GPS-BASED ATTITUDE FOR AIRCRAFT

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Abstract

Key words: GPS, attitude, inertial

GPS was used with ultra-short baselines (2-3 carrier wavelengths) in a triple antenna configuration to obtain aircraft attitude in real time. Through algorithm selection and error source calibration, accuracies of 0.1°, 0.15° and 0.2° rms were obtained for pitch roll and yaw respectively. The accuracy and robustness of the system was enhanced by combining the ultra short-baseline GPS attitude solution with an attitude solution derived using inertial sensors. Both inexpensive automotive grade rate gyros and tactical grade inertial sensors were tested. The solid state auto grade gyros allow coasting through temporary GPS outages lasting 2 minutes with attitude errors less than 6 degrees. The tactical grade inertial sensors use the GPS primarily for initial alignment and are able to coast for up to 30 minutes durring AGPS outage. The combined GPS-inertial system has a 20Hz output sufficient to drive glass cockpit type displays. A prototype system was built and flight tested in a Beechcraft Queen Air. The system installed and flight tested in the Queen Air compares favorably to the performance of the existing vacuum driven instruments. It is currently being used in ongoing research at Stanford with futuristic high resolution displays[1].

Introduction

Attitude information for aircraft is currently obtained by spinning rotor or ring laser gyros. In General Aviation (GA) applications a vertical gyro is used for pitch and roll while a separate directional gyro is used for heading. The display of the information to the pilot is presented mechanically by the gyros themselves. Commercial and military aircraft generally have computer-based CRTs or LCD displays ("glass cockpits") that are driven by inertial measurement units (IMUs). These attitude systems are very precise but cost more than most small aircraft. This research is aimed at reducing the requirements and cost of inertial sensors by augmenting them with GPS attitude.

GPS-based attitude determination is just an extension of differential carrier phase position determination. Very precise relative position (mm level) is determined between a pair of antennas. This relative position can then be translated into angular measurements. Two baselines composed of three antennas completely define the Euler angles associated with aircraft attitude and can be used to compute pitch, roll, and yaw angles.

GPS has been investigated by many researchers for its applicability in determining attitude by differencing signals from multiple antennas [2,3,4,5,6,7,8]. The concept has been used successfully for aircraft attitude in flight [3,4,8]. These systems demonstrated to date; however, have used expensive GPS receivers and have not yet proven acceptably reliable for primary aircraft flight instruments.

As part of the goal of this research, we investigated the use of GPS for attitude, but with reduced requirements on the receiver to reduce cost and a more closely-spaced antenna configuration to provide a more robust design for acceptable aircraft use. Although the closer spacing degrades the accuracy of the GPS attitude solution, our system is enhanced by adding inexpensive, solid-state rate gyros to smooth the noise and to provide a high bandwidth response, even when using GPS receivers with sampling as low as 1Hz.

1. GPS Attitude Determination

A. General

There are two factors that affect the resolution of pointing accuracy derived from GPS carrier wave measurements. The first is the error characteristics of the L1 carrier phase measurements. These effects have to do with the receiver, the mounting of the antennas and the characteristics of the antennas. The second is how a specific attitude determination algorithm maps these errors into the Euler angle domain. This mapping is typically a function of the number of unknowns involved and is a function of the GPS receiver. We explore both these aspects in depth.

An additional factor involved in using L1 carrier phase measurements to determine attitude is the resolution of the integer ambiguity inherent in the carrier phase measurements. In any configuration where antenna separation exceeds $\lambda/2$ (λ=19cm), there are potentially multiple solutions for the attitude problem. We discuss methods to resolve these ambiguities.

B. Historical Perspective

Typically GPS attitude determination systems have used wide antenna separations to improve pointing accuracy. The attitude determination system tested in [4] utilized a baseline with separation between antennas on the order of 10m in order to achieve good accuracy. This approach introduced structural flexibility as an error source thus necessitating an additional antenna for a total of four. The large number of wavelengths between antennas introduced many possibilities for the integer ambiguity and necessitated aircraft motion or extensive searches to initialize the system. In the use of the system described in [4] over the last several years at Stanford University, it has been found that solutions are not reliable and often require extensive taxiing to provide the initialization. If lock is lost in the air, re-initializing takes tens of seconds; however, when properly initialized, the system was shown to provide attitude to within 0.1°.

An alternate approach to the wider baselines for improved accuracy is to better understand and thereby eliminate the GPS phase errors while optimizing the algorithm used to improve the mapping of errors from the phase domain to the attitude domain. There are several inherent advantages to this short baseline approach. The benefit is that the attitude algorithm becomes much simpler and more robust and can be implemented with more inexpensive processors with a much higher level of integrity. In addition, the cost of installation of an operational system is reduced significantly.

Flight tests and static tests have been conducted on two isosceles triangle configurations: one with 36 cm and 50 cm baselines and another with 16 and 36 cm baselines. These configurations are small enough to be installed on top of the fuselage of a high or low wing GA aircraft. The main advantage of such short baselines is that the integer search space is reduced considerably and if any integer is off by one, the attitude solution is drastically different and easily identifiable. This allows for robust integrity monitoring of the system, a requirement in aviation applications. The problem with the short-baseline attitude system is that it is more sensitive to the noise in the phase measurements. The noise level of GPS carrier measurements is on the order of 5 mm rms, which is negligible for large baselines. However, for the short baselines that error can translate to an attitude error of several degrees depending on the algorithm used.

C. Algorithm Selection

Two types of attitude algorithms have historically [6] been used: a known line/clock bias and an unknown line/clock bias. The unknown bias method solves for the bias at every epoch and as such does not require a common clock between all receiver-antenna pairs. This algorithm is used by attitude systems using multiple original equipment manufacturer (OEM) boards with separate clocks. The known bias technique requires a common clock for all antennas and presumes a constant known line bias. Both of these approaches have also been used in a nonlinear form with the added constraint of baseline length. The two equations are derived from the basic attitude equation shown in Eq. 1. The [X YZ] vector represents the baseline in ENU coordinates and [LOS LOS LOS] vector represents the line of sight to the satellite in East, North, Up (ENU) coordinates.

$$
\begin{aligned}
&\Delta\ {}^{i} + N\ \ + d\ \ = H\ \ B \\
&where: \\
&B = \begin{bmatrix} X \\ Y \\ Z \end{bmatrix} \\
&H = \left[LOS_X^i \quad LOS_Y^i \quad LOS_Z^i \right]
\end{aligned}
\tag{1}
$$

The known bias case is solved as follows in Equation 2.

$$
B = \left(H^T H\right)^{-1} H^T \left[\Delta\ {}^{i} + N\ \ + d\ \ \right] \tag{2}
$$

The unknown bias case is solved by moving the clock term to the right side of the equation and including it in the B vector as one of the unknowns. That is,

$$\Delta\ {}^{i} + N = H\ B$$
$$where:$$
$$B = \begin{bmatrix} X \\ Y \\ Z \\ d \end{bmatrix} \tag{3}$$
$$H = \begin{bmatrix} LOS_X^i & LOS_Y^i & LOS_Z^i & 1 \end{bmatrix}$$

The solution to this is identical to the solution for the known bias case given in eq.(1), only now B has 4 elements and H includes a column of ones.

The nonlinear solution can be developed from either the known bias or unknown bias case by adding the baseline length as a known constant. We will derive the nonlinear known bias case below. In general, the baseline vectors generally have nonzero X, Y, and Z components. Applying the baseline constraint in such instances leads to a nonlinear equation; however, if one transforms the baseline vector to a coordinate system that results in the Y and Z components of the baseline vector being zero, then the X component becomes the baseline length and the constraint can be applied easily. This transformation gives us a new matrix Hn shown in eq.(4a). We then define a matrix Hn1 as the first column of Hn and a matrix Hn2 as the last two columns of Hn shown in eq.(4c). These new matrices allow us to rewrite eq.(4b) in a simpler form shown in eq.(4d). We can then take the pseudo inverse of eq.(5d) and solve for the remaining two components of the baseline shown in eq.(4e).

This nonlinear method obviously assumes approximate knowledge of the baseline orientation and would be computed iteratively. We are now left solving for two unknowns in the known bias case and three unknowns in the unknown bias case. Afurther refinement of this nonlinear method is described by Cohen in Ref[7]. Cohen's method solves for changes in Euler angles directly and takes advantage of multiple baselines when available.

$$Hn = H\ A \tag{4a}$$
$$\Delta\ + N\ + d\ = Hn \begin{bmatrix} 0 \\ Y \\ Z \end{bmatrix} + Hn \begin{bmatrix} X \\ 0 \\ 0 \end{bmatrix} \tag{4b}$$
$$Hn = \begin{bmatrix} Hn1 & | & Hn2 \end{bmatrix} \tag{4c}$$
$$\Delta\ + N\ + d\ = Hn2 \begin{bmatrix} Y \\ Z \end{bmatrix} + Hn1\ [X] \tag{4d}$$
$$\begin{bmatrix} Y \\ Z \end{bmatrix} = \left(Hn2^T Hn2\right)^{-1} Hn2^{-1}\left[\Delta\ + N\ + d\ - Hn1\ [X]\right] \tag{4e}$$

The elements of the pseudo inverse of H are analogous to the Dilution of Precission (DOPs) in the navigation equation. By looking at the diagonals of this matrix we can come up with an Attitude Dilution of Precision Matrix (ADOP). This ADOPmatrix is in the units of the LOS vectors and yields accuracies of baseline component estimates, in this case east-north-up directions.

$$ADOP = \begin{bmatrix} {}^2_{east} & & \\ & {}^2_{north} & \\ & & {}^2_{up} \end{bmatrix} = \left(H^T H\right)^{-1} \tag{5}$$

The diagonal terms of this matrix are East DOP, North DOP, and Up DOP of the baselines. The DOP represents how the error in phase measurements maps into the error in relative position between two antenna. For example, an EastDOP of two would mean a L1 Phase error of 5 mm would result in an east position error of 1 cm.

The DOP calculations will vary with the number of satellites in view and the type of algorithm used by the receiver to select satellites. The most common receiver algorithm is to take the highest elevation satellites. This method has been used for the following DOP calculations. A period of 12 hours and a location of Stanford, CA is used to calculate DOP for the five highest satellites. This is shown in Figure 1.

Figure 1 shows that the DOPis not constant over time but varies significantly with satellite geometry. For the unknown bias case, the DOP in the vertical direction is significantly worse than the DOP in the horizontal directions. This is expected since the DOP calculation is essentially the same as the GDOP used for position. This is not the case with the known bias calculation. In fact, the DOP in the vertical direction is better than the horizontal DOPs. This is because the known bias case would be analogous to the GPS position problem if one did not have to solve for a clock offset. If this is the case, we no longer get an improvement in DOP by having satellites on both sides of the user. In addition, for the nonlinear case where the baseline length is known, we get virtually identical results to calculating the three components of differential position.

This analysis can be expanded further by taking different numbers of satellites and looking at the average DOP over a 12 hour period. This is plotted in Figure 2. There are several very critical points brought out by

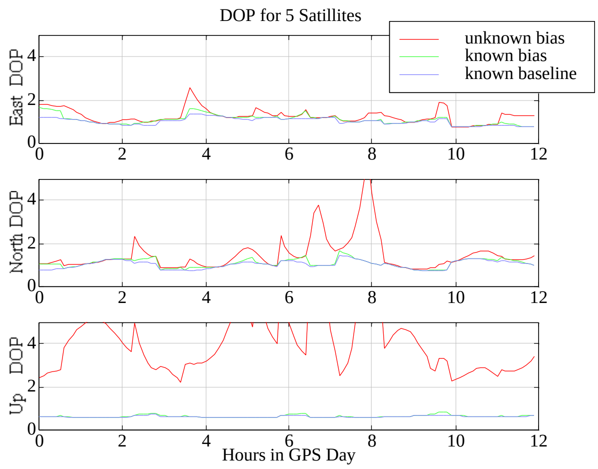

Figure 1. DOP as a function of time and algorithm

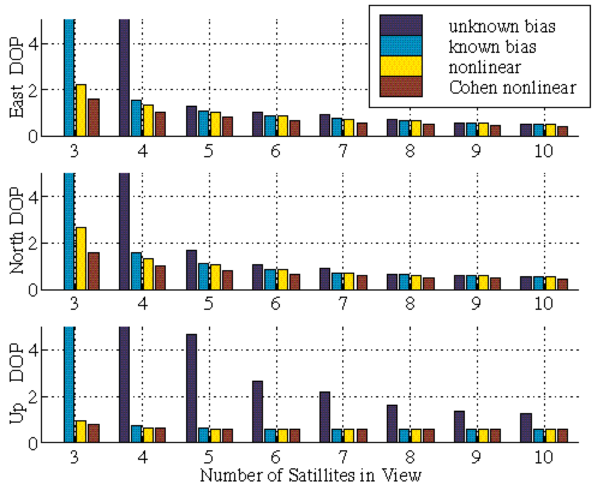

Figure 2. DOP as a function of number of satellites and algorithm

Figure 2. In the East and North directions, as long as there are at least 8 satellites, there is no difference in performance between algorithms. In the up direction however, regardless of the number of satellites in view, at least a factor of 2 improvement is observed with a known bias calculation than an unknown bias calculation. With at least 4 satellites in view, there is no significant advantage to using the nonlinear solution with the baseline length constraint. Cohen's solution does provide a slight improvement in the East and North DOP by taking advantage of both baselines, but gives no improvement in the more critical Up DOP.

In level, or near level flight, the UpDOP will translate to pitch and roll angle errors and the EastDOP and NorthDOP will translate into heading errors. With this in mind, some very general conclusions can be drawn. There is no need use a common clock GPS receiver if heading is the primary concern; however, in aircraft applications, where accurate pitch and roll measurements are critical, it is advantageous to utilize a common clock receiver to allow the short baseline system to attain the best accuracy.

Based on the above derivation, a relationship can be made between baseline length and pointing accuracy for a given level of GPS phase error. The case of L1 phase noise of 5 mm and DOPS derived from tracking 6 satellites is shown in Figure 3 for the known bias and unknown bias algorithms.

It is obviously critical to utilize this improved algorithm in short baseline aircraft attitude configurations. This requires the development of a low cost multiple antenna common clock receiver. Current analysis and testing has been utilizing the Trimble Quadrex which allows both computation methods to be utilized and compared. The points on the right of the graph represent work done by Cohen Ref[7] with long baselines. The point on the right of the graph represents work done by Cohen [7] with the 10 m baseline while the other measured points are from the two baselines studied with this contract.

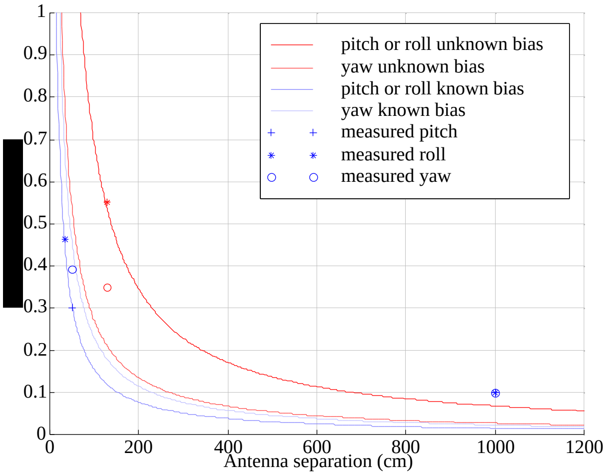

Figure 3. Angle errors vs. baseline length for different algorithms

D. GPS Attitude Error Sources

Once the attitude algorithm has been optimized, additional improvements in the attitude solution can be made by improving the quality of the differential phase measurements. The high frequency noise in the phase measurements may be averaged out by the inertial instruments, but longer term errors must be calibrated out.

Long term phase errors can be broken down to those caused by multipath (i.e. signal reflection) and those caused by variation in antenna phase patterns. Phase delay maps for patch antennas are discusses extensively in [9]. By taking a single phase difference between two antennas, one effectively introduces any differences in the antenna phase delay patterns as phase errors. Both multipath and antenna phase errors have the effect of delaying the phase measurement as a function of the line of sight (LOS) vector from the antenna to the GPS satellites.

In aircraft installations, the primary effect is from antenna phase error, as most multipath disappears when the aircraft is airborne. Indeed, the only multipath remaining on an aircraft in flight is due to the aircraft structure. This effect can be calibrated out in the same manner as the antenna phase error.

Figure 5 shows the repeatability of antenna phase errors for 4 days. The phase data shown is for one satellite taken at 2 Hz and averaged over 100 seconds. The 4 phase error lines were purposely offset by 1 cm increments every 24 hours for clarity.

It is important to notice the repeatability of even the very fine structure of the phase error as the satellite tracks through the same azimuth and elevation path. This implies that this error is deterministic and hence can be calibrated out. In addition, the very steep gradients of the phase error means that a very small change in LOS may cause a relatively large change in phase error. This necessitates a very fine grid when modeling the phase error over the full range of azimuth and elevation angles. However this extreme sensitivity of the error also means that a very small change in the attitude of the platform will cause the phase errors to de-correlate in time. Actual flight conditions are not perfectly static and effectively introduce dither. This dither effect changes the temporal characteristics of the phase error to a much higher frequency and and allows some of it to be filtered out by low grade inertial sensors. This effect reduces the phase error calibration requirement in actual aircraft applications. The irregular spacing between the phase error lines from day to day indicates a slowly varying line bias effect. This phase offset is identical from channel to channel over the same time period and represents an additional error source to be considered later.

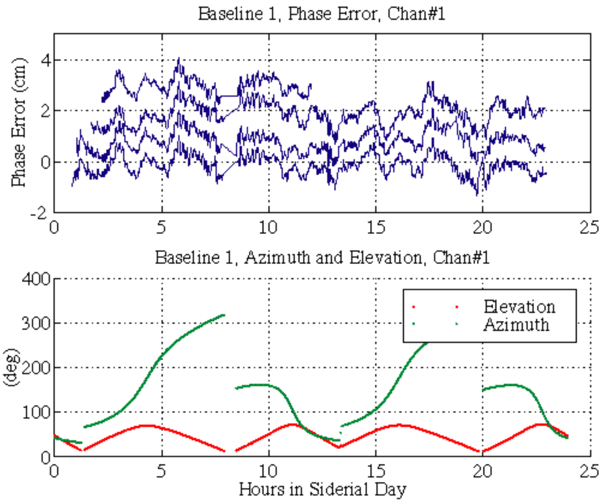

Figure 5. Plot of repeatability of GPS error

In order to calibrate out this repeatable antenna phase error it is necessary to take phase measurements over all combinations of azimuth and elevation. Using a movable platform it is possible to accomplish this by rotating the entire antenna array. The LOS to the satellites relative to the platform is shown in Figure 6 for 3 different orientations over a total of 4 days. The maximum spacing between azimuth and elevation tracks is 4 degrees. Three orientations is the minimum number to adequately cover all azimuth and elevations. Ideally more orientations would be used to better cover the azimuth and elevation space.

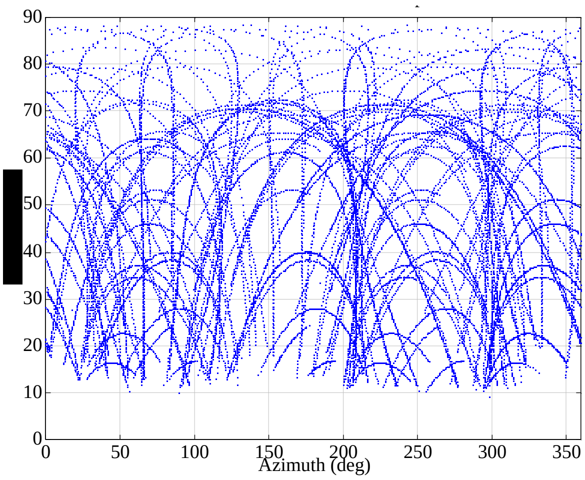

Figure 6. Line of Sight Relative to Platform Orientation

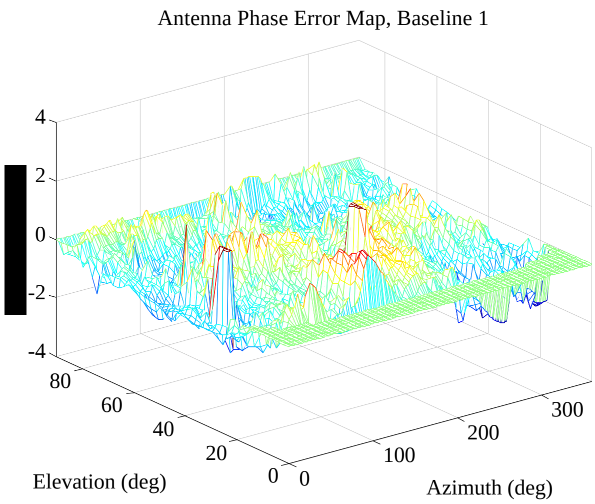

Figure 7. Plot of phase map for 1 baseline

These deterministic phase errors shown in Figure 5 have been modeled as a function of the relative azimuth and elevation of the satellite. The phase delay map for a given baseline is shown in Figure 7. There is as much as a 1 cm phase error introduced depending on the arrival azimuth and elevation angle to the satellite. As seen in Figure 8, by subtracting out the phase error from Figure 7, the rms error in the phase measurements decreases from 5 mm rms to 2.5 mm rms.

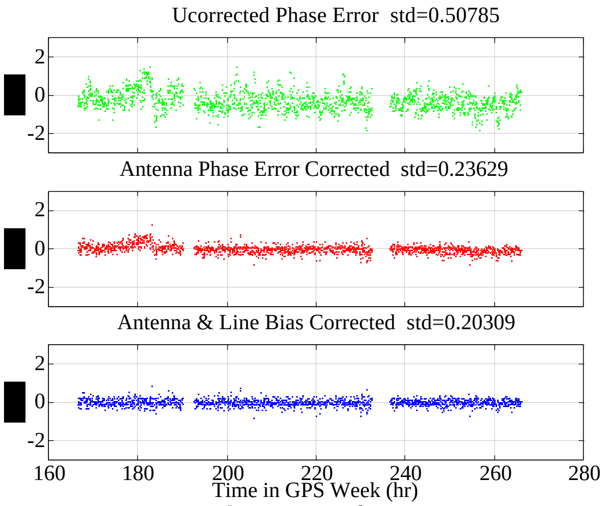

Figure 8. Phase error after antennae phase map removed

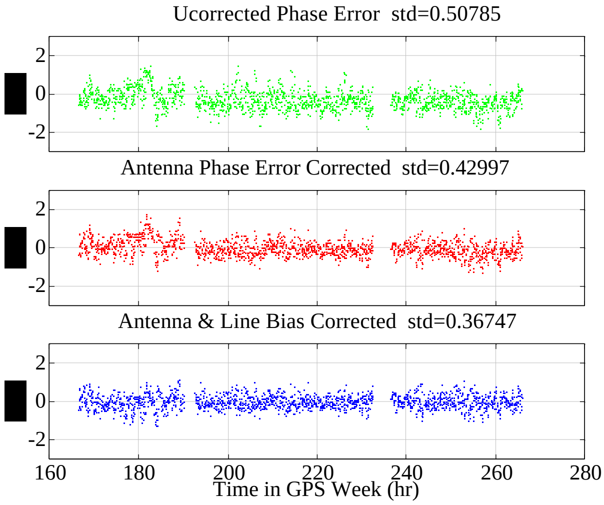

Figure 9. Phase error after external multipath removed

To determine conclusively whether the phase error was in fact due to antenna phase distortion or whether it was due to external multipath, an attempt was made to correlate the phase error data with the absolute LOS as opposed to the LOS relative to the platform. The change in phase error for this case is shown in Figure 9.

There is a slight improvement in this external multipath case as well, but this is due to the fact that there are some azimuth and elevation combinations that have only one data point. Any such points will show an improvement regardless of the LOS frame of reference. If more platform orientations were used there would be a negligible improvement in the absolute LOS case. This conclusively demonstrates that the the phase error is in fact a result of antenna phase error as opposed to external multipath.

The second trace in Figure 8 shows an additional correlation with time. This is a change in line and clock biases that typically occurs as a result of temperature effects on the antenna cables. This error is common to all receiver channels for a given antenna pair and is slowly varying. If the error is averaged over a period of 4 hrs and then subtracted from the phase measurements, the improvement is shown in the 3rd trace of Figure 8.

The sequential improvements for pitch, roll, and yaw due to calibration of all error sources are shown in Figure 10 for a 6 channel receiver. The large gains in pitch and roll are obtained by using a common clock algorithm. Following that, improvements are made by calibrating out antenna phase error and changes in line biases. The final resulting performance is angular errors between 0.1° and 0.2° rms.

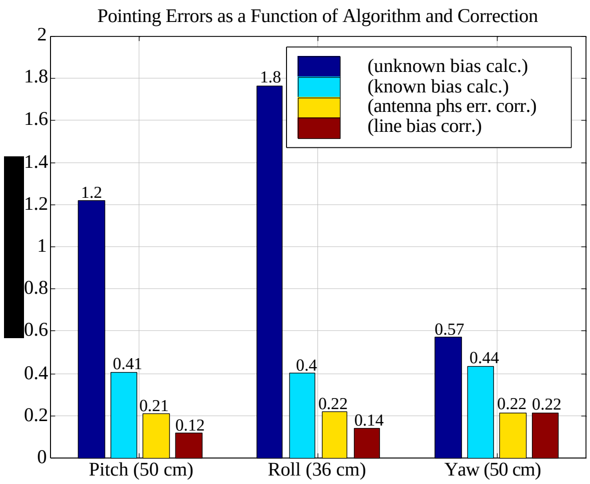

Figure 10. Summary of Pointing Error Improvements

2. Instantaneous Integer Determination

Recently, many integer ambiguity resolution methods have been proposed using motion[10], batch processing[11], dual frequencies[12,13], highly refined algorithms[14,15,16] and attitude bounds[6,17]. The integer ambiguity resolution method used in an Attitude Heading Reference System (AHRS) needs to be reliable, instantaneous, and have a fast computation time. In order to fulfill these requirements, the system utilized here does not choose the most likely set of integers out of a fixed volume as has traditionally been done [16,11]. Instead, it analyzes probable solutions from a variable size group and, using a rigorous set of requirements, continues to search only until it finds the correct integers. The requirements for an integer matrix to be considered correct are so stringent that in rare instances the correct integers will not pass and the search will not return any integers; however, this is preferable to the alternative of returning false integers, and the quick search can easily be repeated at the next epoch.

Recognizing that the computation time for the integer search is inversely proportional to the number of combinations analyzed [18], the integer search is performed over the space of probable attitudes [17] as opposed to the entire integer space [15,16]. This results in a tremendous reduction in the number of integer combinations analyzed. With 6 satellites and 3 baselines of magnitude 2 λ, 3 λ, and 3 λ, the entire integer search space contains 10^{12} combinations. After the inclusion of a few minimal constraints, the variable sized attitude space used here contains at most 1620 combinations and when level an average of just 18. This represents a decrease in the size of the search space by a factor of 10^9 to 10^{11}.

In order to reliably determine if a set of integers is correct, multiple levels of checking are employed. A priori knowledge of the antenna configuration[6] is fully exploited in order generate 4 criteria for each of the 3 baselines. The four criteria for the integers are: a computed baseline length close to the measured value[6], a residual with magnitude smaller than a set maximum, a computed angle between the baselines close to the measured angle on the antenna configuration[17], and a resulting clock bias within predetermined bounds. All constraints are set such that, when tracking 6 satellites, at least 99.9% of the correct integer combinations will fulfill each criterion. The selection of the order in which the selection criteria operations were performed minimized computation time by eliminating attitude possibilities after as few computations as possible.

The overall reliability of the system is dependent on the number of visible satellites. Figure 11 shows that the system has exceptional reliability against returning a false solution and a high degree of success at returning the correct solution after only one epoch.

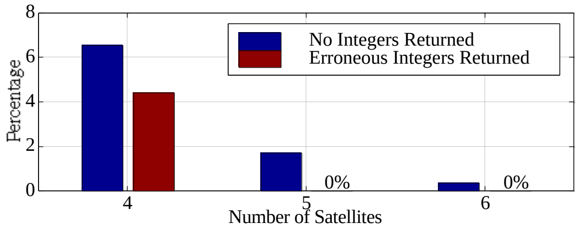

Figure 11 Integer Search Reliability

3. Inertial Sensors

As noted previously, there is a limit to the attitude accuracy that can be obtained by a GPS alone attitude system. The accuracy of the system can be enhanced by combining GPS with inexpensive inertial sensors. Other benefits that are also realized when GPS is fused with inertial sensors is an increased bandwidth and robustness. That is, inertial sensors can provide attitude information at rates as high as several hundred Hz and can be used in high dynamic environments. They will also provide a degree of immunity against temporary GPS outages.

Combining GPS with inertial sensors is not a new idea. Much work has been done in this area in the recent years. Unlike the work reported in Refs [19,20,21,22], most of the work that has been done has involved fusing GPS with expensive inertial sensors. It should be noted that the term "inexpensive" is relative. Inertial sensors that would be considered inexpensive for application in a certain field would be considered expensive in another. For example, Ref [23] reports the operation experiences obtained using a Litton LN-200 Inertial Measurement Unit (IMU) with GPS and classifies this system as inexpensive. Ref [24] reports similar experiences with a system that fused the Trimble TANS Vector with a Systron Donner Motion Pack. The Systron Donner Motion pack sells for approximately $13,000, and the LN-200 costs even more. These inertial sensors, although inexpensive for some applications, would be prohibitively expensive in the GA sector.

In view of the above, the focus of this research has been on using what are called "automotive grade" inertial sensors. This term is employed because such sensors are currently being used for skid control, active suspension and navigation in automobiles. These sensors range in cost from $25 to $1000 in large quantities.

4. Integrated System

A. Introduction

Automotive grade rate gyros and a Trimble Quadrex GPS receiver were combined in an integrated short baseline GPS-inertial AHRS. This system outputs pitch, roll and yaw to an artificial horizon display mounted in the instrument panel for pilot evaluation. A schematic of the system is shown in Figure 12.

The inertial sensors used in this system consisted of three Systron-Donner "Horizon" rate gyros that were mounted orthogonally in a compact 4" x 5" 3" enclosure also containing the microprocessor and all interface electronics. Flight testing was performed on a Beechcraft Queen Air. The Queen Air utilized a 36 x 50 cm baseline configuration shown in Figure 13.

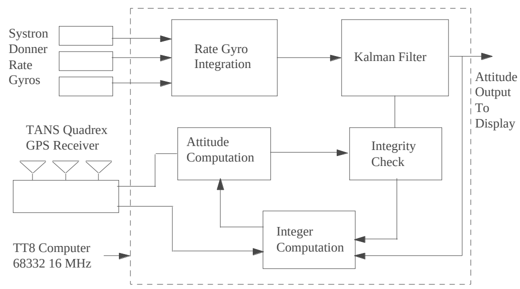

Figure 12. Block Diagram of the Integrated GPS-Inertial AHRS.

B. Real-Time GPS Attitude Algorithm

The known bias GPS attitude algorithm was used in the real time integrated AHRS. The determination of the integer ambiguities at initialization was performed extremely fast due to the fact that one antenna pair (rear two) was 36 cm apart and the other pairs were 50 cm apart. The typical integer search took approximately 2 sec, partially because it was aided initially by the fact that the airplane is near level at system start on the ground, but mostly because there were so few possible values of the integers. In the air, the gyros would provide a good attitude estimate for reinitialization; however, this was rarely required. This integer resolution performed robustly, never computing false integers during any of the flight tests.

Figure 13. Antenna Installation on the Queen Air Test Airplane

C. Integration of Inertial Sensors

The algorithm for blending the GPS attitude solution with the inertial attitude solution in real-time sampled the output from the three gyros at 20 Hz. The output from the gyros was numerically integrated to provide an estimate (time update) of the three Euler angles. This information was subsequently sent to the display at the same rate. The GPS receiver output was sampled at 2 Hz. A Kalman filter was used to blend the GPS attitude solution with the estimates obtained by straight integration of the gyros. The GPS measurements also provided a means for estimating the gyro drift rate. To minimize the computational burden, the estimator used constant gains that were computed "off-line." The filtering and integration algorithms were performed using a TattleTaleTM Model 8 by Onset Computer. The TattleTaleTM Model 8 consists of a Motorola 68332 Processor with 8 12 bit Ato D lines and 16 Digital I/O lines running at 16MHz. The algorithms were written in C and compiled using the MotoCrossTM cross compiler by Peripheral Issues.

D. Integrity Monitoring

In addition to the Kalman filtering and integer resolution, the microprocessor performed an integrity check on the GPS attitude solution prior to sending the attitude information to the filter. This check utilized a constraint on the line biases computed in parallel with the attitude solution. This integrity check was 100% effective in correcting for a small number of cycle slips.

E. Flight Test Data

Several flights tests were conducted utilizing the AHRS as the primary attitude reference by the pilot. Several of the flight tests were also conducted in conjunction with the advanced research display described in Ref [1]. These flights involved flying simulated instrument approaches. Throughout the flight test period the display attitude solution was evaluated for latency and compared with the other attitude reference instruments and the view of the horizon outside the window. The 20 Hz update rate was found to be sufficient to present a fluid display with no jitter or lag observed by the pilots.

The estimates of gyro biases stabilized after about 4 minutes from power up. The stability of these bias values allows the system to accurately estimate attitude when the GPS feedback is removed. Figure 14 shows the convergence of the gyro biases.

As a demonstration of the capability of the system to survive GPS outages, feedback from the GPS attitude solution was deliberately turned off (in post processing) for an extended period of time. The plots in Figure 15 show the deviation between the gyro integrated attitude solution and the GPS attitude solution during this outage. There is less than a 4 degree error

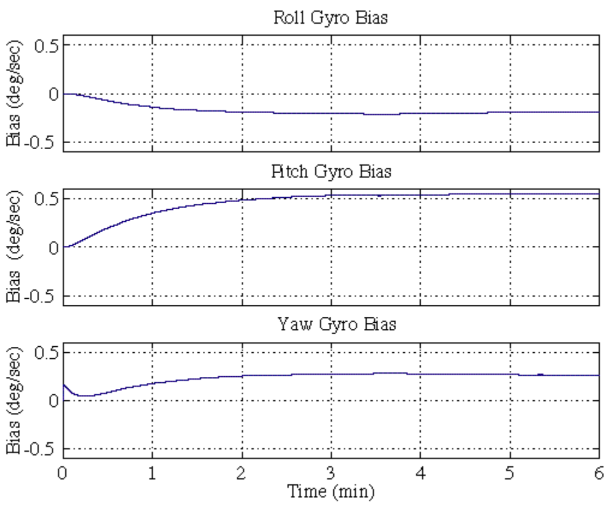

Figure 14. Gyro Bias Convergence

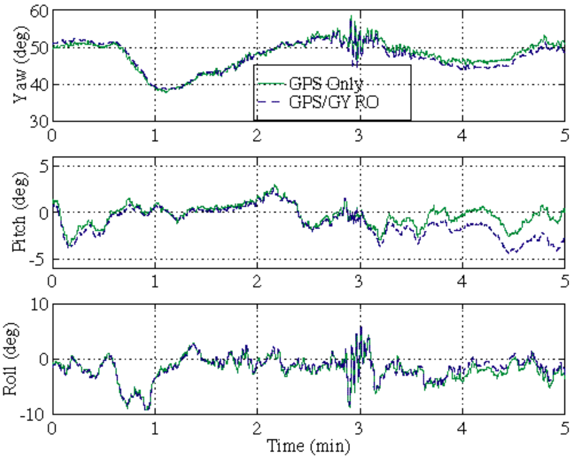

Figure 15. Gyro Coast Capability
(GPS outage occurs at time t = 0).

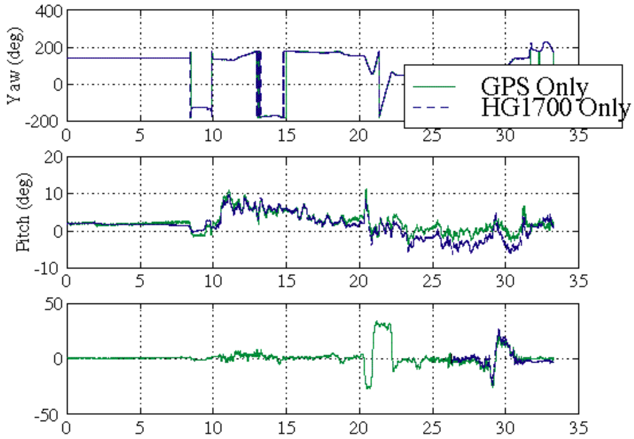

Figure 16. Gyro Coast Capability
(GPS outage occurs at time t = 5).

in all axes 5 minutes after the GPS feedback has been removed. Experience has shown that GPS outages in flight are rare and of a short duration lasting at most a few seconds. Figure 15 clearly shows that the Systron Donner Horizon rate gyros can adequately coast through such short outages. Using a more expensive set of inertial sensors will extend the allowable coasting time during GPS outages; however, using such inertial sensors effectively changes the paradigm: now GPS is being used as a means of initial alignment of the rate gyros and subsequent periodic resets instead of being the primary attitude sensor. That is, the importance of the GPS measurement update is reduced as the quality of the inertial sensors increases. This is shown clearly in Figure 16. This plot is generated from data that was collected during the above mentioned flight tests using a Honeywell HG1700 Tactical Grade Inertial Measurement Unit (IMU). GPS measurement updates are removed after the initial five minutes and the inertial rate gyro outputs are integrated open-loop. Even after 20 minutes without measurement updates, it is seen that the deviation due to gyro drift between the GPS and inertial attitudes is less than 5° along all axes.

Figure 17 shows the integrated output of the system with Systron Donner Horizon gyros. Plots of the integrated output of the HG1700 are identical. The only difference has to do with the way that slowly varying GPS attitude error is rejected. The HG 1700 has the ability to reject GPS errors with time constants on the order of 10's of minutes while the Horizon system can only reject GPS errors with time constants of 10's of seconds.

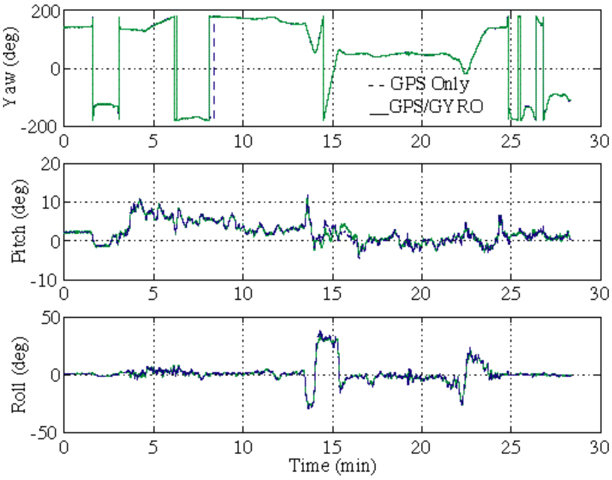

Figure 17. Attitude Time History.

Conclusion

In order to achieve sub-degree accuracies in pitch and roll for the ultra-short baseline attitude systems it was necessary to utilize the known bias algorithm with a common clock receiver. By mapping the inter-antenna phase patterns, the phase error is reduced to 2.5 mm rms. More importantly, the slowly varying nature of the error is removed. This allows the automotive grade inertial instruments to be used effectively to filter out noise and provide higher bandwidth output.

Low cost automotive grade rate gyros adequately filter the high frequency noise in a short baseline GPS attitude system. This filtering enables the use of such an AHRS for pilot-in-the-loop control of aircraft. It has also been demonstrated that in-flight calibration of low cost inertial sensors can cause the sensors to perform at the level of low grade tactical inertial sensors for short periods of time. Tactical grade gyros have been shown to provide extended coasting time with good bias stability. These two inertial sensors span a quality range that are best suited for integration with a short baseline attitude system.

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