{-2\alpha R}}{(4\pi)^2 R^4}\,\text{Watts} \qquad (38)$$

where

P_R = Power received
P_T = Laser peak output power
G = Optics antenna gain
A_E = Effective aperture
σ = Laser radar cross section
T_{os} = One way losses within laser radar
α = Atmospheric attenuation coefficient
R = Range to target

The signal-to-noise ratio (SNR) is the ratio of the power received to the noise power contained in the receiver and the atmosphere. The SNR shown in Equation 39 assumes the laser radar receiver uses direct detection and is signal photon noise limited.

$$\frac{S}{N} = \frac{P_T\, G\, A_E\, \sigma T_{os}^2\, e^{-2\alpha R}}{(4\pi)^2 R^4 \left[\dfrac{2\, h\, \nu B}{\eta}\right]} \qquad (39)$$

where

h = Planck's Constant = $6.626\text{x}10^{-34}$ J · s
ν = Photon frequency
B = Noise bandwidth
η = Quantum efficiency

The SNR determines the probability of detecting the target. As shown in Figure 7, the higher the SNR, the higher the probability of detection.

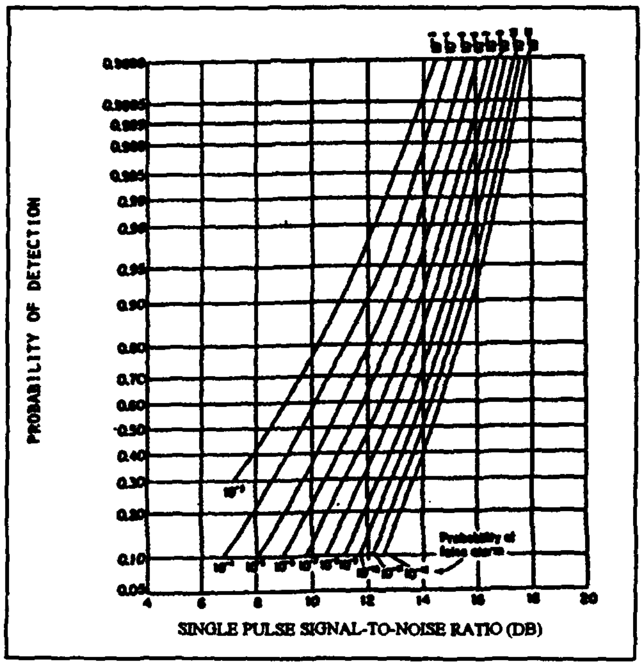

Figure 7
Signal Detection Probability

The radar range equation shown in Equation 40 is similar to the laser radar range equation.

$$P_R = \frac{P_T G^2 \sigma \lambda^2}{(4\pi)^3 R^4} \quad (40)$$

where

P_R = Power received
P_T = Power transmitted
G = Antenna gain
λ = Transmit wavelength
σ = Radar cross section
R = Range

The signal-to-noise ratio is defined in Equation 41.

$$\frac{S}{N} = \frac{P_T G^2 \sigma \lambda^2}{(4\pi)^3 R^4 k T_0 B_n} \qquad (41)$$

where

k = Boltzmann's Constant = 1.38 x 10^{-23} J/deg
B = Receiver Noise Bandwidth
T_0 = Ideal receiver noise temperature = 290 deg K

Electro-optical and RF seekers operate at different wavelengths. Ladar, an electro-optical system, is an active system that operates at wavelengths ranging from below 1mm to over 10mm. Millimeter wave RF seekers operate at frequencies in the 30 GHz - 300 GHz range which is equivalent to a wavelength region of 1 mm - 10 mm. In general, seeker resolution is better at shorter wavelengths since the resolution is determined by the ratio of the seeker wavelength to the seeker aperture diameter (λ/D).

Weather has different effects on these various wavelength regions. This must be taken into account when considering a seeker for an autonomous weapon application. The lower the transmission frequency, the better the weather performance. Haze, fog, and clouds affect ladar more severely than millimeter wave radar. A millimeter wave seeker's performance is sensitive to rain rate and raindrop size distribution. To achieve an all weather seeker, "low" frequencies (10-18 GHz) for a fixed aperture size must be used.

Synthetic aperture radar (SAR) is a radar concept that often, but not always, operates in the millimeter wave region. SAR takes advantage of the forward motion of the radar to produce the equivalent of a long antenna. As shown in Figure 8, each time a radar pulse is transmitted, the radar occupies a position a little further along the flight path. Since distances from successive array elements to a distant point on the boresight line are equal, the returns from the point add up in phase. Therefore, by pointing a reasonably small antenna out to one side and summing the returns from successive pulses, it is possible to synthesize a very long sidelooking linear array.

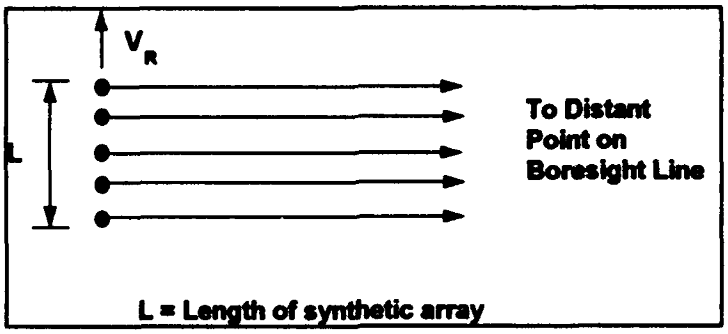

Figure 8

SAR Concept

The length of the array must be short enough in relation to the range to the swath being mapped such that the lines of sight from any one point at the swath's range to the individual array elements are essentially parallel. In an unfocused array the array length is an appreciable fraction of the swath's range. Thus, the lines of sight will not be parallel and will have different distances to the elements, yielding differences in the phases of the returns. Equations 42 and 43 show the maximum effective array length, L_{eff}, and the azimuth resolution distance, d_a, respectively, for an unfocused array.

$$L_{eff} = 1.2\sqrt{\lambda R} \quad (42)$$

λ = Wavelength
R = Range

$$d_a \cong 0.4 L_{eff} \quad (43)$$

By focusing the array, the length of the array can be greatly increased. An array can be focused by applying an appropriate phase correction to the returns received by each array element. This phase correction is shown in Equation 44.

$$\text{Phase Correction} = \frac{-2\pi}{\lambda R} d_n^2 \quad (44)$$

d_n = Distance of element n from array center

For a focused array, the maximum length of the array and the minimum resolution distance are shown in equations 45 and 46, respectively.

$$L_{max} = \frac{\lambda}{l} R \quad (45)$$

l = Diameter of real antenna

$$d_a = \frac{\lambda}{2L} R \quad (46)$$

Whether the seeker is RF or EO, the received energy is sent to a signal/image processor which processes the seeker output and sends it through a target detection and classification algorithm. The target detection algorithm detects the target in the received energy and determines the target aimpoint. The image processor outputs parameters such as range or range rate of the target aimpoint and sends them to a guidance processor. The guidance processor contains a terminal track Kalman Filter which combines the target aimpoint measurements with inertial measurements. The output of the filter is the relative position of the missile and target aimpoint. This target aimpoint and the present missile position are input into the Kalman Filter. This information is then used by the autopilot for guidance and control of the missile to target impact.

SECTION VIII

GUIDANCE AND CONTROL

Guidance, navigation, and control are terms that are sometimes used interchangeably when discussing navigation or guidance systems. There are, however, differences between the three terms. As discussed in Section II, a navigation system is one that automatically determines the position, velocity, and acceleration of a vehicle with respect to a reference frame such as the earth. A guidance system is one that automatically makes the corrections necessary to keep the vehicle on course by sending the proper signal to the control system or autopilot. The guidance system then performs all the functions of a navigation system in addition to generating the required correction signal to be sent to the control system. Finally, the control system controls the direction of the motion of the vehicle, or the orientation of the velocity vector, based on the input signal from the guidance system.

There are two main phases of guidance for an air-to-ground missile, midcourse and terminal. The midcourse phase begins immediately following transfer alignment from the host aircraft and may last only a matter of seconds for a tactical mission, or several hours for a cruise missile mission. The purpose of the midcourse phase is to guide the missile into a certain acquisition basket so that terminal guidance can begin. This section is concerned with terminal guidance and discusses the most commonly used guidance system for air-to-ground missiles, proportional navigation guidance.

For proportional navigation the missile is guided in one of three ways. The first two ways are through the use of energy transmitted by an active seeker and reflected from the target or the radiant infrared (IR) energy from the target. These two methods require the use of a seeker, as described in Section VII. The third method of guidance uses no seeker, but instead uses only an INS aided with GPS updates of position and velocity. In this case the missile uses the WGS 84 earth coordinates of the target determined from previous reconnaissance data to guide to the target.

Proportional navigation is based upon commanding the missile to turn at a rate proportional to the angular velocity of the line of sight (LOS). The LOS is defined as an imaginary line from the missile to the target. The ratio of the missile turning rate to the angular velocity of the LOS is called the proportional navigation constant, N. The value of N usually ranges from 2 to 6. If the value of N is 1, the missile is turning at the same rate as the LOS, or simply homing on the target. If the value of N is greater than 1, the missile is turning at a rate faster than the LOS, thus building up a lead angle with respect to the LOS. In a system utilizing a seeker, the seeker tracks the target, thereby establishing the direction of the LOS. The angular velocity of the LOS is then output to the guidance system. In a seekerless system, the predetermined target coordinates are differenced with the current coordinates of the missile. The missile is commanded to turn at a rate to null out the angular difference between the missile and target coordinates.

The output from the guidance system goes to the autopilot and may be in the form of a heading or attitude command, a pitching or turning command, or a pitch or yaw acceleration command, depending upon the type of guidance scheme used. The autopilot translates the commands produced by the guidance processor into a form suitable for properly driving the control actuators while maintaining flight stability and integrity.

Autopilots may be open-loop or closed-loop. An open-loop autopilot has no inertial feedback sensors and relies solely on the seeker for missile guidance. Antitank missiles are applications where an open-loop autopilot is used. A closed-loop autopilot uses inertial sensors to form a feedback signal to stabilize the missile airframe in pitch, yaw, and roll. Roll autopilots provide roll position stabilization or roll rate stabilization. Lateral autopilots provide stabilization of pitch and yaw motion by increasing the airframe damping and compensating for aerodynamic instability with changes in the missile's center of gravity as rocket fuel is expended.

There are various types of guided missiles. Missiles such as cruise missiles are banked to turn, meaning they are flown in the same manner as manned aircraft. The autopilots used for bank-to-turn missiles are similar as well to those used for manned aircraft. Missiles such as AMRAAM use aerodynamic lift to control the direction of flight and are known as skid-to-turn missiles. Many skid-to-turn missiles are roll stabilized. A block diagram of a roll stabilization autopilot is shown in Figure 9.

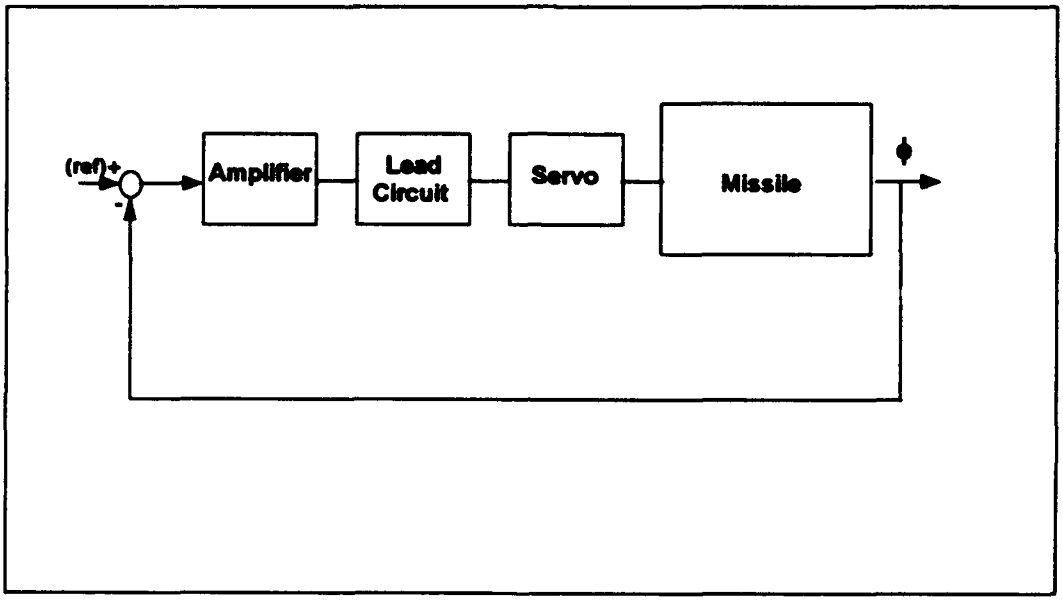

Figure 9

Roll Stabilization Autopilot Block Diagram

Three example missile guidance systems are given in the succeeding section. The operational versions of these systems would employ bank-to-turn autopilots. However, detailed autopilot designs are not provided since two of the programs resulted in a captive flight test and, hence, free-flight autopilot control is not needed.

SECTION IX

MISSILE GUIDANCE SYSTEM EXAMPLES

Sections I - VII have served as a tutorial providing background to equip the reader with some basics for at least a top-level understanding of guidance and navigation systems. Building on this background, this section provides examples of three Air Force munition guidance system programs. These programs are the Advanced Technology Ladar System (ATLAS), Autonomous Synthetic Aperture Radar Guidance (ASARG), and the Operational Concept Demonstration (OCD). ATLAS and ASARG are 6.3 Laboratory research and development programs which resulted in captive flight testing. Thus, some the design implementations which will be described are for a captive flight test and would not be implemented in an operational system. The OCD program, however, resulted in a free flight test. The design implementation for the OCD test was more representative of an operational system.

A. ATLAS

The ATLAS program investigates the feasiblity of using a carbon dioxide (CO_2) laser radar as a seeker for a cruise missile application. The operational scenario for this application begins on the ground with mission planning. During mission planning an operational flight program is generated with the waypoints for midcourse guidance and the earth coordinates of the ground target. This program is then downloaded to the missile guidance computer. The missile is carried on a fighter aircraft as ingress to the launch location occurs. Before launch, a transfer alignment between the host aircraft INS and the missile INS occurs. After the missile INS has been aligned, the missile is launched, and the fighter aircraft returns to base. The midcourse phase of guidance begins once the missile has been launched. The missile flies through the preprogrammed waypoints using the INS with GPS position and velocity updates. When the missile is approximately 2 kilometers away from the preprogrammed target location, the GPS updates are suspended and the CO_2 laser radar seeker is activated to begin to search for the target. After target acquisition, the seeker tracks the target until the missile is about 300 meters from the target location. At this time the seeker is turned off and the missile flies inertially to target impact. This operational mission scenario is illustrated in Figure 10.

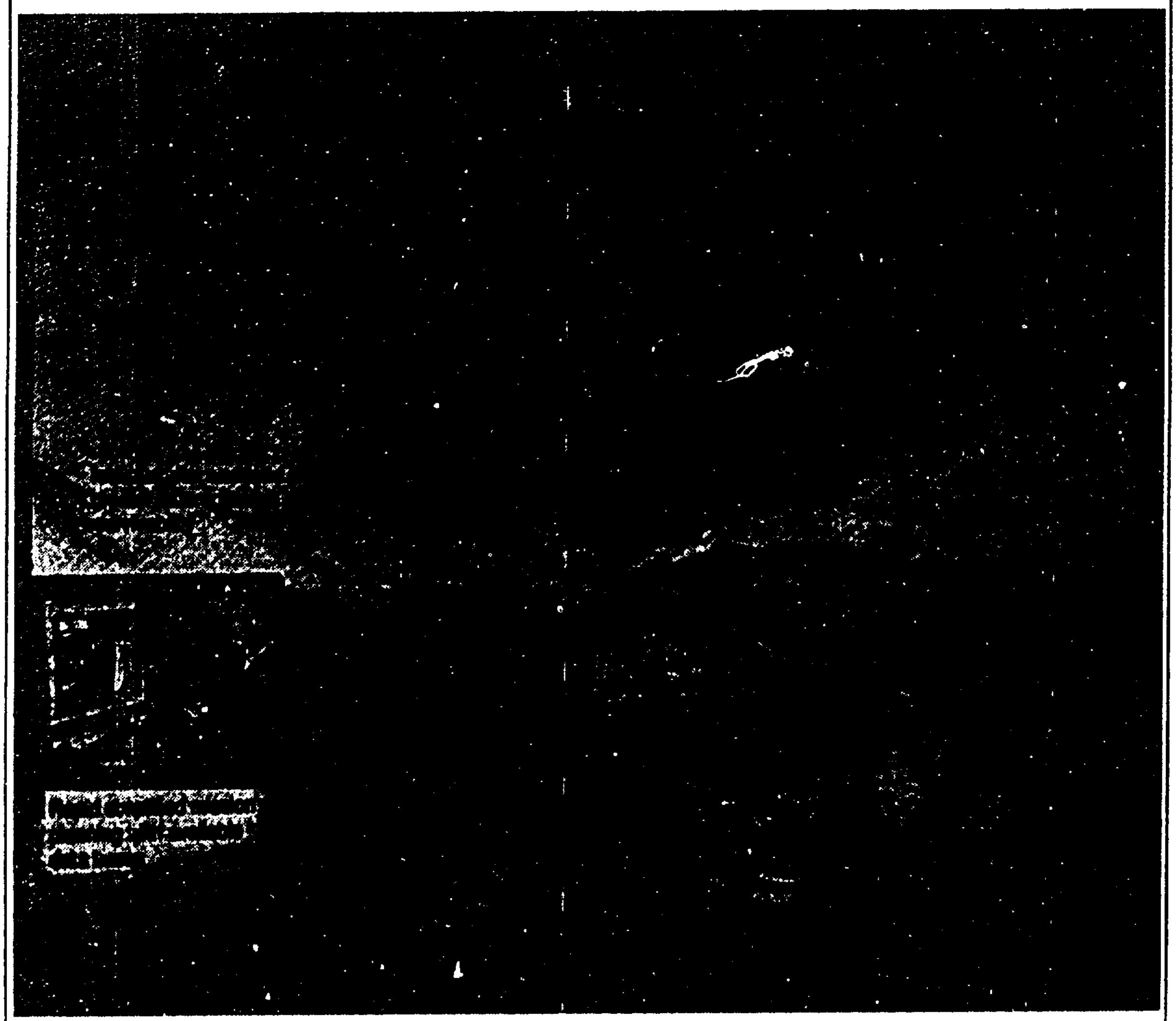

Figure 10

ATLAS Operational Mission Scenario

Development of the ATLAS seeker required a new guidance, navigation, and control system to interface with this seeker. This guidance system also includes a sophisticated algorithm for target detection and classification. An overall guidance system block diagram is shown in Figure 11.

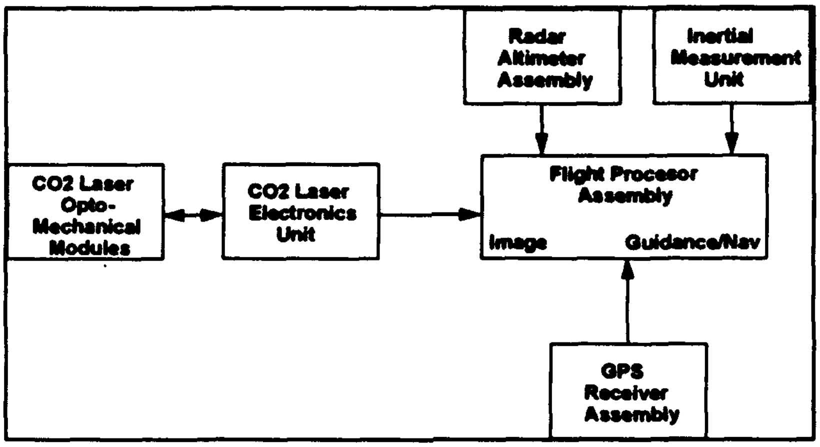

Figure 11

ATLAS Guidance System Block Diagram

The heart of the guidance system is the Flight Processor Assembly (FPA) which includes the guidance and navigation system processor and the image processor. The FPA takes inputs from the CO_2 Ladar seeker, the GPS receiver, the radar altimeter, and the inertial measurement unit to perform the guidance and navigation functions.

Before the missile is launched from the aircraft, a 22-state transfer alignment Kalman Filter is used to initialize and align the missile INS. These 22 states consist of 3 components of position error, 3 components of velocity error, 3 components of tilt error, 1 true vertical misalignment, 3 components of gyro bias, 3 components of accelerometer bias, 3 components of gyro scale factor, 2 components of accelerometer scale factor, and 1 reference altitude error. After the missile INS has been aligned, midcourse guidance begins. The same 22-state Kalman Filter used for transfer alignment is also used for midcourse navigation. The INS consists of 3 single-axis Litton Iron gyros with a 1 degree per hour drift rate and 3 single-axis accelerometers. The coordinate frames used by the INS are the seeker frame, the body frame, the local level frame, the navigation frame, the earth centered earth fixed frame, and the local tangent plane frame. Direction cosine matrices are used for the required coordinate frame transformations, and quaternions are used for calculating the Euler angles. The navigation equations are processed at 100 Hz. The GPS receiver is a Magnovox 5-channel P-code receiver that is loosely coupled with the INS. The GPS receiver provides position updates at 0.016 Hz and velocity updates at 0.1 Hz. The INS provides inertial velocity aiding to the GPS receiver at a 5 Hz rate. Once the missile enters the terminal area, about 2 km from the target, terminal guidance begins. The GPS updates are suspended and the seeker is activated. A block diagram illustrating the terminal guidance interfaces is shown in Figure 12.

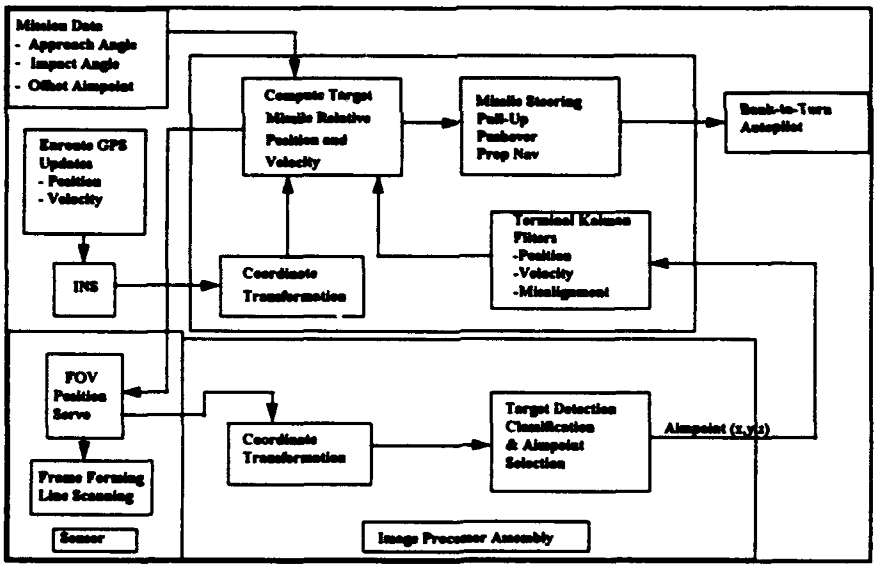

Figure 12

ATLAS Terminal Guidance Block Diagram

The seeker sends out pulses of energy and forms an image frame from the energy returns. The seeker frame rate is 1.5 Hz nominal. This frame is sent to the image processor assembly which coordinate transforms this image from the seeker frame to the local tangent plane frame. The target detection and classification algorithm then determines if a candidate target is contained within the coordinate transformed image. If a candidate target is present in the image, the algorithm selects the three dimensional target aimpoint for the missile to impact. This aimpoint is transferred to one of ten 7-state terminal Kalman Filters whose output is the optimum estimate of the target aimpoint. The 7 filter states consist of 3 components of position error, 3 components of velocity error, and 1 azimuth misalignment. If this candidate target has a high enough correlation value, the algorithm will consider it to be the target of interest and will only update the one Kalman Filter. However, if the original candidate target's correlation value was too low, the algorithm continues to determine other candidate targets. Thus, after subsequent seeker updates, if other candidate targets are determined by the algorithm, additional terminal Kalman Filters are initialized. If a subsequent candidate target has a high enough correlation value, the algorithm will consider this to be the actual target, and no additional terminal Kalman Filters will be initialized. A maximum of ten candidate targets could simultaneously be tracked by the Kalman Filters. If no candidate target resulted in a high enough correlation value, once the missile is approximately 500 meters from the preprogrammed target location, the algorithm will consider the candidate target with the highest correlation value to be the actual target. Thus, only the terminal Kalman Filter tracking that candidate target will continue to be updated.

Also at this time, the INS sends its current position and velocity calculations to the guidance, navigation, and control assembly. The INS position and velocity data is then coordinate transformed from the navigation frame to the local tangent plane frame. This coordinate transformed data is then differenced with the terminal Kalman Filter target aimpoint estimate to form the relative missile target position and velocity vectors. This information is then used by the proportional navigation guidance law to compute a control command for the autopilot.

B. ASARG

The ASARG program investigates the feasibility of using a synthetic aperture radar as a seeker for a tactical missile application, such as the AGM-130, or a cruise missile application. The operational scenario for this application is very similar to that for ATLAS, as shown in Figure 13.

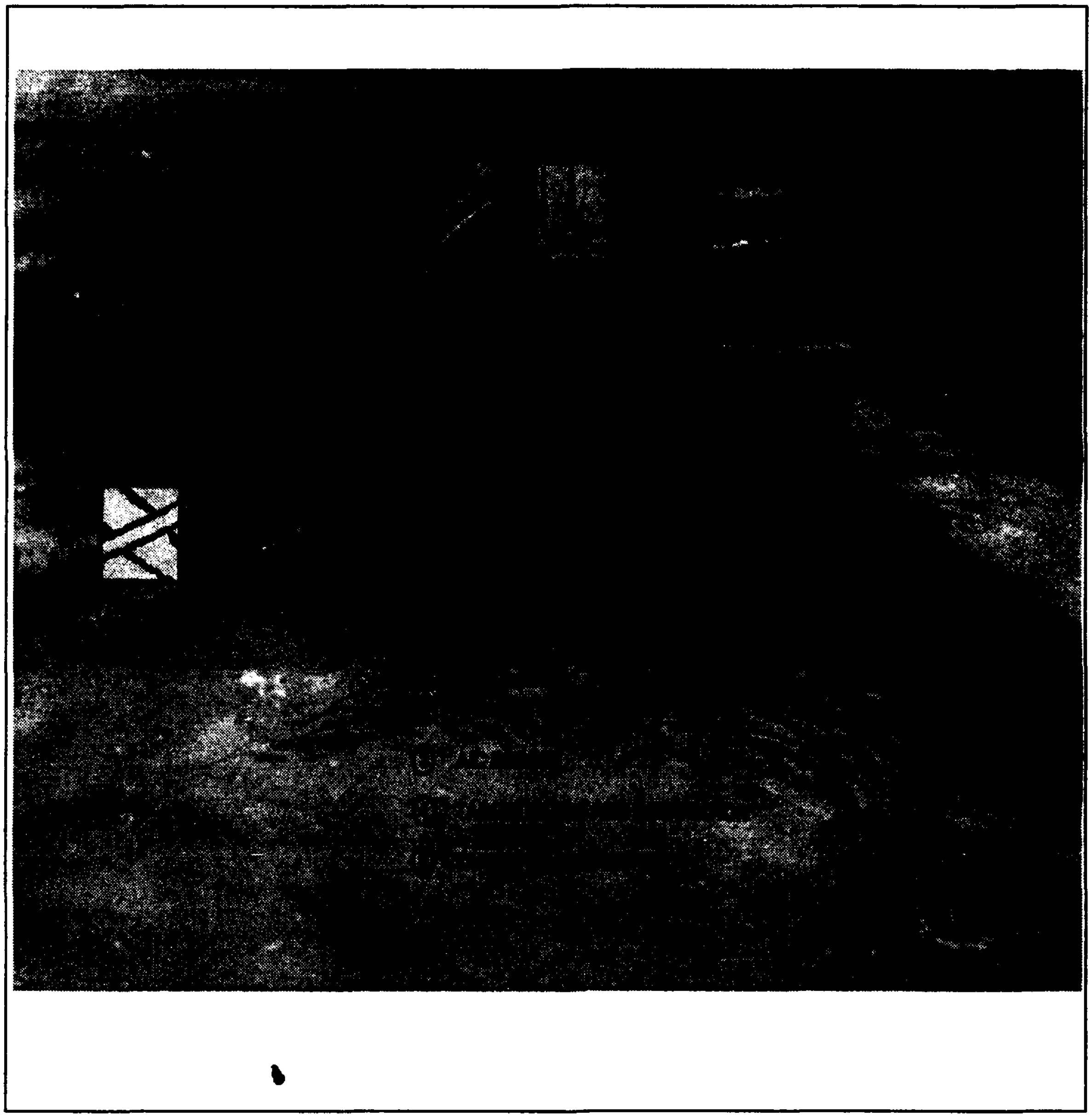

Figure 13

ASARG Operational Scenario

Development of the ASARG seeker required a new guidance, navigation, and control system to interface with the SAR seeker. A system block diagram for the ASARG captive flight test is shown in Figure 14.

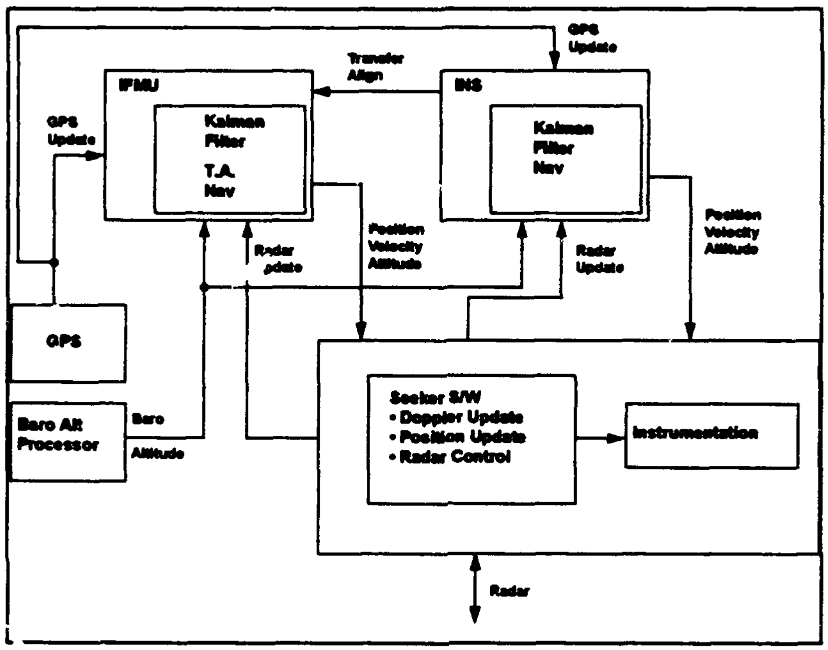

Figure 14

ASARG Block Diagram

Two inertial navigation systems are shown in Figure 14. This is an implementation for the captive flight test and would not be present in an operational system. The Honeywell Integrated Flight Management Unit (IFMU) is a tactical grade IMU and is used for inertial navigation for the tactical, AGM-130 application. The INS is a higher quality IMU and is used for the cruise missile application. The INS is also used to transfer align the IFMU in the AGM-130 application. The GPS provides position and velocity updates to the inertial systems. The inertial systems also provide position, velocity, and attitude data to the radar for stability and motion compensation. The GPS receiver updates the inertial systems with position and velocity data, and the barometric altimeter stabilizes the altitude channel calculations.

Prior to the mission, the missile is provided with WGS 84 earth coordinates (latitude, longitude, and elevation) of the target and a set of target templates corresponding to several LOS headings. After the aircraft has ingressed to the target area but prior to missile launch, a 12-state Kalman Filter performs a transfer alignment between the aircraft INS and the missile INS. The 12 states consist of 3 components of velocity error, 3 attitude errors, 3 components of accelerometer bias errors, and 3 components of gyro bias errors. The goals of the transfer alignment are to provide accurate matching of missile INS velocities with those of the aircraft INS and to determine the major INS sensor errors of gyro and accelerometer bias. An "S" turn maneuver is performed by the aircraft during transfer alignment to help separate attitude errors from accelerometer bias errors. After alignment, the missile is launched and midcourse guidance begins.

For the tactical mission, midcourse guidance lasts about 100 seconds, and a 21-state Kalman Filter is used for midcourse inertial navigation error estimation. These 21 states consist of 3 components of position error, 3 components of velocity error, 3 attitude errors, 3 components of gyro bias, 3 components of accelerometer bias, 2 GPS clock errors, seeker LOS azimuth error, seeker LOS elevation error, seeker range rate error, and seeker range error.

The inertial system used for the tactical mission, the IFMU, is a low cost 1 degree per hour strapdown system. The coordinate frames used by the IFMU are the seeker frame, the body frame, the local level frame, the navigation frame, and the earth centered earth fixed frame. Direction cosine matrices are used for performing the coordinate transformations, and quaternions are used for calculating the Euler angles. The navigation equations are processed at 100 Hz. The GPS receiver is a Collins/Rockwell 5-channel, P-code receiver that is loosely coupled with the IFMU. The GPS receiver provides position and velocity updates at 1 Hz to the IFMU. The IFMU inertially aids the GPS receiver with position and velocity data at 10 Hz.

For the cruise missile mission, midcourse guidance lasts about two hours. During the first 6 minutes of the midcourse phase, the INS receives GPS updates of position and velocity. After the first 6 minutes, the INS receives velocity and landmark postion updates from the SAR seeker for the remainder of midcourse. A 15-state Kalman Filter is used for inertial navigation error estimation. These 15 states consist of 2 components of position error, 2 components of velocity error, 3 attitude errors, 2 components of gyro bias, 2 components of accelerometer bias, seeker LOS azimuth error, seeker LOS elevation error, seeker range rate error, and seeker range error.

The inertial system used for the cruise missile mission, the INS, is a higher quality 0.8 nmi CEP strapdown system. The coordinate frames used by the INS are the seeker frame, the body frame, the local level frame, the navigation frame, and the earth centered earth fixed frame. Direction cosine matrices are used for performing the coordinate transformations, and quaternions are used for calculating the Euler angles. The navigation equations are processed at 100 Hz. The Collins/Rockwell GPS receiver is loosely coupled with the INS. The GPS receiver determines the error in INS position and velocity. This error data is calculated at a 1 Hz rate. The velocity error is sent to the INS Kalman Filter at 0.1 Hz and position error is sent at 0.017 Hz. The INS inertially aids the GPS receiver with position and velocity at 10 Hz.

Once the missile enters the terminal area, about 3 km from the target, terminal guidance begins. The GPS updates are suspended and the SAR seeker is activated. The terminal guidance system interfaces are depicted in Figure 15.

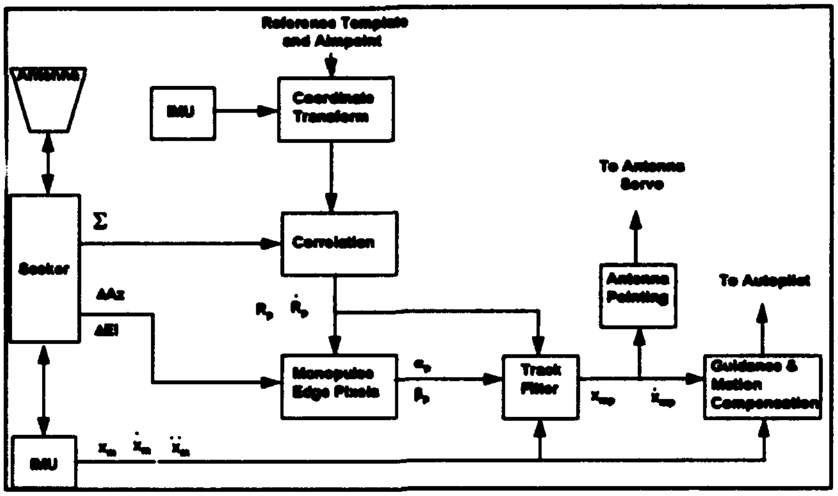

Figure 15

ASARG Terminal Guidance Block Diagram

The seeker sends out pulses of energy and forms a SAR image from the energy returns. The seeker frame rate is 1 Hz nominal. A target detection algorithm then correlates this image with a coordinate transformed template of the target. If the target is contained in the SAR image, range and range rate of the target are determined and input to the terminal Kalman Filter. Current missile inertial position and velocity data are input to the filter as well. The outputs of the filter are the relative missile/target position and velocity vectors. This information is then used by the proportional navigation guidance law to compute a control command for the autopilot.

C. OCD

The OCD program verifies the operational value of direct attack using seekerless GPS aided inertial guidance with an end-to-end system demonstration. Six weapons were dropped in a free flight demonstration. For this demonstration, an Mk-84 warhead was guided with GPS aided inertial guidance using a modified GBU-15 control system. A system block diagram is shown in Figure 16.

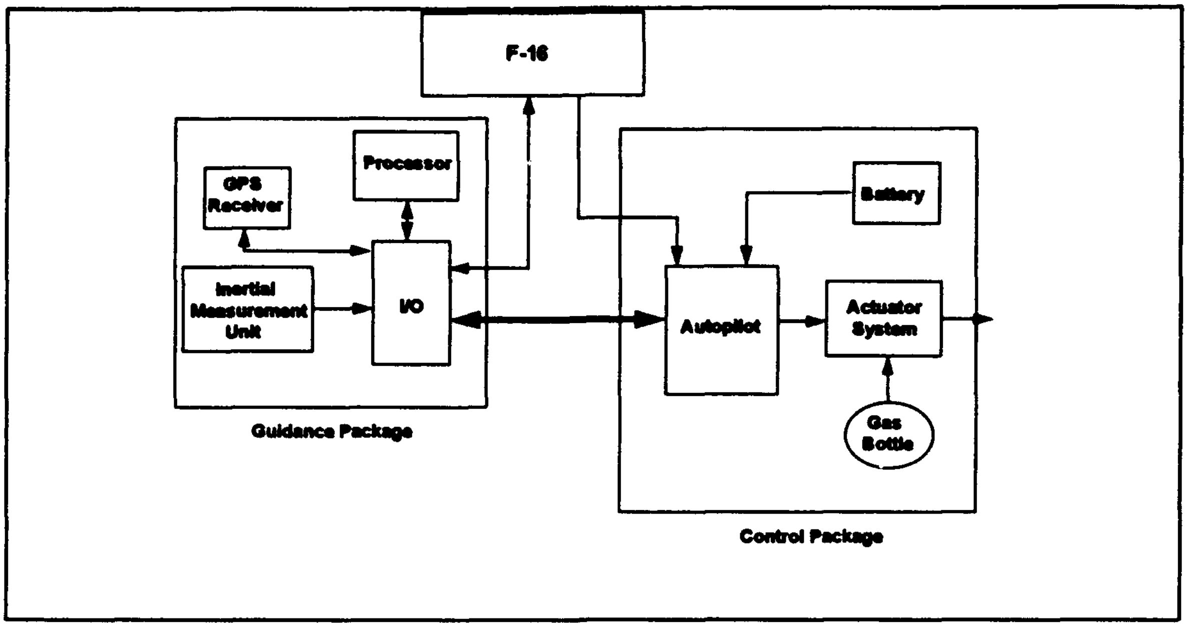

Figure 16

OCD System Block Diagram

The system is broken into two groups, the guidance package and the control package. The guidance package consists of the GPS receiver, inertial measurement unit, and guidance computer. Guidance package outputs are used in the proportional navigation guidance law to compute a control command for the autopilot. The output from the autopilot feeds the pneumatic actuators which drive the fins for steering the munition.

In the OCD operational scenario, the munition is carried, initialized and released from an F-16 C at subsonic speeds. The munition then enters the midcourse phase of guidance. Inertial guidance is used until the GPS satellites have been acquired. After this time the GPS aids the inertial system with position and velocity. The midcourse phase ends after the munition guides to an entry point to the terminal trajectory. After the midcourse phase, GPS updates are suspended, and the test vehicle guides to the target to attain a subsonic near vertical impact.

Prior to the mission, target latitude, longitude, and elevation in WGS-84 coordinates are provided to the munition. This target data could also be provided in-flight from the aircraft pilot. Preplanned targeting makes use of the Mission Support System II (MSS II) mission planning system to preprogram targeting parameters. These parameters are transferred to the aircraft using a data transfer cartridge prepared by the MSS II.

Before the munition is launched from the aircraft, a 16-state transfer alignment Kalman Filter is used to align and initialize the munition inertial system. These 16 states contain 3 components of position error, 3 components of velocity error, 3 attitude errors, 3 components of gyro drift, 3 components of accelerometer bias, and 1 accelerometer scale factor error. After transfer alignment has occurred, midcourse guidance begins, and the munition glides to an area over the target. An 18-state navigation Kalman Filter is used during midcourse guidance. These 18 states contain the 16 states in the transfer alignment filter plus a GPS clock error state and a GPS clock drift state. The inertial navigation system is a Honeywell integrated flight management unit (IFMU) strapdown ring laser gyro system. The ring laser gyros have a drift rate of 1 degree per hour. The coordinate frames used by the inertial system are the body frame, the local level frame, the navigation frame, and the earth centered earth fixed frame. Direction cosine matrices are used for the required coordinate transformations, while quaternions are used for calculating the Euler angles. The navigation equations are processed at 100 Hz. The GPS receiver is an Interstate Electronics Corporation, 5-channel, P- code receiver. This receiver is tightly coupled with the inertial system and is inertially aided with acceleration data at 1 Hz. The GPS receiver provides position updates at 0.2 Hz to the inertial system. Once the munition enters the terminal area, the munition pitches over and uses proportional navigation to guide to the target. The GPS updates are suspended 10 seconds before the munition's near vertical, subsonic impact of the target.

In all, there are many similarities in the guidance concepts used in these three demonstration programs. A comparison of some of the guidance system implementations is given in Tables 2 - 6.

	ATLAS	ASARG	OCD
GPS RECEIVER			
Brand	IEC	Magnovox	Collins
# Channels	5	5	5
Code	P-Code	P-Code	P-Code

Table 2

GPS Receiver Comparisons

IMU CHARACTERISTICS	ATLAS	ASARG	OCD
Brand	Litton Iron Gyros	Honeywell IFMU RLG	Honeywell IFMU RLG
Drift Rate	1 deg/hr	1 deg/hr	1 deg/hr
Equations Of Motion Processing	100 Hz	100 Hz	100 Hz

Table 3

IMU Comparisons

INS/GPS COUPLING	ATLAS	ASARG	OCD
	Loosely	Loosely	Tightly
GPS Position Update Rate	0.016 Hz	0.016 Hz	0.2 Hz
GPS Velocity Update Rate	0.1 Hz	0.1 Hz	N/A
INS Position Update to GPS	N/A	10 Hz	N/A
INS Velocity Update to GPS	5 Hz	10 Hz	N/A
INS Acceleration Update to GPS	N/A	N/A	10 Hz

Table 4

INS/GPS Coupling Comparisonss

NAV KALMAN FILTER STATES	ATLAS	ASARG	OCD
X Position Error	1	1	1
Y Position Error	1	1	1
Z Position Error	1	1	1
X Velocity Error	1	1	1
Y Velocity Error	1	1	1
Z Velocity Error	1	1	1
Attitude Error	3	3	3
Platform Tilt	3	0	0
Gyro Bias	3	3	0
Gyro Drift	0	0	3
Gyro Scale Factor	3	0	0
Accelerometer Bias	3	3	3
Accelerometer Scale Factor	2	0	1
GPS Clock Bias	0	1	1
GPS Clock Drift	0	1	1
Seeker Azimuth Error	0	1	0
Seeker Elevation Error	0	1	0
Seeker Range Error	0	1	0
Seeker Range Rate	0	1	0
True Z Misalignment	1	0	0
Reference Altitude Error	1	0	0
(TOTAL)	22	21	18

Table 5

Filter Error State Comparisons

TERMINAL FILTER STATES	ATLAS	ASARG	OCD*
X Position Error	1	1	
Y Position Error	1	1	
Z Position Error	1	1	
X Velocity Error	1	1	
Y Velocity Eror	1	1	
Z Velocity Error	1	1	
Attitude Error	0	3	
Azimuth Misalignment	1	0	
* Same as Navigation Filter			
(TOTAL)	7	9	

Table 6

Terminal Kalman Filter Comparisons

SECTION X

CONCLUSION

The terminology for autonomous weapon guidance listed in Section I has been discussed throughout the text of this document. Each part of the mission scenario given in Section I was described in at least a top-level engineering manner. This description should have given the reader a better feel for the elements that comprise an autonomous guidance system, as well as some of the equations used for design. This description was then reinforced through the discussion of three Air Force missile guidance system programs.

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