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Integration of the Global Positioning System

With an Inertial Navigation System

By

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11 May 1998

Abstract

Navigation is the determination of the position and velocity of a moving vehicle. Navigation systems used to measure this state vector can be one of two types, either positioning or dead-reckoning.

Positioning systems, such as the Global Positioning System (GPS) measure the state vector without regard to the path traveled by the vehicle in the past. On the other hand, dead-reckoning navigation systems, such as the Inertial Navigation System (INS) determine the state vector from a continuous series of measurements relative to an initial position.

By integrating the unique and complementary characteristics of each system into one integrated INS/GPS system, accuracies as well as additional benefits can be achieved even though unattainable by either system independently.

The optimal method of integrating these two systems is through the use of a Kalman filter. This mathematical technique is used for computing the best estimate of the state of a process which varies with time. Approaches to this filtering can either be centralized in a main filter or federated, where filtering is done at individual sensors.

This theory can then be applied to real world scenarios, whether it be an aircraft during flight, an aircraft during precision approach landings, or the failure detection and isolation of a GPS signal.

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Introduction

Navigation is the determination of the position and velocity of a moving vehicle. The three components of position and the three components of velocity make up a six component state vector that fully describes the motion of the vehicle. Navigation systems used to measure this state vector can be one of two types, either positioning or dead-reckoning.

Positioning systems measure the state vector without regard to the path traveled by the vehicle in the past. A typical example of this which most people are aware of today is the Global Positioning System (GPS). On the other hand, dead-reckoning navigation systems determine the state vector from a continuous series of measurements relative to an initial position. A common example of this which is found in most commercial and military aircraft today is the Inertial Navigation System (INS).

Each of the examples stated above are widely used around the world today. Both are considered to be the best and most accurate source of navigation information available in their respective classes. However, by integrating the unique and complementary characteristics of each system into one integrated INS/GPS system, accuracies along with additional benefits can be achieved even though unattainable by either system independently.

The optimal method of integrating these two systems is through the use of a Kalman filter. This mathematical technique is used for computing the best estimate of the state of a process which varies with time. In a GPS/INS integration, filtering can either be performed in a central location, accepting unprocessed data from the various sensors; or it can be performed at the sensor locations, and then their solutions combined in a master Kalman filter, also known as the federated approach. Again, each of these methods have their own benefits and drawbacks.

This theory can then be applied to real world applications, whether it be an aircraft during flight, an aircraft during precision approach landings, or the failure detection and isolation of a GPS signal. In flight tests, the GPS/INS integration produces results twice as accurate as a GPS stand-alone system. In precision approach landings, both Category I and II approaches can be accomplished through the integration of GPS/INS and other common sensors. Last of all, a GPS/INS integration is able to quickly determine and isolate failures from a bad GPS satellite signal before that signal is allowed to produce large errors in a navigation solution.

Background

NAVSTAR Global Positioning System

The NAVSTAR GPS was developed as a U.S. Department of Defense multi-service program in 1973. Today, this program serves as an all-weather global radio-navigation system. GPS is a space-based system that provides users with highly accurate three-dimensional position, velocity, and time information anywhere on or near the surface of the Earth. The GPS is comprised of three major segments: space, control, and user. These segments can be seen in Figure 1 below.

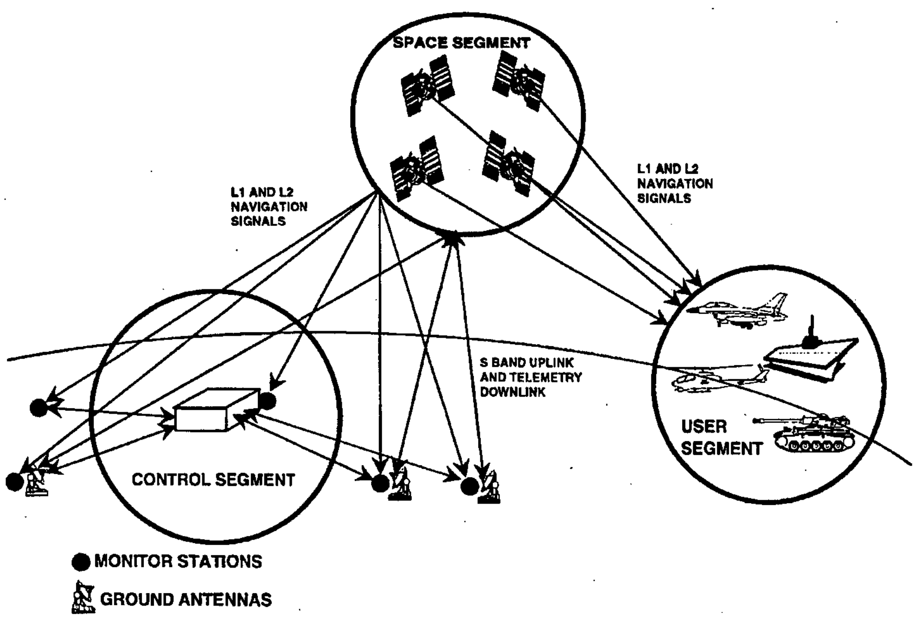

Figure 1
Global Positioning System

The space segment is composed of 24 GPS satellites that are in approximately 12-hour orbits (11h 57m 57.27s) at an altitude of 20,200 km (10,900 nmi). The satellites are in near-circular orbits in six planes, each at an inclination of 55 degrees. Each satellite transmits signals at two frequencies in the L-Band: L1-1575.42 MHz and L2-1227.6 MHz. These signals are modulated with pseudorandom noise (PRN) codes that provide the instantaneous ranging capability of GPS.

The control segment consists of five monitor stations, three of which have uplink capabilities. The information from the monitor stations is processed at the Master Control Station in Colorado Springs to determine satellite orbits and to update the navigation message of each satellite.

The user segment consists of a receiver containing an antenna and a processor which will use the GPS signal data to provide position, velocity, and accurate timing to the user.

Table 1 below summarizes the functions of each segment, along with the required inputs to complete their respective functions, and the outputs of each segment.

Table 1
GPS Segments

	Inputs	Functions	Products
Space	satellite commands navigation messages	provide atomic time scale generate PRN RF signals store and forward nav messages	PRN RF signal navigation message telemetry
Control	PRN RF signals telemetry Universal Coordinated Time (UTC)	estimate time and ephemeris predict time and ephemeris manage space assets	navigation message satellite commands
User	PRN RF signals navigation messages	solve navigation equations	position, velocity, and time

GPS provides two positioning services, the Precise Positioning Service (PPS) and the Standard Positioning Service (SPS). The PPS can be denied to unauthorized users, but the SPS is available to any user worldwide. Special keys allow the authorized user to acquire and use the encrypted precise (P) code on both frequencies and to correct for intentional degradation of the signal. The coarse/acquisition (C/A) code, however, is available to all users but is only carried on the L1 frequency along with the P code

The intentional degradation of the GPS signal, referred to as selective availability (SA), is meant to deny accuracy to an unfriendly force. SA may be turned on or off at any time. If it is turned off, SPS accuracy is the same as PPS. Recently, the vice-president announced the U.S. intentions to turn off SA by the year 2005.

GPS is basically a one-way ranging system. To provide accurate ranging measurements, the GPS satellites contain atomic frequency standards which are accurate to one nanosecond. To compensate for inaccurate clock readings in receivers, four satellites in view are needed so that the fourth variable (user clock bias) can also be determined.

Determining the navigation information of the user requires the calculation of the following equations:

$$R_i = \eta c(TOA) \tag{1}$$

$$R_i = \sqrt{(x - x_{si})^2 + (y - y_{si})^2 + (z - z_{si})^2} - \eta ct \tag{2}$$

where

R_i = pseudorange from the receiver to the ith satellite (km)

η = average index of refraction in the propagation medium

c = speed of light in vacuum = 2.99792458 x 10^5 km/sec

TOA = time of arrival of the signal from satellite to receiver (sec)

x, y, z = unknown position of the receiver (km)

x_{si}, y_{si}, z_{si} = known position of the ith satellite (km) t = time offset of the receiver clock (sec)

However, sometimes simply having four satellites visible may not be sufficient. This case occurs when the geometry of the satellites does not offer means to compute a highly accurate position. A satisfactory navigation solution is one where there are at least four satellites visible to the user above a desired elevation angle, usually taken to be 5°, and line-of-sight vectors to those satellites provide an adequate geometric solution. A measurement of this concept is know as the geometric dilution of precision (GDOP) where

$$GDOP = PDOP + TDOP \tag{3}$$

where PDOP is the position dilution of precision defined as:

$$PDOP = \sqrt{\sigma_x^2 + \sigma_y^2 + \sigma_z^2} \tag{4}$$

where σ^2 are the variances in each direction of the satellite's position in the Earth-centered, Earth-fixed (ECEF) reference frame, and TDOP is the time dilution of precision, the contribution of clock error to the error in pseudorange. Adequate coverage is usually defined by the DoD when PDOP is less than 6 for elevation angles greater than 5°.

Horizontal and vertical dilution of precision (HDOP and VDOP) measurements can also be determined by transforming coordinates to a local tangent plane coordinate frame (N, E, Up). Generally, HDOP is at least twice as good as VDOP. [6, 8]

Two measures of performance for the GPS are the circular error probable (CEP) and the 2drms (two times the root mean square). The CEP is considered to be the radius of a circle, centered at the actual position (or the mean position of a group of measurements) that encloses

50% of the measurements. GPS errors are also frequently defined in terms of a circle of radius 2drms where

$$2drms = 2\sqrt{\sigma_x^2 + \sigma_y^2} \tag{5}$$

Using these measures of performance, GPS can determine a user's position with the following degree of accuracies:

Authorized L1/L2 user	8.1 m CEP
C/A code L1 user with SA	100 2drms
C/A code L1 user w/o SA	42.2 2drms

Accuracies of the GPS are dependent on errors that can exist within any of the three segments described earlier. Some of the typical errors found in the GPS are found in the following:

SA Errors Typical pseudorange errors have a standard deviation of about 30 meters, but they have the potential to be higher

Ionospheric Delays These propagation errors can be as high as 20 to 30 meters during the afternoon hours to 1 to 6 meters at night if not removed using two frequency corrections. Ionospheric models can reduce this by approximately 50%.

Tropospheric Delays Can be as much as 30 meters to a low-elevation satellite but are predictable and can be modeled.

Ephemeris Errors The difference between the actual satellite location and the computed satellite location. Usually less than 2 or 3 meters, but can be increased significantly with SA

Satellite clock errors Difference between the actual satellite clock time and that computed from the broadcast corrections.

A more recent concept which can significantly reduce the errors associated with the GPS is know as differential GPS (DGPS). This method requires a reference station at a known location that receives the same GPS signals as a normal user. This reference station processes its GPS measurements, and then transmits the corrections to participating users in the area. The user then applies these corrections to his measurements, in effect canceling all common errors. Accuracies of less than 1 meter to 10 meters have been experienced using DGPS. [8]

Ending this discussion on GPS, keep in mind the potential advantages and disadvantages that exist. The main advantage of the GPS satellite navigation system is that it provides a highly accurate all-weather worldwide navigation capability. However, the major disadvantages are that it can be vulnerable to intentional or unintentional interference and temporary unavailability due to signal masking or lack of visibility coverage.

Inertial Navigation Systems

As opposed to positioning systems such as GPS, dead-reckoning systems provide navigation information by the measure of acceleration or velocity with respect to an Earth-referenced coordinate system. One such dead-reckoning system used in many commercial and military aircraft today is the inertial navigation system (INS).

In simple terms, in an INS, displacement is calculated from the measured acceleration. In some applications velocity is also desired. This displacement, velocity, and acceleration can be found by integrating the acceleration with respect to time as shown in the following equation:

$$x = \int v dt = \iint a dt dt \tag{6}$$

Although advantages and disadvantages of an INS will be discussed further, one drawback already apparent is the system's time dependency. By integrating the above equation, we see that the position is proportional to the square of the time. Therefore, any errors in the system will continue to grow as time progresses.

However, one immediate advantage is that the means to acquire a navigation solution are self-contained within the vehicle itself and does not require an external signal. The fact that this makes them unsusceptible to jamming or spoofing makes them very popular with the military. Many ships, submarines, guided missiles, space vehicles, and virtually all modern military aircraft are equipped with inertial navigation systems.

The basic theory behind inertial navigation is that a series of accelerometers are used to measure the vehicle's acceleration while a set of gyroscopes are used to measure the attitude of the accelerometers in order to calculate which direction the acceleration forces are acting.

In the early inertial navigation systems, the accelerometers and gyroscopes sat on a platform stabilized by gimbals in order to isolate the instruments from the angular motions of the vehicle. The gyroscopes acted as error-sensors, whose purpose was to sense the small misalignment in the gimballed axes. A motor would then be activated to keep the platform stabilized in inertial space. This permitted the accelerometer outputs to be integrated into velocity and position. [2, 13] This type of INS is pictured in Figure 2.

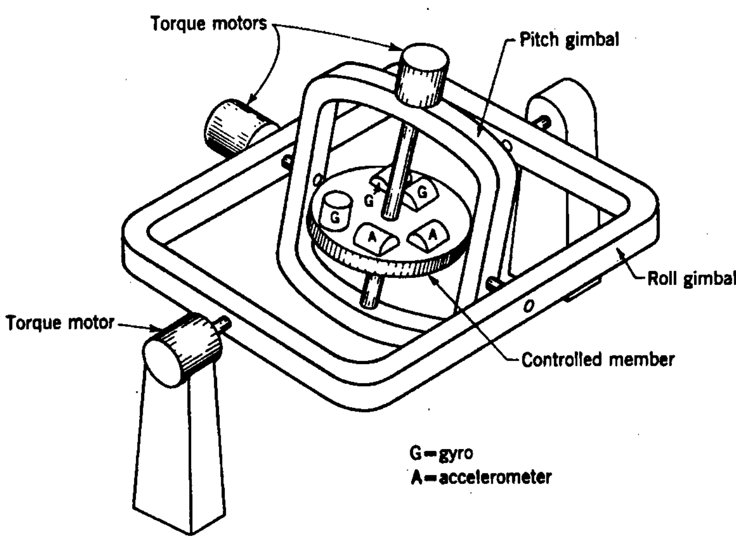

Figure 2
Inertial Navigation System

In the early 1980's, the strapdown inertial system was developed. In this system, the gyroscopes and accelerometers are mounted directly on the vehicle. The gyroscopes track the rotation of the vehicle, and algorithms in the computer convert accelerometer measurements from vehicle coordinates to the navigation coordinates where they can be integrated.

The basic theory behind an accelerometer is that a device with a known mass will produce some type of output (usually electrical) when acted on by an outside force. In this case, the acceleration can be measured based upon Newton's first law where the known force is equal to the known mass times the vehicle's unknown acceleration.

The purpose of the gyroscopes in an INS is to stabilize the accelerometers in inertial space. In gimballed platforms, the gyros measure the small rotation of the platform, and restore the platform to its stable position using gimbal servos. In strapdown systems, the gyroscopes are fixed to the vehicle and follow its angular motion. Thus, the accelerometers remain on an "analytic platform".

On today's military aircraft, gyroscopes must sense angular rates as low as 0.005 deg/hr and as high as 400 deg/sec. For long commercial flights, requirements on the order of 0.01 deg/hr are needed. Because of the need for high accuracy in this wide range, ring laser gyroscopes (RLG) have become the predominant inertial navigators for military and commercial aircraft. (8)

The RLG operates as follows. The laser gyro detects and measures angular rates by measuring the frequency difference between two contra-rotating laser beams. The two laser beams circulate in the "ring" cavity simultaneously. If the cavity is rotating in an inertial sense, the propagation times of the two light beams are different. The delay manifests itself in the form of a phase shift between the two beams, and the phase shift is detected by a pair of photo detectors. Figure 3 on the following page displays the process of a ring laser gyro. Devices of this type are extremely reliable due to the absence of moving parts. [6]

However, in any INS there are many places in which errors can enter into the positioning solution. Gyro drift errors are caused by temperature variations, accelerations, magnetic fields, and vibrations. Accelerometer errors can be caused by variations in temperature or by vibration inputs. There can be assembly errors in that the gyroscopes and accelerometers may not be aligned perfectly. Computational errors exist due to readout accuracy, as well as approximations inherent in the algorithms. Last of all, errors in the initial conditions (position, velocity, tilt, and azimuth) will always be present as the system measures over time.

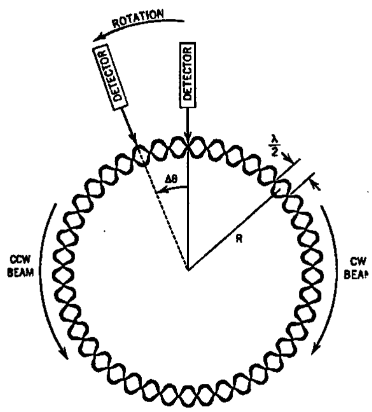

Figure 3
Ring Laser Gyroscope

All of these errors listed above lead to a degradation of the navigation information over time, whether the vehicle is moving or is stationary. Another drawback of the INS is that it must be aligned when turned on. This is to initialize the position and velocity measurements so the computer can process the correct initial orientation of the platform. The final drawback of the INS is its cost. Two years ago, the cost of an INS ranged anywhere from \$50,000-\$120,000, depending on the accuracy which was desired. [8]

However, there are advantages of inertial navigation systems which make them very attractive. First of all its measurements of position and velocity are instantaneous and continuous. Likewise, as mentioned earlier, the system is completely self-contained, since it is based on measurements of acceleration and angular rate made within the vehicle itself. Because of this, an INS does not radiate energy, and is non-jammable. And last of all, the navigation information is obtainable at all latitudes, in all weather, without the need for any ground stations.

Theory

GPS-INS Integration

To this point, we've discussed two of the predominant navigation instruments in use today, the Global Positioning System and inertial navigation systems. Each in its own has certain limitations as to the accuracy and reliability of the system. However, by integrating the two systems, the advantages of each can be exploited to give a highly accurate, reliable navigation solution in virtually any flight environment.

Table 2 below is a description of the major characteristics of the GPS and INS and the method in which they achieve a navigation solution. From this brief summary, we can take a look as to how an integrated GPS/INS system would be most beneficial.

Table 2
GPS and INS Comparison

GPS	INS
Time-independent system errors	Time-dependent errors
Needs external inputs	Self-contained
Outages due to limitation in visibility	Continuously available information
Self-initializing	Initialization required
Low accuracy in high dynamics	High accuracy in high dynamics
Data available at moderate rates	Data available at high rates

The first comparison is the dependence of errors on time. With the INS, the navigation errors are a function of time or the distance traveled. Eventually, this will lead to unacceptable errors within the system. However, the GPS solution is time-independent, and errors will not vary based on the time or distance traveled.

Another complementary aspect of the two systems is that GPS relies on sensors requiring external signals in order to calculate the navigation solution. Therefore, access can be denied, due to intentional (jamming) or nonintentional interference (transmission blockage or interfering transmissions) of the signal. On the other hand, INS receives, gathers, and processes all information on-board, not relying on any external sources to calculate its position, therefore, able to give a continuous navigation output.

As discussed in the previous section, in order for INS to give proper data, the initial conditions of the vehicle must be known (initial position, velocity, and orientation). However, there is no initialization required for a GPS receiver. Although it may take additional time to begin tracking satellites, the navigation information will be readily available without any knowledge of past information.

A benefit of the INS that supports a shortcoming of a stand-alone GPS navigation system in many aircraft is its high accuracy in highly dynamic maneuvers. Even with several antennae, the problem of shadowing can become a problem in an aircraft that is undergoing highly dynamic maneuvers (steep turns, rolls, loops, etc.). Additionally, the position and attitude are changing so rapidly that a GPS navigation system may not be able to update its data as rapidly as the aircraft is performing maneuvers. However, an INS is capable of continuously providing navigation information to the user, even during high dynamic maneuvers. The reason for this is the high tolerance of the INS, whose accelerometers can accurately sense loads up to 20 g's and whose gyroscopes can accurately sense rotations up to 400 deg/sec. [5, 7, 8]

With these complementary characteristics, it becomes quite clear how an integrated GPS/INS navigation system would produce a much more accurate and reliable navigation solution. However, some of the techniques in which to exploit these individual strengths may need further discussion.

For example, the integration of GPS information with INS data will inherently limit the inertial position and velocity errors, thus giving a more accurate position. However, this position can be calculated even more accurately by correcting the natural drift in position by re-initializing the INS during flight. The simplest method of doing this is to reset the position to the coordinates of a point on the surface of the Earth. However, this can also be achieved by resetting the INS according to GPS coordinates with which it is integrated.

Another way to exploit the strengths of the two systems is by having an INS that will work independently when GPS signals are lost or corrupted. Whether it be because of transmission blockage, satellite failure, or interference, the GPS signal may not always be able to be tracked. Therefore, the self-contained INS will be capable of navigating independently until tracking conditions improve sufficiently for GPS. At this point, the INS provides data on position, velocity, and acceleration which can be used in aiding the GPS to reacquire its signal more quickly. [7]

Closely related to the above situation, is just the natural uncertainties of the GPS signal at times. For example, delays in the GPS signal due to the ionosphere are already the largest natural source of GPS navigation errors. With the approach of a new solar cycle maximum over the next three or four years, many people are worried about the reliance on a single stand-alone GPS navigation system. Especially after the last solar maximu in March of 1989 created an unstable ionosphere that made GPS untrackable for periods of time. Once again though, an integrated GPS/INS navigation could detect this failure of the GPS signal in the navigation solution and could isolate it from the computations. [12]

Another exploitation of an integrated system is the ability of a GPS/INS system to enhance antijamming performance. Narrow-beam antenna can be pointed more directly at GPS satellites to avoid jamming. Based on data from the INS as to the attitude of the vehicle as well as data from the GPS or INS as to the position of the vehicle with respect to the satellites, pointing requirements can be sent to those narrow-beam antenna. Even if the GPS signal does get jammed, the inertial aspect of the system can once again take over as the principal navigation sensor until the aircraft has flown beyond the range of the jammer. [3]

Several specific applications of an integrated GPS/INS navigation system will be discussed later in this paper. For now, it is important to remember that the GPS/INS integration provides better accuracy and reliability because it exploits the individual strengths and minimizes the weaknesses of each stand-alone system. For instance, the GPS maintains accuracy over a long period of time, while the INS will drift from its initial alignment. The GPS is susceptible to jamming, while the INS is completely self-contained. Finally, the GPS performs poorly in highly dynamic environments due to the shadowing of satellites and high rates of change; however, the INS has excellent performance in high dynamic manuevers.

Mathematical Technique

Kalman Filtering

Now that we've discussed the reasons why we should integrate an INS with GPS, the next question is how to do so. The optimal method of performing such an integration is by using a Kalman filter. Simply speaking, the Kalman filter is a recursive mathematical technique for computing the best estimate of the state of a process which varies with time. The filter achieves its best estimate by minimizing the mean square error in its estimates of the modeled state.

In its basic form, the algorithm for a Kalman filter consists of merely predicting the errors in the state, and then once new observations have been taken, correcting these state errors to obtain an optimal solution. As time progresses, errors within the system will be modeled more accurately to obtain an even better estimate of the state. This is accomplished through the covariance matrix. This matrix defines the probable error in the filter's estimate of the state vector. The diagonal elements of the error covariance matrix are the variances of the error in the estimate of the state vector, while the off-diagonal elements are covariances between different states in the vector which describe the amount of correlation between different members of the state vector.

To begin a Kalman filter, the equations describing the process and the relationships must first be determined. They follow the general form of:

$$x_{k+1} = \phi_k x_k + q_k \tag{7}$$

where

x_k is the n x 1 state vector

q is the n x 1 white noise vector

ϕ is the n x n state transition matrix from time k to time k+1

The state transition matrix, ϕ, describes how the state vector propagates from one time to the next. The noise variance, Q, tells us about the noise in the state process. If the variance is large, a large amount of randomness is inserted into the process with each step

The comparison of measurements from the navigation sensors must then satisfy the following relationship:

$$z_k = H_k x_k + v_k \tag{8}$$

where

z_k is the m x 1 measurement at time k

v_k is the m x 1 measurement zero mean, white noise vector with variance R

H_k is the m x n matrix containing the ideal relationship between z_k and x_k

The H matrix describes the relationship between sensor measurements and the error states to be estimated. R describes the mean-square measurement noise variance. Large values generally mean poor measurements.

The most challenging aspect of applying Kalman filtering is establishing mathematical equations to model the physical situation at hand and putting them in the form of the two equations above. Once this is determined and an initial state vector and covariance matrix are estimated, the process is simple by following the algorithm below. [7, 8]

1. Initialize filter with estimates for state vector, x, and covariance matrix, P

2. Compute the Kalman gain, K

$$\mathbf{K} = \mathbf{P}^{-}\mathbf{H}^{T}(\mathbf{H}\mathbf{P}^{-}\mathbf{H}^{T} + \mathbf{v})^{-1} \tag{9}$$

3. Update the estimate with the new measurement

$$\mathbf{x} = \mathbf{x}^{-} + \mathbf{K}(\mathbf{z} - \mathbf{H}\mathbf{x}^{-}) \tag{10}$$

4. Compute the covariance matrix for the updated estimate

$$\mathbf{P} = [\mathbf{I} - \mathbf{K}\mathbf{H}]\mathbf{P}^{-} \tag{11}$$

5. Propagate the state vector and covariance matrix

$$\mathbf{x}^{-} = \phi\, \mathbf{x} \tag{12}$$

$$\mathbf{P}^{-} = \phi\, \mathbf{P}\, \phi^{T} + \mathbf{q} \tag{13}$$

6. Goto step number 2

The most common error state model used in a GPS/INS integration is an 11-state vector consisting of nine variables used to model INS errors and two variables used to model GPS errors. A typical state vector may therefore consist of the following:

1. velocity error
2. platform tilt about y-axis
3. north position error
4. north east position error
5. east velocity error
6. platform tilt about x-axis
7. vertical position error
8. vertical velocity error
9. platform azimuth error
10. user clock bias
11. user clock drift

However, some models exist with a state vector containing as many as 69 variables. This is composed of a 39-state INS error model in conjunction with a 30-state GPS error model. [5]

Whatever the state error vector size may be, though, once calculated in the Kalman filter algorithm, it now directly gives us the information necessary to better correct and update our measured position by adding or subtracting these calculated errors to their respective navigation variables.

Applying this Kalman filtering theory to a GPS/INS integration now gives us one major decision, at what point in the integration do you apply the filter. Two possible solutions to this question exist. One is in a centralized location where raw sensor data is first combined to produce a solution, known as the tightly-coupled or centralized filter. The other is where the main filtering is accomplished with sensor data that has already been processed through filters at the sensor locations themselves. This is known as loosely-coupled or federated Kalman filtering. Figures 4 and 5 below show examples of the two systems.

As you can see, the centralized implementation uses the most basic of information from each sensor (sometimes referred to as raw or unprocessed information). On the other hand, the federated Kalman filter requires that each sensor already process a navigation solution before passing data to a master Kalman filter. These characteristics provide both benefits and drawbacks for each method.

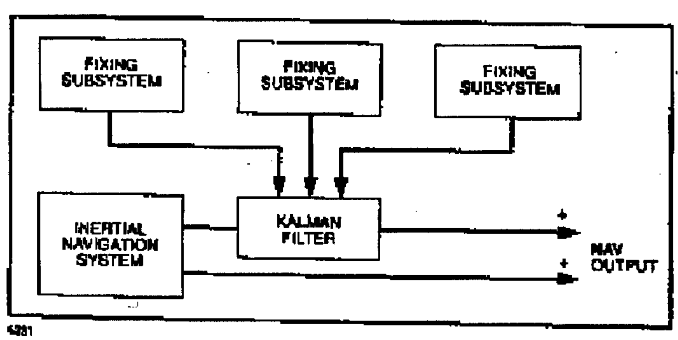

Figure 4
Centralized Structure

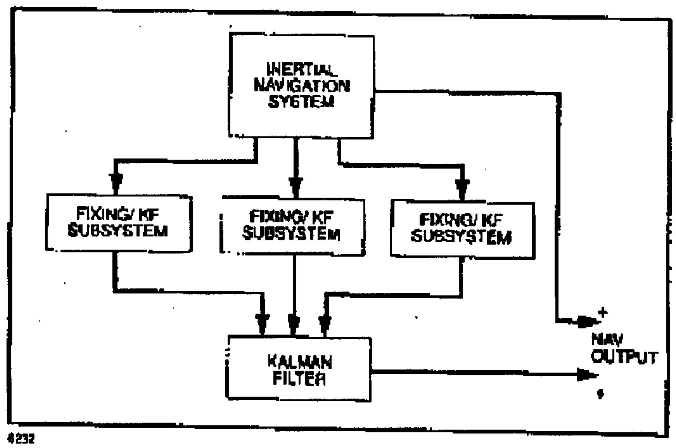

Figure 5
Federated Structure

For example, since the tightly-coupled method sends raw data to a centralized filter, less than four satellites in view are needed to be of benefit in an integration. Pseudorange and range-rate signals from even one satellite can be fused to the data from the INS in order to enhance a better navigation solution. However, using a federated Kalman filter requires that at least four satellites be in view, due to the fact that the GPS sensor must solve for a complete solution before passing data on to a master filter. This passage of data through two or more filters in the federated approach can also be detrimental since errors from one filter will be passed along to the next. [1,7]

A benefit of the federated approach, however, is its simplicity. With existing inertial navigation systems, a GPS receiver can easily be integrated by simply adding its solution to that of the INS in a Kalman filter. An integration using the centralized approach would be much more complex and expensive since it requires fusion of the signal data, not the solutions, of each sensor. However, new software packages are beginning to make this small barrier obsolete. Months of integration using a centralized filter are being accomplished in weeks due to this standardization of software. All that is left is for a systems integrator to determine how to interface the INS and GPS into a Kalman filter, and the software will perform the actual integration of the data.

A final benefit of the federated approach is that of integrity monitoring of each sensor, whether it be the GPS, INS, altimeter, etc. Since each sensor is producing its own independent solution, uncorrelated to the other sensors, the solution of each sensor can be compared to that of the global navigation solution. If a sensor's solution varies beyond a predetermined threshold compared to that of the global solution, that sensor's data can be ignored until it has recovered from its failure. The following figures show an example of this integrity monitoring. A failure has been introduced into the GPS sensor. Using a centralized filter, this error goes uncorrected as seen in Figure 6. However, by employing a federated filter, this error and its source are identified quickly so that the sensor's solution can be ignored until its recovery, thus providing an uncorrupted signal, as shown in Figure 7. [1]

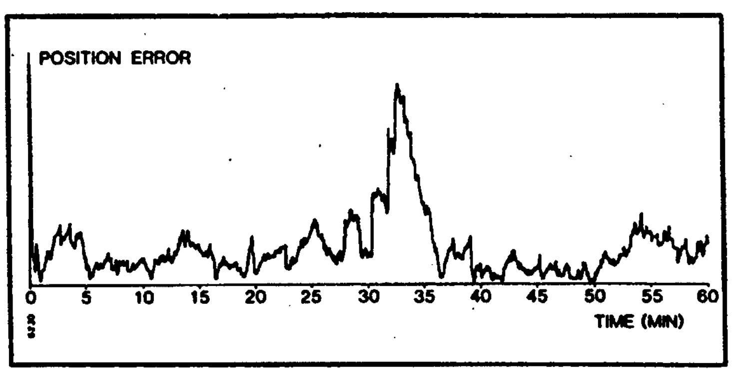

Figure 6
Performance using centralized filter

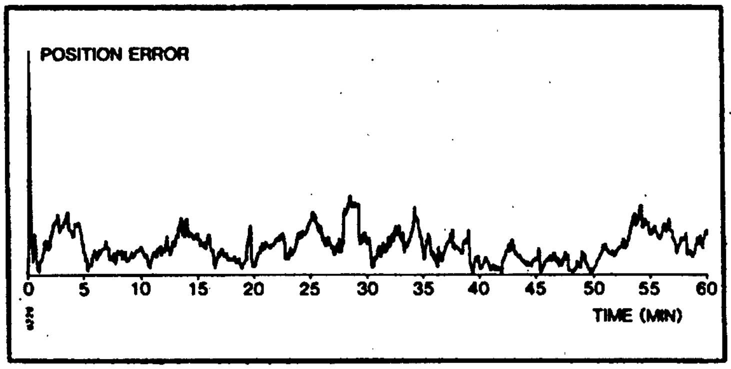

Figure 7
Performance using federated filter

One other option that can also be employed in an integrated system is one in which a feedback implementation estimates the errors in the inertial system, but then feeds these errors back to the INS in the form of corrections. Therefore, the inertial errors are not allowed to grow unchecked. However, a disadvantage of the feedback method is that the INS is dependent on the Kalman filter estimates. Without proper integrity monitoring, this reliance on the filter could allow the INS to be reset to false positions and possibly become unstable. Such integrity monitoring is one application that will be discussed in the next section of this paper. [9]

Applications

Having explained the benefits and techniques for integrating the Global Positioning System with an inertial navigation system, what remains is a discussion on actual applications and the test data achieved from this integration.

The first application is a normal flight test performed on a simulation at Wright Patterson AFB by Evans and Riggins. This test consisted of a 71 minute simulated flight in fairly static conditions (easy turns, no high dynamics, etc.). Integration was achieved using a Litton LN-94 ring laser strapdown INS with either NAVSTAR's XR-4PC or 5PC GPS receiver. Additionally, they tested the accuracy of using different error states in the Kalman filter: a 69-state, a 41-state, and a 13-state. Results of this simulation of the integrated system along with the stand-alone GPS systems are found in Table 3.

Table 3
Integrated vs. Stand-alone GPS and INS

Filter	Averaged Mean Error (feet)
69-state (39 INS, 30 GPS)	118.17
41-state (39 INS, 2 GPS)	137.54
13-state (11 INS, 2 GPS)	140.37
XR-4PC only	252.42
XR-5PC only	318.34

As one can see, there is a significant increase in the accuracy of the navigation solution by integrating the GPS receivers with the INS. There is nearly a two-fold increase in accuracy of an integrated system compared to that of the stand-alone GPS system. [5]

An application which has drawn much attention recently is the problem of GPS integrity monitoring, which is the system's ability to provide timely warnings to users as to when it should not be used because of a poor navigation signal to the user. Currently, it takes the control segment from fifteen minutes to two hours to determine that there is a problem, identify it, determine a course of corrective action, and implement that action.

One method of detecting these failures is receiver autonomous integrity monitoring (RAIM). RAIM refers to a method that is based on a consistency check among various ranging signals to detect an unacceptably large satellite error. The RAIM approach requires five satellites in view to detect a failure, and six satellites to identify the satellite that failed. However, this can become a problem since six satellites are in view of a given user usually no more than thirty percent of the time.

Another method that has drawn the attention of the navigation community today is a failure detection and isolation (FDI) structure for GPS integrity monitoring in an integrated INS/GPS navigation system. It employs the concept of using a bank of auxiliary integrated INS/GPS Kalman filters, each of them processing a portion of the GPS measurements, to provide a consistency check between main integrated GPS/INS Kalman filter and those of the auxiliaries. With this method, only four satellites need to be in view at any given time, thus approaching 100% availability.

The failure detection and isolation is determined by testing the consistency between the state vectors and covariance matrices of the auxiliary Kalman filters with those of the main filter. This consistency value, S, for any filter can be calculated as follows:

$$S = (X_i - X_t)^T(P_i + P_t)^{-1}(X_i - X_t) \tag{14}$$

where

X_i is the state vector of the filter under test

P_i is the covariance matrix of the filter under test

X_t is the state vector for the entire system solution

P_t is the covariance matrix for the entire system solution

If the value of S exceeds a certain threshold then that certain navigation subsystem is classified as having failed, and thus can be isolated.

A simulation was run consisting of an 800 second flight due west at a constant velocity of 300 m/s. The effectiveness of this GPS integrity monitoring system was investigated by assuming a failure in satellite number 4. Results of this simulation can be found below where Figure 8 shows the results of the global solution, while Figure 9 shows the results of the auxiliary Kalman filter using signals from satellites 1, 2, and 3.

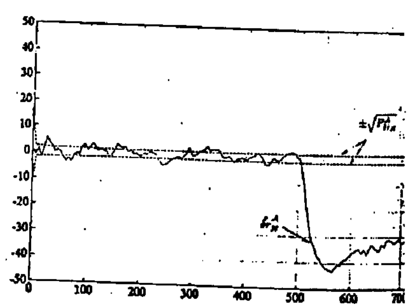

Figure 8
Global solution position error

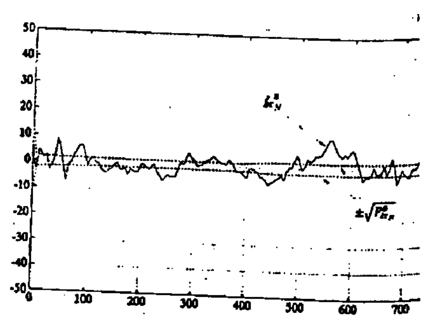

Figure 9
Auxiliary solution position error

The simulation results demonstrate that the FDI structure is capable of not only detecting the presence of a GPS satellite failure well before large position estimation errors result, but also of identifying which satellite is malfunctioning. Therefore, future data from this satellite could be ignored until this satellite has recovered from its failure. [4]

To date, Litton has developed a software modification for its LTN-101 INS/GPS system The software uses the inertial reference system to spot GPS satellite errors before they can cause major navigation errors. Most significantly, it can do this even with bad satellite geometry or as few as four satellites in view. [10]

A final application that can be beneficial with the integration of GPS with an INS is that of aircraft precision approach landings. Currently, most research in this area has consisted of using stand-alone GPS receivers combined with differential correction. However, with the standardization requirements and the vastly enormous number of airports, this may not be a viable option. This also fails to mention that the undetected loss of a navigation signal or the failure of a receiver could be catastrophic, especially during a landing relying on this system alone. An additional problem that could arise deals with RF interference effects. The low level of power received from the GPS satellites makes the satellite-based landing system more susceptible to RF interference and receiver noise than that experienced with the ILS and MLS. Currently, the Department of Defense and the commercial airline industry are utilizing the ILS and MLS during aircraft landings for precision approaches.

However, studies by Gray and Maybeck [13] at the Air Force Institute of Technology show that by integrating INS and GPS along with barometric altimeter, radar altimeter, and a pseudolite, Category I and II precision approach requirements established by the FAA can be met.

Table 4 below shows the requirements for each category for aircraft on precision approaches. A failure detection of ten seconds is also required for category I and II landings and two seconds for category III landings.

Table 4
Precision Approach Accuracy Requirements at Decision Heights

Category	Azimuth	Elevation
I	+/- 28.1	+/- 6.8
II	+/- 8.6	+/- 2.8
III	+/- 6.8	+/- 1.0

(in feet, all 1σ values)

Using the flight profile of a KC-135 tanker, four different integration test cases described below were performed. Additionally, they used a P-code GPS receiver with four satellites in view.

Test 1-INS, GPS, Baro-Altimeter

Test 2-INS, GPS, Baro-altimeter, radar altimeter

Test 3-INS, GPS, Baro-altimeter, one pseudolite

Test 4-INS, GPS, Baro-altimeter, radar altimeter, one pseudolite

These test results shown in Table 5 conclude that currently, Type 1 precision approach requirements can be met by the integration of an INS and GPS along with a radar altimeter. Additionally, the addition of one pseudolite in the integration structure can meet the FAA Type II precision approach requirements.

Table 5
Precision Approach Average Errors (feet)

Error State	Test 1	Test 2	Test 3	Test 4
Latitude	8.9	8.8	4.6	4.4
Longitude	9.2	9.2	3.9	3.3
Altitude	15.0	2.6	11.8	2.4
Precision Approach	None	Type I	None	Type II

Conclusion

By exploiting the individual strengths and weaknesses of the Global Positioning System and an inertial navigation system, an integrated GPS/INS can provide better accuracy and reliability than each stand-alone system.

This integration can be applied to various resources, whether it be an aircraft in normal flight, an aircraft on a precision approach landing, or for the failure detection and isolation of the GPS.

Whatever the purpose may be, because of its tremendous performance benefits, the combination of a low-cost, low-to-moderate performance inertial navigation system and a GPS receiver will be a widely used multisensor system for many types of air vehicles for years to come.

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