Inertially Aided GPS Based Attitude Heading Reference System (AHRS) for General Aviation Aircraft

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BIOGRAPHIES

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ABSTRACT

GPS was used with ultra-short baselines (2-3 carrier wavelengths) in a triple antenna configuration to obtain attitude for General Aviation (GA) aircraft. 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 inexpensive automotive grade rate gyros. The solid state gyros allow coasting through temporary GPS outages lasting 2 minutes with attitude errors less than 6 degrees. 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].

I. INTRODUCTION

Attitude information for small GA aircraft is currently obtained by gyros with spinning rotors. 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 cost more than most small GA aircraft. This research is aimed at bringing glass cockpits to GA at an affordable price.

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]. The concept has been used successfully for aircraft attitude in flight [3,4]. 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).

II. 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 base line 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 3 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

The selection of the attitude computation algorithm is critical to obtaining the minimum error in the GPS solution. Two methods have historically[6] been used: a known line/clock bias and an unknown clock/line 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. The unknown bias can also be eliminated from the equations rather than solved for which is known as double differencing. The single difference technique requires a common clock for all antennas and presumes a constant known (or estimated) line bias. The unknown bias algorithm is used by attitude systems using multiple OEM boards with separate clocks. Both 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:

$$\begin{aligned} &\Delta\phi^i + N\lambda + d\tau = H \cdot B \\ &\mathit{where}: \\ &B = \begin{bmatrix} X \\ Y \\ Z \end{bmatrix} \\ &H = \begin{bmatrix} LOS_X^i & LOS_Y^i & LOS_Z^i \end{bmatrix} \end{aligned} \tag{1}$$

The known bias case is solved as follows in Equation 3.

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

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,

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

The solution to this is identical to the solution for the known bias case given in eq.(2), only now B has 4 elements and H includes a column of 1's.

The nonlinear solution can be thought of as an extension of either the known bias or unknown bias case where the baseline length is now a known constant. The baseline vectors generally have non zero X, Y, and Z components. Applying the baseline constraint in such instances leads to a nonlinear equation. By transferring the baseline vector to a coordinate system aligned with the axis of the vector we can apply the baseline length constraint. We are now left solving for 2 unknowns in the known bias case and three unknowns in the unknown bias case. A further 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.

The elements of the pseudo inverse of H are analogous to the DOPs in the navigation equation. In doing this we can come up with an Attitude Dilution of Precision

Matrix (ADOP). This ADOP matrix is in the units of the LOS vectors and yields accuracies of baseline component estimates, in this case east-north-up directions.

$$ADOP = \begin{bmatrix} \sigma^2_{east} & & \\ & \sigma^2_{north} & \\ & & \sigma^2_{up} \end{bmatrix} = \left(H^T H\right)^{-1} \qquad (4)$$

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 2 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 5 highest satellites. This is shown in Figure 1.

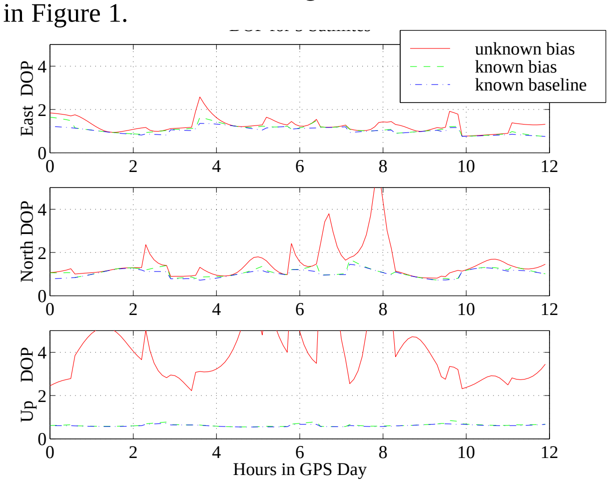

Figure 1. DOP as a function of time and algorithm

Figure 2 shows that the DOP is 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 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.

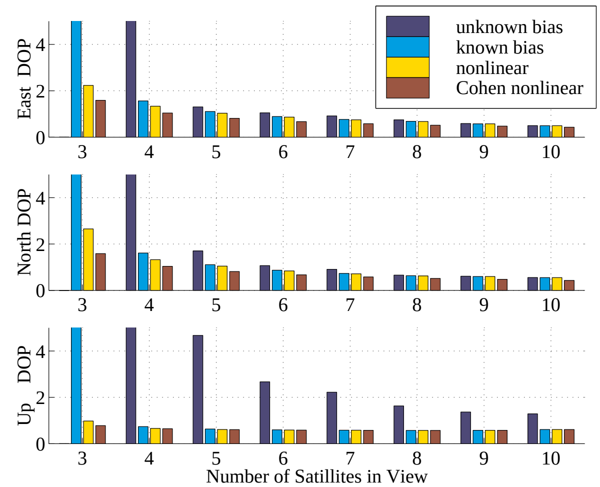

Figure 2. DOP as a function of number of satellites and algorithm

With 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 important to utilize a common clock receiver to allow the short baseline system to attain the required 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.

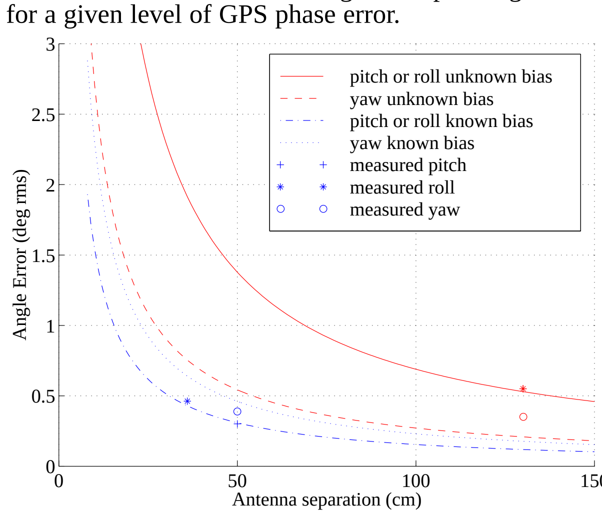

Figure 3. Angle errors vs. baseline length for different algorithms

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.

Figure 4 show the improvement in the pitch attitude solution between the known bias and the unknown bias attitude computation for identical GPS phase data collected with the Trimble Quadrex at a known attitude.

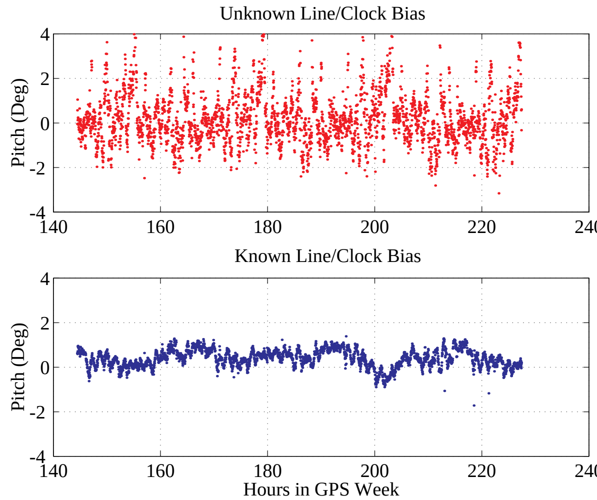

Figure 4. Angle errors for different algorithms

In addition to the unknown bias analysis done at Stanford utilizing the Quadrex data, Seagull Technology has implemented a GPS/inertial attitude instrument architecture that utilizes multiple independent commercial off-the-shelf (COTS) Motorola Oncore GPS boards. Typical results from the Seagull prototype system are also shown in Figures 3 for a run lasting 1000 sec. The data shown were calculated in real time using phase measurements at three antennas statically mounted in an isosceles triangular configuration of width and length 131 cm. For the duration of the plots shown, eight GPS satellites were tracked and used continuously for attitude determination. Integer ambiguity resolution was achieved with a single epoch of data at the start of the run. Figure 3 shows the Seagull system with a 131 cm baseline still has a slightly worse pointing pointing error in the roll axis than the 36 cm system. However, the cost advantages of using multiple COTS GPS receivers for attitude determination make this approach attractive for applications without restrictive size constraints on the dimensions of antenna arrays. In this example, precisions of 0.35° in heading and 0.55° in pitch and roll are achieved using a three-antenna configuration with total width and length of about 1.3 m. This dimension is acceptable in many applications and is especially attractive where yaw angle or heading is of primary interest.

D. GPS Attitude Error Sources

In order to utilize the short baseline concept a complete understanding of all error sources is required. As mentioned earlier, noise in the carrier phase measurement contributes to the error in the attitude solution. These errors become more important in the short base line system.

We break down these errors to those caused by multipath (i.e. signal reflection) and those caused by variation in antennae phase patterns. Ref[9] discusses extensively the phase delay maps for patch antennas. By taking a single phase difference between two antennas we effectively introduce 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 LOS vector from the antenna to the GPS satellites. In the aircraft case we are primarily concerned with antenna phase error effects 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 and this effect can be calibrated out in the same manner as the antenna phase error.

The repeatability of this effect is shown in Figure 5 for 4 days worth of data taken at 2 Hz and averaged over 100 seconds for one satellite. The 4 phase error lines are offset by 1 cm increments for clarity.

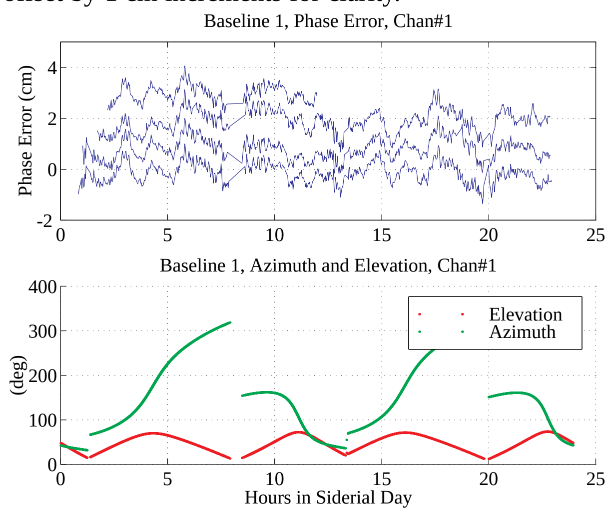

Figure 5. Plot of repeatability of GPS error

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 nature 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 decorrelate in time. This constantly changing attitude has the effect of dither and changing the temporal characteristics of the phase error to a much higher frequency 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.

These deterministic phase errors shown in Figure 5 have been modeled as a function of the azimuth and elevation of the satellite. There is as much as a 1 cm phase error introduced depending on the arrival azimuth and elevation angle. Figure 6 shows that by subtracting out the effect of this error source the rms error in our phase measurements decrease from 5 mm rms. to 2.5 mm rms.

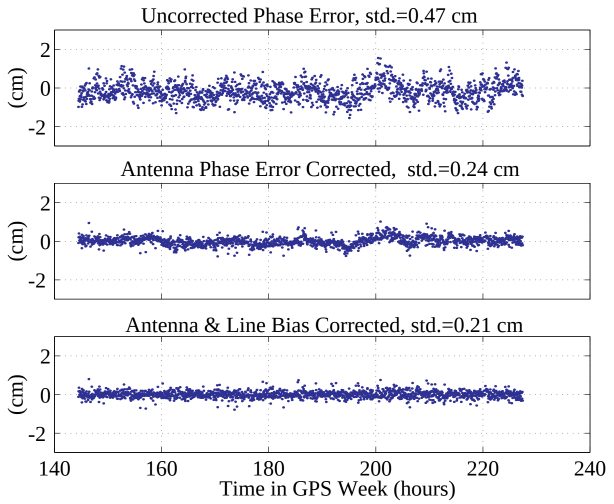

Figure 6. Phase error after antennae phase map subtracted

The second trace in Figure 6 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. Removing this error gives an improvement in phase error from 2.4mm to 2.1mm rms. These improvements in phase error translate directly to an improvement in attitude. The improvement due to phase map and bias corrections are shown in figure 7.

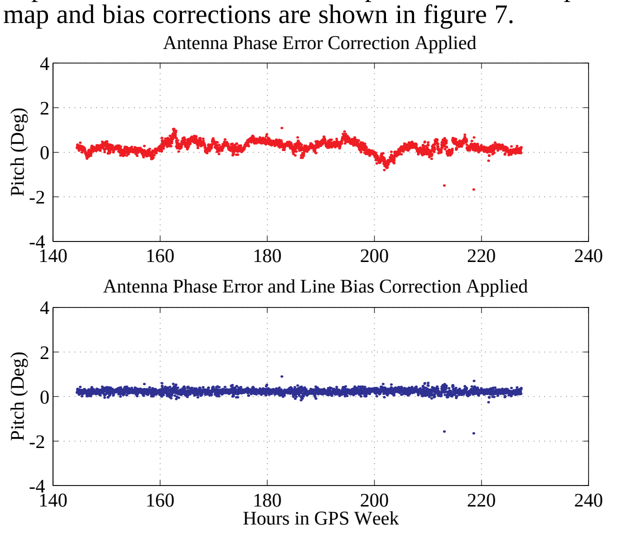

Figure 7. Attitude error after antennae phase map and biases subtracted

The sequential improvements for pitch roll and yaw due to calibration of all error sources are shown in Figure 8 for 6 satellites tracked. The large gains in pitch and roll are obtained by using a common clock algorithm. Following that, incremental 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.

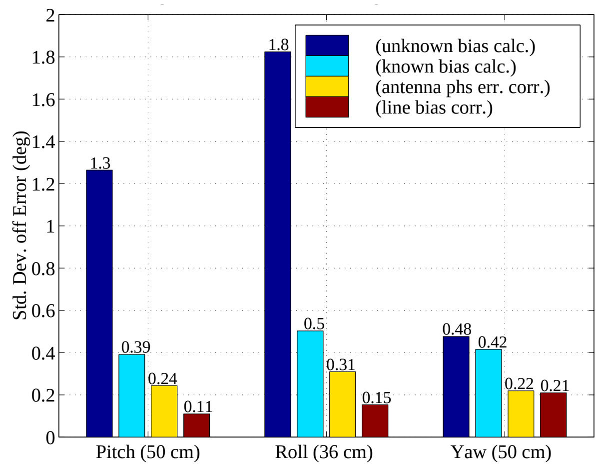

Figure 8. Summary of Pointing Error Improvements

E. Conclusions

By mapping the inter-antennae phase patterns it is possible to reduce the phase error to 2.5 mm rms level. This preliminary factor of two improvement demonstrates the need to calibrate out antenna phase delay and any other LOS dependent phase error.

By further exploring the various GPS error sources it may be possible to approach the theoretical limit for carrier noise on the .5 mm level. This would allow the design of GPS attitude systems with even greater accuracy or shorter baselines than demonstrated here.

In order to achieve sub-degree accuracies in pitch and roll for ultra-short baseline attitude systems it is necessary to utilize a known bias algorithm. This necessitates the development of multiple antenna common oscillator GPS receivers.

III. INSTANTANEOUS INTEGER DETERMINATION

A. Background

Recently, many integer ambiguity resolution methods have been proposed utilizing a variety of techniques. Some use satellite or configuration motion to lock onto a set of integers [10], and others require several minutes of data [11]. Instantaneous methods, which use data from a single epoch, consist of single frequency and those that rely on dual frequency data [12,13]. Many instantaneous single frequency techniques start with a search space and from this determine the best integers based on either a minimal residual or cost function. This can be a very time consuming process and numerous highly refined algorithms have been proposed in order to increase the efficiency [14,15,16]. In order to increase both the accuracy and efficiency of these searches, a priori knowledge of the antennae configuration and reasonable bounds on the attitude can be included [6,17].

The integer ambiguity resolution method used in an 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.

B. Algorithm

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}.

The integer ambiguity that would be present at a given probable attitude can quickly be computed given the single differenced receiver phase measurements, the line of sight matrix, and the actual measured dimensions of the antenna array. First, the measured baselines are transposed to an East North Up coordinate system using the assumed pitch, roll, and yaw. Next, the predicted phase measurements for the given attitude can be computed. The integers are then the rounded difference of the actual and the predicted phase measurements.

Using data taken from the 36cm by 50cm antenna configuration, this method has shown a high degree of accuracy when the predicted attitude was close to the actual attitude. The method is 100% reliable as long as the difference between the predicted and actual pitch, roll, and yaw angles are each less than 6 degrees.

Since the integer search ends as soon as the correct integers are found, the attitude space is analyzed in order of maximum likelihood. Recognizing that the initial integer search will most likely be performed on level ground, roll and pitch both start at zero degrees and then bank up then down 10 degrees at a time until pitch reaches 20 degrees and roll reaches 40 degrees. Since the desired integers can be obtained 100% of the time for all angles less than 6 degrees, the correct integers will always be checked as long as the magnitude of pitch and roll are less than 26 and 46 degrees respectively during the integer search. These constraints on the attitude were chosen since they place little actual constraint on the motion of a GA aircraft and yet greatly cut down on the volume of the search space [19].

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 effectiveness of each test at weeding out incorrect solutions is shown for the combination of all baselines in figure 9. 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.

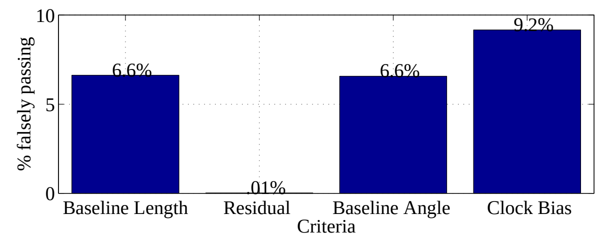

Figure 9 Incorrect integer combinations falsely satisfying criteria for all three baselines

The overall reliability of the system is dependent on the number of visible satellites. Figure 10 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.

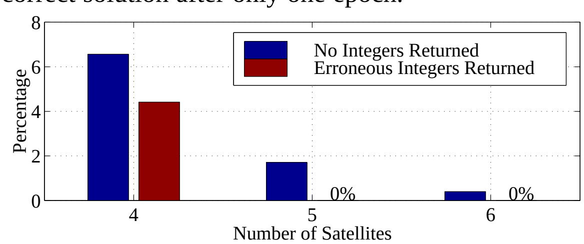

Figure 10 Integer Search Reliability

IV. INERTIAL SENSORS

A. Background

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 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[20,21,22], however, 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.

B. Problem Statement

The focus of this research has been to evaluate the benefit added by automotive grade rate gyros to a short baseline GPS attitude system. Specifically, the issues that need to be addressed are: 1) How stable are these automotive grade rate gyros and will they permit coasting through a momentary GPS outage? An acceptable coasting time would be the time required for the attitude errors due to gyro drift to be greater than 6 degrees. If the attitude errors are greater than 6 degrees the integer search algorithm discussed earlier can not be used for GPS integer initialization. 2) Can these sensors filter GPS attitude noise? It was noted earlier that the attitude errors observed in a GPS short baseline attitude system are colored and have long time constants. Since the attitude solutions obtained from GPS are blended with the inertial solution using a Kalman Filter, filtering these GPS attitude errors will either require a gyro with exceptional stability or a deterministic and precise gyro error model.

C. Simulations

To answer the above questions, simulations were performed. These simulation used a rate gyro error model and a GPS attitude error model. The gyro error model was for the Systron Donner Horizon Gyro. The approximate price for these gyros is $700 for a single unit, $300 for 10-500 units, and $70 if purchased at a rate of at least 3000 units per year. This gyro is of very similar construction to gyros currently being installed in automobiles for turn rate sensing. Its sensing element is a vibrating tuning fork which deflects in proportion to the angular rate applied. As normally is the case, the information on the data sheet [25] is not detailed enough to allow construction of a good error model. To be able to do so and better characterize the Horizon’s performance, experiments were performed to determine the effects of temperature on the bias drift, the effect of temperature on scale factor and the effect of accelerations on the bias drift. The objective of these experiments was to see whether any of these errors were deterministic. From these tests it was concluded that the effect of ambient temperature changes on scale factor is minimal (i.e., less than 2.6% change over the 0 C and 60 C range) and, therefore, excluded from the error model. The effect of linear accelerations was also noted to be less than the gyro output noise and, therefore, excluded from the error model. Finally, it was concluded that *short term* (15 min) effect of ambient temperature changes on the gyro output was negligible and, as such, was not specifically accounted for in the error model. To assess, the long term bias stability of these gyros, the output from the gyros was sampled and recorded for six hours. The output data from these tests was used to construct an allan variance chart using the procedures outlined in Refs [26] and [27].

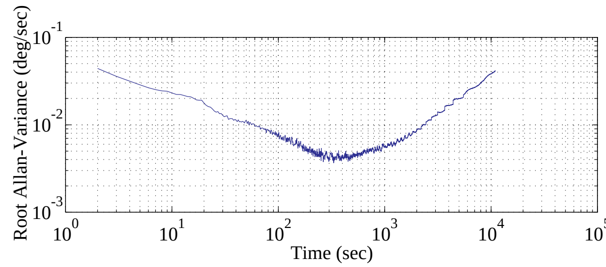

Figure 11 Allan Variance of Horizon

A representative allan variance chart for the Horizon is shown in Figure 11. The allan variance for these gyros shows that for roughly the first 300 sec, the output error is dominated by white sampling noise. Therefore, if the rate output from these gyros is integrated to give angle, the primary error would be angle random walk in this time period. Thus, for at least the first 300 sec filtering will minimize the error in the output. However, the gyros exhibit a long term instability which tends to dominate the output error after about 300 sec. The initial upward slope of approximately +1/2 indicates that the output error in that time period is predominately driven by an exponentially correlated process with a time constant much larger than 300 sec or a rate random walk. Accordingly, for the simulations below, the model for the gyro rate bias was assumed to be a first order markov process with a long time constant (1000 sec). A rate random walk was not selected because it represents a physically unrealistic process.

A comparison was also made between the Horizon and the Andrew Autogyro which is a Fiber Optic Gyro (FOG) and sells for approximately $800. Based on an allan variance chart for this gyro, which is shown in Figure 12, a reasonable error model for short periods of time for this gyro is a random constant (i.e., a constant corrupted by sampling noise).

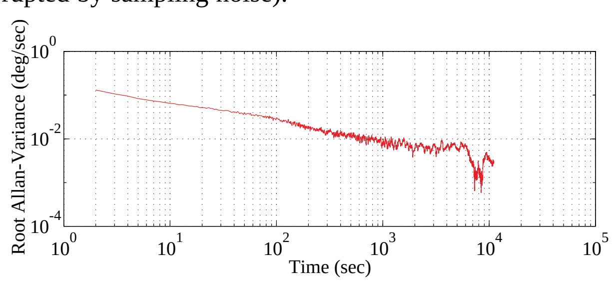

Figure 12 Allan-Variance for Andrew Autogyro

The GPS error model assumed that the GPS attitude noise was white with a 1σ of 0.25 deg on the yaw and pitch axes and 0.3 deg on the roll axis. The simulation included three solid-state rate gyros mounted orthogonally. The gyros were sampled at 20 Hz and their output was integrated to yield Euler angles. At 2 Hz, a GPS attitude solution was computed. The GPS and gyro solutions were blended using a Kalman Filter .

The first simulation was performed to asses the length of time that one can coast through a GPS outage. In the simulation below both sets of gyros are operating in conjunction with GPS for 60 seconds to allow estimation of biases. At that point, the GPS attitude solution was turned off and the gyros were allowed to coast. Figure 13 shows the coast time for the two gyros.

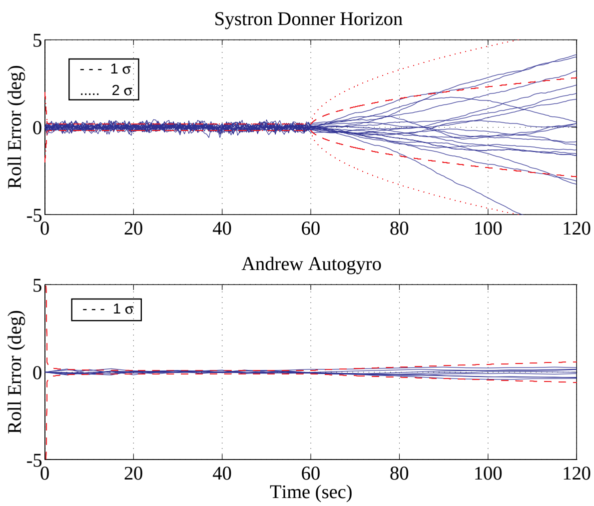

Figure 13 Simulation results. Coast time for Systron Donner Horizon and Andrew Autogyro

For the Systron Donner Horizon it is seen that the 1σ envelope shows a growth of angle error at the rate of approximately 2.5 degrees per minute. A more conservative bound is the 2σ which is seen to grow at a rate of roughly 6 degrees per minute. Thus, there is a 1 to 2 minute coasting time allowed by the Systron Donner Horizon. A similar analysis of the performance of the Andrew Autogryo shows 1σ error growth of approximately 0.5 degree per minute. These results also provide the answer to the second question posed in the problem statement. That is, these gyros will not adequately filter the GPS attitude noise that is present before the various attitude error corrections are applied.

V. INTEGRATED SYSTEM

A. Introduction

The rate gyros and GPS receiver were combined in an integrated short baseline GPS-inertial AHRS system.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 14.

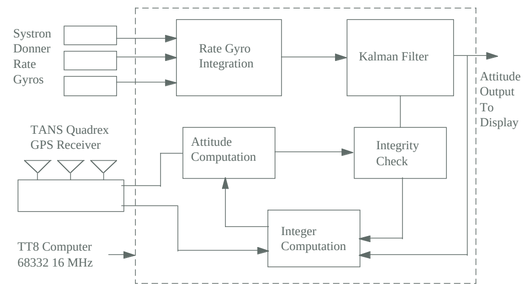

Figure 14. Block Diagram of the Integrated GPS-Inertial AHRS.

Flight testing was performed on a Beechcraft Queen Air. The Queen Air utilized a 36 x 50 cm baseline configuration shown in Figure 15.

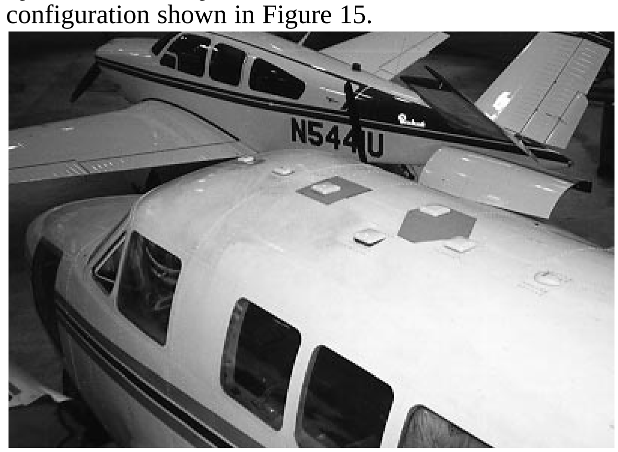

Figure 15. Antenna Installation on the Queen Air Test Airplane

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 re-initialization; however, this was rarely required. This integer resolution performed robustly, never computing false integers during any of the flight tests.

C. Integration of Inertial Sensors

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.

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 input to the computation of the gains (i.e., the process noise and measurement noise) were obtained from experimental noise measurement and the gyro error models discussed in the previous section.

The filtering and integration algorithms were performed using a TattleTale™ Model 8 by Onset Computer. The TattleTale™ Model 8 consists of a Motorola 68332 Processor with 8 12 bit A to D lines and 16 Digital I/O lines running at 16MHz. The algorithms were written in C and compiled using the MotoCross™ 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. Through out the flight test period the display was evaluated for latency and correlated 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 observable jitter or lag by the pilots.

The estimates of gyro biases stabilized after about 4 minutes from power up. The stability of these bias values is what allows the system to accurately estimate attitude when the GPS feedback is removed. As a demonstration of the coasting capability of the system, feedback from the GPS attitude solution was deliberately turned off for an extended period of time. Figure 16 shows the deviation between the gyro integrated attitude solution and the GPS attitude solution during this outage. The bottom plot is a blow up of the period when GPS attitude was reintroduced. There is less than a 2 degree error in pitch and roll in the 10 minute period of time that the GPS signal has been removed, this is in spite of constant maneuvering in the pitch axis shown in the upper graph. This performance is significantly better than predicted by our theoretical model of the gyros.

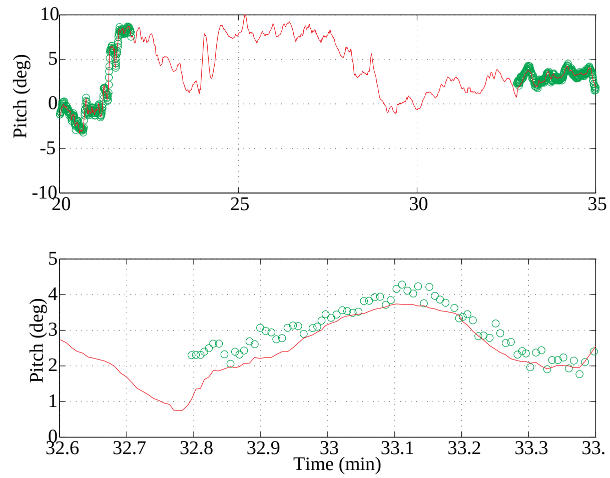

Figure 16. Gyro Coast Capability

A comparison of the raw GPS attitude solution with the integrated gyro solution, shows no lag in the gyro smoothed attitude solution displayed. Pilots who flew the Beech Queen Air test aircraft using the attitude information displayed by this system reported they had no difficulty controlling the aircraft using only the attitude information generated by the integrated GPS and gyro system. In fact during a period of 60° angle of bank turns the vacuum driven Artificial horizon was observed to precess approximately 10° while the combined short baseline GPS inertial AHRS displayed no variation from the actual horizon.

VI. CONCLUSIONS

A promising way of making GPS-based attitude determination system inexpensive is by reducing the spacing of the antennas to the order of two to three carrier wavelengths. Reducing the spacing of the antennas has the significant advantage of allowing installation of the antenna array on the top of the fuselage. This minimizes the potential of blocking the line of sight from the antennas to the satellites by the aircraft structure, minimizes the need for long wire runs, and eliminates the need for a fourth antenna to sense wing flexing. These factors minimize the installation costs of such a system. This is a significant concern for an AHRS aimed at GA aircraft. To obtain the sub-degree accuracy required for a GA AHRS utilizing short baselines it is necessary to utilize a common clock GPS receiver and attitude algorithm. To date this has only been accomplished using expensive receivers such as the TANS Quadrex. Ongoing research at Stanford is aimed at developing an inexpensive alternative. The inherent bandwidth limitation of GPS will always require some level of inertial aiding in high dynamic and high bandwidth operations. Automotive grade rate gyros can allow coasting through a 1 to 2 minute GPS outage with attitude errors between 2-6 degrees.

ACKNOWLEDGEMENTS

The authors gratefully acknowledge the many friends and colleges at Stanford University who have helped us in this effort. We also gratefully acknowledge the help of Dr. Zeev Berman and test pilots Sky, and Kevin McCoy from Sky Research Incorporated.

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