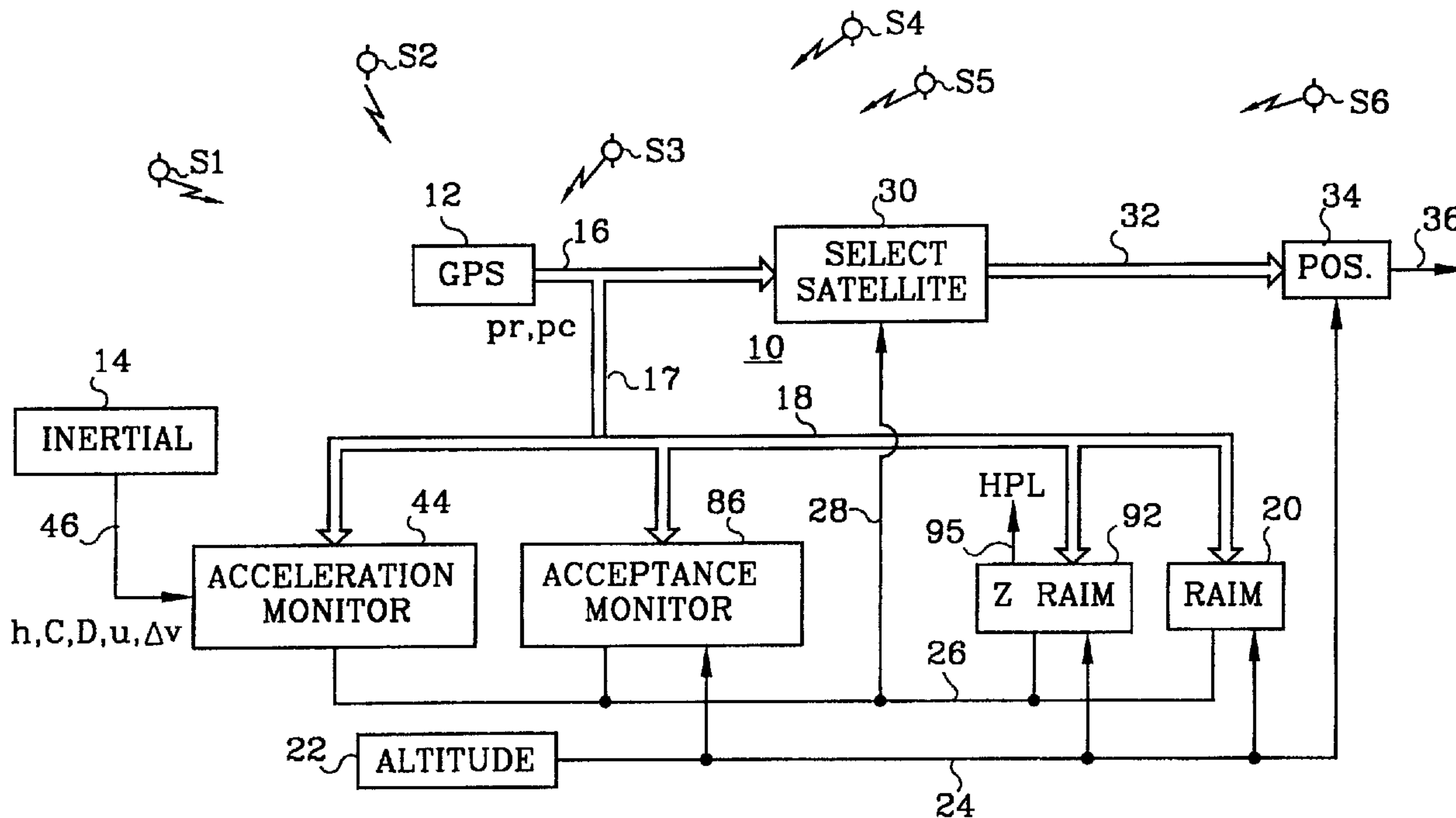




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(54) Titre : MONITEUR DE DERIVE DE SATELLITES GPS  
 (54) Title: GPS SATELLITE DRIFT MONITOR



(57) Abrégé/Abstract:

A system to determine when a GPS satellite signal has begun to drift prior to acquisition by producing a discriminator which is used to determine the satellite error bound and the satellite error bound is used to determine the drift error bound indicative of an existing drift in the satellite.



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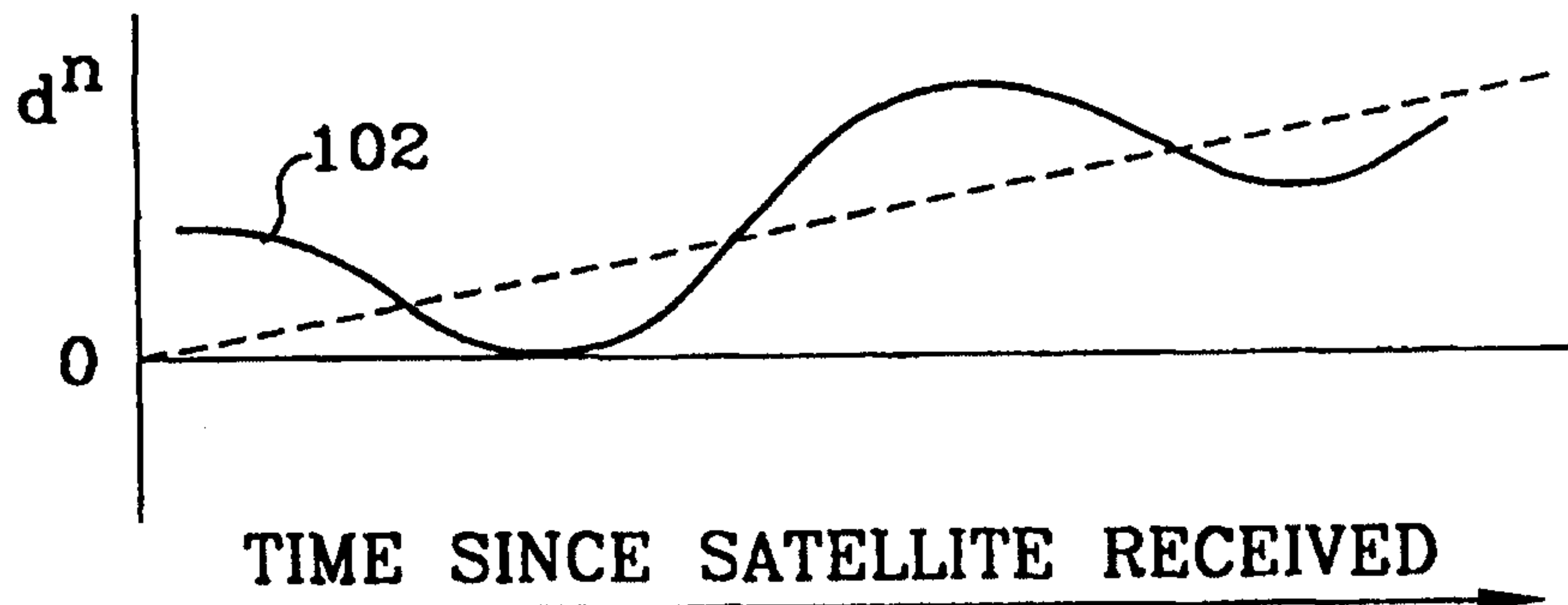
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<p>(21) International Application Number: PCT/US99/13762 (22) International Filing Date: 17 June 1999 (17.06.99) (30) Priority Data: 09/118,297 17 July 1998 (17.07.98) US (71) Applicant: HONEYWELL INC. [US/US]; Honeywell Plaza, Minneapolis, MN 55408 (US). (72) Inventor: BRENNER, Mats, A.; 4275 Deerwood Lane, Ply- mouth, MN 55441 (US). (74) Agent: UNGEMACH, Charles, J.; Honeywell Inc., Honeywell Plaza - MN12-8251, P.O. Box 524, Minneapolis, MN 55440-0524 (US).</p>	<p>(81) Designated States: CA, JP, European patent (AT, BE, CH, CY, DE, DK, ES, FI, FR, GB, GR, IE, IT, LU, MC, NL, PT, SE).  <b>Published</b> <i>With international search report.</i></p>	

(54) Title: GPS SATELLITE DRIFT MONITOR

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A system to determine when a GPS satellite signal has begun to drift prior to acquisition by producing a discriminator which is used to determine the satellite error bound and the satellite error bound is used to determine the drift error bound indicative of an existing drift in the satellite.



**GPS SATELLITE DRIFT MONITOR**  
**BACKGROUND OF THE INVENTION**  
**FIELD OF THE INVENTION**

5 The present invention relates generally to aircraft navigation and more particularly to a system employing a Global Positioning System (GPS) and an Inertial Reference System (IRS) to allow for an early determination of when a satellite signal becomes undependable and when a newly acquired satellite signal is undependable.

**DESCRIPTION OF THE PRIOR ART**

10 In the prior art, GPS systems have been used to determine aircraft position by receiving signals from a plurality of satellites. The signals each have information as to the position of the satellites and the time of transmission so that the GPS receiver, on the aircraft, can calculate its own position. Since there are four variables (position in 3 axes and time), signals from at least 4 satellites are necessary for a determination of receiver  
15 position. If there are at least five satellites having good geometry each subset of four signals can be used for positioning and they can be compared with each other to determine if one of the signals is in error (Fail safe). If there are at least six satellites in view, then, when there is a faulty signal, it is possible to use the groupings to determine which signal is in error (Fail operational). This procedure has heretofore been  
20 accomplished with a system identified as RAIM (Receiver Autonomous Integrity Monitor) which is described in a Honeywell Inc. article titled "Implementation of a RAIM Monitor in a GPS Receiver and an Integrated GPS/IRS" found in a publication entitled Global Positioning System Volume V published by the Institute of Navigation, Alexandria VA.

25 Since RAIM cannot detect an erroneous satellite signal with only four satellites or identify which signal is in error with only five satellites, the pilot cannot rely on the position information he receives in these situations and the GPS input must be disregarded. Accordingly, a need has arisen to provide a system which can produce a reliable output when only four satellites are in view.

30 In a patent application entitled "Navigation System with Solution Separation Apparatus for Detecting Accuracy Failure" Serial Number 08/712,232, filed September 11, 1996 and assigned to the assignee of the present invention, one solution to this problem is disclosed. More particularly, that application discloses an integrity monitor and positioning algorithm which receives signals from a plurality of remote transmitters,

detects failures and determines a protection limit (i.e. the furthest statistical distance by which the position determined by the apparatus will have an almost certain probability of being bounded in a situation where one satellite is in error). This positioning algorithm preferably uses Kalman filtering techniques and incorporates inertial reference data therein to enhance the detection capability. While this system allows confident use of signals from four satellites for a limited time, the protection limit provided by the Kalman filter bank will sometimes exceed the required protection for the most stringent phases of flight (non-precision approach and terminal area navigation) and is therefore not sufficiently available. Furthermore, the bank of Kalman filters is computationally expensive and complex to implement in other equipment than the inertial reference unit. Therefore, a need has arisen for a system which is less complex and expensive, which can more readily be incorporated into equipment of suppliers and which can detect satellite signal errors at an early point in the operation.

In a copending application entitled GPS Signal Fault Isolation Monitor serial number 09/118,046 filed on even date herewith by the present inventor and assigned to the assignee of the present invention, these needs were addressed by providing a new acceleration monitor that produces a first acceleration output along a predetermined axis from the GPS signals and a second acceleration output along the same predetermined axis from the inertial reference units (IRU's).

Although the IRU outputs are not very accurate for determining position, the measurement of acceleration along a predetermined axis is quite accurate. Accordingly, the IRU acceleration signal is compared to the GPS acceleration signal along the predetermined axis. When there is an error greater than a predetermined amount, the faulty satellite signal is identified and can be eliminated from the calculations. Thus only four satellites are necessary for continued operation. In the case of a signal drift, which may be considered as a ramp output, the initial change at the beginning of the ramp causes an acceleration transient which is detected by the acceleration monitor so that early identification of this condition is made. This feature is claimed in the above identified copending application. However, detection of signal drift in a satellite which has begun drifting prior to being seen by the GPS receiver is not detected by the new acceleration monitor since the acceleration transient occurred previously.

### **SUMMARY OF THE INVENTION**

The present invention solves this problem by providing a novel new satellite drift monitor sometimes referred to herein as a zero crossing RAIM (or Z-RAIM) monitor that detects drift in a newly acquired satellite by producing a discriminator  
 5 which is used to determine the satellite error bound (the bound or limit associated with a predetermined confidence level) and the satellite error bound is used to determine the drift error bound, which provides an indication that drift has occurred.

Both the acceleration monitor and the new satellite drift monitor of the present invention will be explained in detail below but the acceleration monitor is claimed in the  
 10 above mentioned copending application and the new satellite drift monitor is claimed herein.

### **BRIEF DESCRIPTION OF THE DRAWINGS**

FIG. 1 shows a schematic block diagram of the system utilizing the present  
 15 invention;

FIG. 2 is a block diagram showing the operation of the acceleration monitor of the above mentioned copending application; and,

FIGS. 3A and 3B are graphs showing how the new satellite drift monitor of the present invention operates.

20

### **DETAILED DESCRIPTION**

Referring to Figure 1 wherein a system 10 for use in aircraft navigation is shown comprising a GPS receiver 12 and an inertial reference unit 14. GPS receiver 12 is of a type well known in the art which operates to receive signals from a plurality of  
 25 satellites, e.g. S1 - S6 in Figure 1, indicative of their positions and of the time of transmission. While 6 satellites have been shown, it will be understood, that more or less than 6 satellites may be in view of the GPS receiver 12 and that the number may change over a period of time. GPS receiver 12 operates on the signals from satellites S1 - S6 to determine the position of the receiver and to produce the GPS code based pseudo  
 30 range measurements,  $pr$ , to each satellite. Also, by tracking and counting the cycles (including fractions of cycles) in the carrier from each satellite, the GPS receiver 12 provides the accumulated carrier count,  $pc$ , which can be used to obtain the change of pseudo range over a time  $\Delta t$  according to the equation:

$$\Delta pr = \Delta pc = pc(t + \Delta t) - pc(t) \quad (1)$$

The change of the pseudo range obtained by differencing the carrier count is accurate to within centimeters ( 0.01 meter ) while the pseudo range change formed by differencing the pseudo range measurements is accurate to within 2 to 5 meters. The code based  
 5 change in pseudo range  $\Delta pr$  , the carrier based change in pseudo range,  $\Delta pc$ , or a combination thereof , referred to as a smoothed code measurement are used in the present invention as will be described below.

The pr and pc signals are provided as outputs represented by a path 16 from GPS receiver 12 and, via paths 17 and 18 to a standard RAIM shown in box 20 which also  
 10 receives a pressure altitude signal from an altitude sensing means 22 via a path 24. In the event that an undependable satellite signal is encountered and detected by the standard RAIM 20, a deselect signal is presented from RAIM 20 as an output represented by a path 26. This occurs if there are at least 6 satellites in view with good geometry, in which case, groups of five satellite signals can be compared so as to  
 15 determine which one, if any, of them is faulty. If one is so identified, then this information is passed (via path 26 and a path 28) to a satellite selection function represented by a box 30 which also receives the GPS pr and pc from path 16. The satellite selection function 30 eliminates the faulty satellite signal and passes the remaining valid signals as an output represented by a path 32 to a Position function  
 20 represented by box 34. Position function 34 also receives a pressure altitude signal from an Altitude Sensing means 22 via path 24 and produces an output indicative of the aircraft position (as represented by a path 36) to downstream equipment such as indicators or flight management systems (not shown). If there are only 5 satellites in view with good geometry, then groups of four satellite signals can still be compared  
 25 against at least one other satellite signal so as to determine if any of them are faulty but since the faulty satellite cannot be identified no de-selection can be performed. In such a case the downstream equipment and the pilot will be notified that the GPS signals should not be used (this signal is not shown in figure 1).

In accordance with the invention of the above mentioned copending application,  
 30 the signal output represented by path 16 is provided via paths 17 and 18 to an acceleration monitor as represented by a box 44. The acceleration monitor 44 also receives an input from the inertial reference unit 14 (which is also well known in the art and comprises a plurality of gyros and accelerometers) that produces outputs as

represented by a line 46 indicative of  $h$  (altitude),  $C$ , (attitude matrix),  $D$  (latitude, longitude, wander angle matrix),  $v$  (velocity) and  $\Delta v$  (change of velocity). A minimum of 3 gyros and 3 accelerometers are employed, but to ensure fail safe or fail operational operation and high reliability, two or three redundant systems are preferably employed.

5 The  $\Delta v$  signal from path 46 and the  $pr$ ,  $pc$  signals from paths 16, 17 and 18 are used by the Acceleration Monitor, 44, of the present invention by an operation which will be better understood with reference to Figure 2.

In Figure 2, the Acceleration Monitor 44 is shown receiving the inertial signals  $h$ ,  $C$ ,  $D$ ,  $v$ , and  $\Delta v$  over a path shown by arrow 46 which corresponds to figure 1 and receiving the GPS input,  $pr$  and  $pc$ , over a path shown by arrow 40 which corresponds to

10 Figure 1.

The inertial signal  $\Delta v$ , consists of filtered velocity increments at a 10 - 60 Hz rate which are provided to a function box 54 labeled "Transform to L frame". Function box 54 also receives the attitude input  $C$ , shown by arrow 46 and transforms the filtered

15 velocity increments to the local vertical frame,  $L$ , and outputs the transformed increments to a function represented by a box 58 labeled "Form 2nd 1 Hz Time Difference". In box 58 "double position difference signals" (i.e. signals representing the difference in the change of position in the current  $\Delta t$  interval and the change of position in the previous  $\Delta t$  interval) along each axis are formed by integrating the inertial

20 acceleration (i.e. the transformed high rate velocity increments). The output of the function 58 reflects an inertially measured acceleration which contains earth rotation induced acceleration components that will not be present in the acceleration derived from the GPS signal (to be explained below). Accordingly, the output of function 58 is provided as input to a function represented by a box 60 labeled "Remove Earth Rotation

25 Induced Acceleration" where the undesired earth rotation components are removed. Finally, since the acceleration derived from the GPS signal will only be available in the direction of the satellite (along the line of sight), the inertially based reference signal from function 60 is presented to a function represented by a box 62 labeled "Project Along Line Of Sight" which also receives a signal,  $LOS$  (unit vector along the line of

30 sight to the satellite) shown by arrow 64 and the output of function 62 is an inertial double position difference projected along the line of sight to the satellite. This signal is presented to a function represented by box 66 labeled "Discriminator & time removal transformation" where the acceleration monitor discriminator is formed.

With respect to the GPS signal, the pseudo range measurements  $pr$  and accumulated carrier cycle counts  $pc$  for all tracked satellites are presented from path 18 to a function represented by a box 70 labeled "Form Range Difference" which also receives an initial GPS position,  $r_{init}$ , shown on an input represented by arrow 72 which is the GPS position at the time of initialization and received from the position function 34 in figure 1. As mentioned above, the signal,  $pc$ , consists of accumulated carrier cycles (each cycle corresponds to about 0.19 m position change in the usual satellite signal) and the signal  $pr$  are code based pseudo range measurements. The signals  $pr$ ,  $pc$  or any combination thereof (smoothed measurements) include the motion of the satellite and function 70 operates to remove the satellite motion component and provide the result to a function represented by a box 74 labeled "Form 2nd 1 Hz Time Difference". The double difference signal (i.e. the difference in the change of cycle count (or smoothed pseudo range) in the current  $\Delta t$  interval and the change of cycle count (or smoothed pseudo range) in the previous  $\Delta t$  interval) is formed by function 74 and an output which represents the acceleration of the GPS receiver (but which also contains components that relate to the change in the line of sight vector at the current position and the line of sight vector at the initial reference position) is provided to a function represented by a box 76 labeled "Line of Sight Compensation". Now, the satellite acceleration is projected differently when seen from the current position versus when seen from the initial reference position. Accordingly, the output formed by function 76 is one where the components related to the line of sight are removed. The output from function 76 is presented to a function represented by a box 78 labeled "Satellite Acceleration Compensation". The cycle count includes the motion of the satellite and function 78 operates to remove the satellite motion acceleration component. The final result is a GPS signal based double position difference along the line of sight and this is provided to the function represented by box 66. The "Discriminator & Time Removal Transformation" function 66 operates on the GPS and Inertial Reference acceleration signals from functions 78 and 62 respectively to form a discriminator by differencing the two signals. This function also subtracts an average (over all of the satellites) of all discriminators from each satellite specific discriminator thereby eliminating the receiver clock offset. This signal is presented to a function represented by a box 80 where the discriminator output is averaged over time and compared to a fixed threshold value to produce a deselected signal on a line 82 which is used to

deselect any satellite whose acceleration exceeds the threshold. It should be noted that all of the functions performed by function boxes described above can be performed by a computer program with each function being readily programmable by one having ordinary skill in the programming art.

5 In Figure 1, the deselect signal on line 82 of figure 2 is shown being provided via lines 26 and 28 to the Select Satellite function 30 and the deselected satellite signal will not proceed to the Positioning function 34.

10 An Acceptance Monitor 86 is shown in figure 1 receiving the GPS signal over paths 16, 17 and 18 and further receiving a pressure altitude signal from the Altitude box 22 over path 24. The Acceptance Monitor 86 is used in the art to detect GPS signals which are obviously incorrect because, for example the satellite pseudo range is far out of reasonable bounds. When such an erroneous satellite is received, the Acceptance Monitor 86 produces a deselect signal to path 26 and via path 28 to the Select Satellite function 30 which then operates to prevent the erroneous signals from  
15 being used by the Positioning function 34.

In accordance with the present invention, a new satellite drift monitor", or "Z RAIM" is shown as a box 92 which receives the GPS signal over paths 16, 17 and 18 and the pressure altitude signal over path 24. Z RAIM 92 provides an output 95 indicative of the Horizontal Protection Limit, HPL, to the pilot or to downstream aircraft  
20 equipment such as the Flight Management System (not shown) over a path 95

The new satellite drift monitor or Z RAIM 92 of the present invention operates to determine if a drift has already begun when a signal from a new satellite is first acquired. This is explained with reference to Figures 3A and 3B.

25 Assuming that there are N satellites with good geometry, the RAIM discriminator as known in the art for the nth satellite, is formed by the equation:

$$d^n = \sum_{k=1}^N b_k^n \Delta pr_k \quad (2)$$

30 Where  $b_k^n$  is a well known satellite geometry dependent coefficient as seen in the above referred to publication, N is the number of satellites, k indicates the kth satellite and  $\Delta pr_k$  is the difference between the measured pseudo range (or smoothed pseudo range) and the predicted pseudo range for the kth satellite. Due to selective availability (SA, a

deliberate noise signal superimposed on the output of the GPS by the DOD), the discriminator will vary as seen in Figure 3A

The error bound,  $\varepsilon$ , defined to have a predetermined confidence level of  $1-p_{md}$  (which is typically 99.9%), is defined by the equation:

$$5 \quad \varepsilon = |d^n| + K_{md} \sigma_{SA} / |b_n^n|. \quad (3)$$

Where  $|d^n|$  is the absolute value of the discriminator  $d^n$ ,  $K_{md}$  is the statistical sigma number corresponding to the missed detection probability  $p_{md}$ ,  $\sigma_{SA}$  is the selective availability noise sigma and  $|b_n^n|$  is the absolute value of the satellite geometry dependent coefficient in equation (2) where the upper and lower indices are the same and equal to the satellite index,  $n$ .

Based on the properties of the SA noise, the discriminator will cross zero or reach a minimum value from time to time i.e. at least every 5 minutes. As the discriminator crosses zero or reaches the minimum absolute value  $|d^n|_{min}$ , the error in the estimate of the  $1-p_{md}$  error bound is at a minimum:

$$15 \quad \varepsilon_{min} = |d^n|_{min} + K_{md} \sigma_{SA} / |b_n^n|. \quad (4)$$

and the estimated drift rate,  $r$ , at a  $1-p_{md}$  confidence level is

$$r = \varepsilon_{min} / t \quad (5)$$

Where  $t$  is the time since the satellite was first received.

As  $t$  increases, more accurate drift error bounds are obtained for each encountered minimum  $|d^n|_{min}$ . Note that if the drift error bound is calculated as a minimum recursively according to the equation

$$r(t) = \min(r(t-\Delta t), \varepsilon/t) \quad (6)$$

where  $\Delta t$  is the time step, the result will be the same.

The satellite error limit, SEL, is the minimum of the current satellite error bound  $\varepsilon$  and the previous satellite error limit with the estimated  $1-p_{md}$  drift error bound added. This is recursively determined by the equation:

$$25 \quad \text{SEL}(t) = \min(\varepsilon, \text{SEL}(t+\Delta t)) \quad (7)$$

Where  $\Delta t$  is the time step.

The horizontal protection limit, HPL (a required output in avionics equipment) associated with satellite  $n$ , can be calculated as:

$$\text{HPL}_n = t_{\text{hn}} \text{SEL} + K_{\text{md}} \text{rf} \sigma_{\text{SA}} \quad (8)$$

Where  $K_{\text{md}}$  is the statistical sigma number corresponding to the missed detection probability  $p_{\text{md}}$ ,  $\text{rf}$  is a reduction factor less than 1 and  $t_{\text{hn}} = \sqrt{t_{\text{n1}}^2 + t_{\text{n2}}^2}$  where  $t_{\text{n1}}$  and  $t_{\text{n2}}$  are elements of the least square solution matrix,  $T$ , that is well known in the art. Even if  
 5 no action is taken to remove a satellite with an initial drift, the horizontal protection limit,  $\text{HPL}_n$ , associated with this satellite, will always bound the error as long as the drift is constant i.e. assure a fail safe operation. If it is not constant, the Acceleration Monitor 44 of the above mentioned copending application should detect an acceleration. To  
 10 achieve fail operation capability, a de-selection of a new satellite via paths 22, 23 and function 24 in figure 1 should be performed if the rate,  $r$ , exceeds a predetermined time dependent threshold. Only newly acquired satellites need to be monitored this way since satellites that are already used for positioning, are being monitored by the Acceleration Monitor 44.

The use of boxes and paths herein has been for purposes of explaining the  
 15 operation of the preferred embodiment and it should be understood that these boxes and paths may be implemented either by software or hardware or a combination of both.

It is therefore seen that I have provided a drift monitor which can detect when a newly acquired satellite has begun to drift prior to acquisition. Many alternate  
 20 embodiments will occur to those having skill in the art and the invention disclosed herein may be embodied in other specific forms without departing from the spirit of the present invention. Accordingly, the embodiments described herein are to be considered in all respects illustrative and not restrictive and the scope of the invention is to be indicated by the appended claims, rather than by the foregoing description, and all  
 25 changes which come within the meaning and range of equivalency of the claims are intended to be embraced therein.

**CLAIMS**

1. Apparatus for use with a GPS receiver which receives information from a plurality of satellites entering and leaving the field of view of the receiver, comprising:  
 5 a new satellite drift monitor receiving said information from satellites entering the field of view and determining a discriminator which will vary to a minimum value from time to time, said drift monitor utilizing the minimum values to determine an upper bound on the drift error at a predetermined confidence level.
- 10 2. Apparatus according to claim 1 where the upper bound on the drift error is compared to a time variable threshold to produce a deselect signal indicative of a drift in a new satellite entering the field of view.
- 15 3. Apparatus according to claim 1 where a satellite error limit, SEL, is recursively calculated at a  $1-p_{md}$  confidence level based on the drift error bound and the current satellite error bound and where a horizontal protection limit, HPL, is calculated from the SEL based on the solution matrix T.
- 20 4. Apparatus according to claim 2 where the satellite error bound is calculated according to the equation
- $$\varepsilon = |d^n| + K_{md} \sigma_{SA} / |b_n^n|.$$
- Where  $|d^n|$  is the absolute value of the discriminator  $d^n$ ,  $K_{md}$  is the statistical sigma number corresponding to the missed detection probability,  $p_{md}$ ,  $\sigma_{SA}$  is the selective availability noise sigma and  $|b_n^n|$  is the absolute value of the satellite geometry
- 25 dependent coefficient parameter wherein the upper and lower indices are the same and equal to the satellite index, n and the drift error bound is calculated as the minimum of the previously calculated drift error bound and the currently observed drift error bound is determined recursively according to equation
- $r(t) = \min ( r(t-\Delta t), \varepsilon / t )$  where t is the time since the new satellite was first received
- 30 and  $\Delta t$  is the time step.

5. Apparatus according to claim 3 where the satellite error bound is calculated according to the equation

$$\varepsilon = |d^n| + K_{md} \sigma_{SA} / |b_n^n|.$$

Where  $|d^n|$  is the absolute value of the discriminator  $d^n$ ,  $K_{md}$  is the statistical sigma number corresponding to the missed detection probability,  $p_{md}$ ,  $\sigma_{SA}$  is the selective availability noise sigma and  $|b_n^n|$  is the absolute value of the satellite geometry dependent coefficient parameter wherein the upper and lower indices are the same and equal to the satellite index,  $n$  and the drift error bound is calculated as the minimum of the previously calculated drift error bound and the currently observed drift error bound is determined recursively according to equation

$$r(t) = \min(r(t-\Delta t), \varepsilon/t)$$

where  $t$  is the time since the new satellite was first received and  $\Delta t$  is the time step.

where  $r$  is the estimated drift rate at a  $1-p_{md}$  confidence level and  $t$  is the time since the satellite was first received.

6. Apparatus according to claim 5 where a satellite error limit (SEL) is recursively determined according to the equation  $SEL(t) = \min(\varepsilon, SEL(t + \Delta t))$  where  $\Delta t$  is the time step and a horizontal protection limit associated with satellite  $n$  ( $HPL_n$ ) is calculated according to the equation

$$HPL_n = t_{hn} SEL + K_{md} rf \sigma_{SA}.$$

Where  $K_{md}$  is the statistical sigma number corresponding to the missed detection probability  $p_{md}$ ,  $rf$  is a reduction factor less than 1 and  $t_{hn} = \sqrt{t_{n1}^2 + t_{n2}^2}$  where  $t_{n1}$  and  $t_{n2}$  are elements of the least square solution matrix,  $T$ .



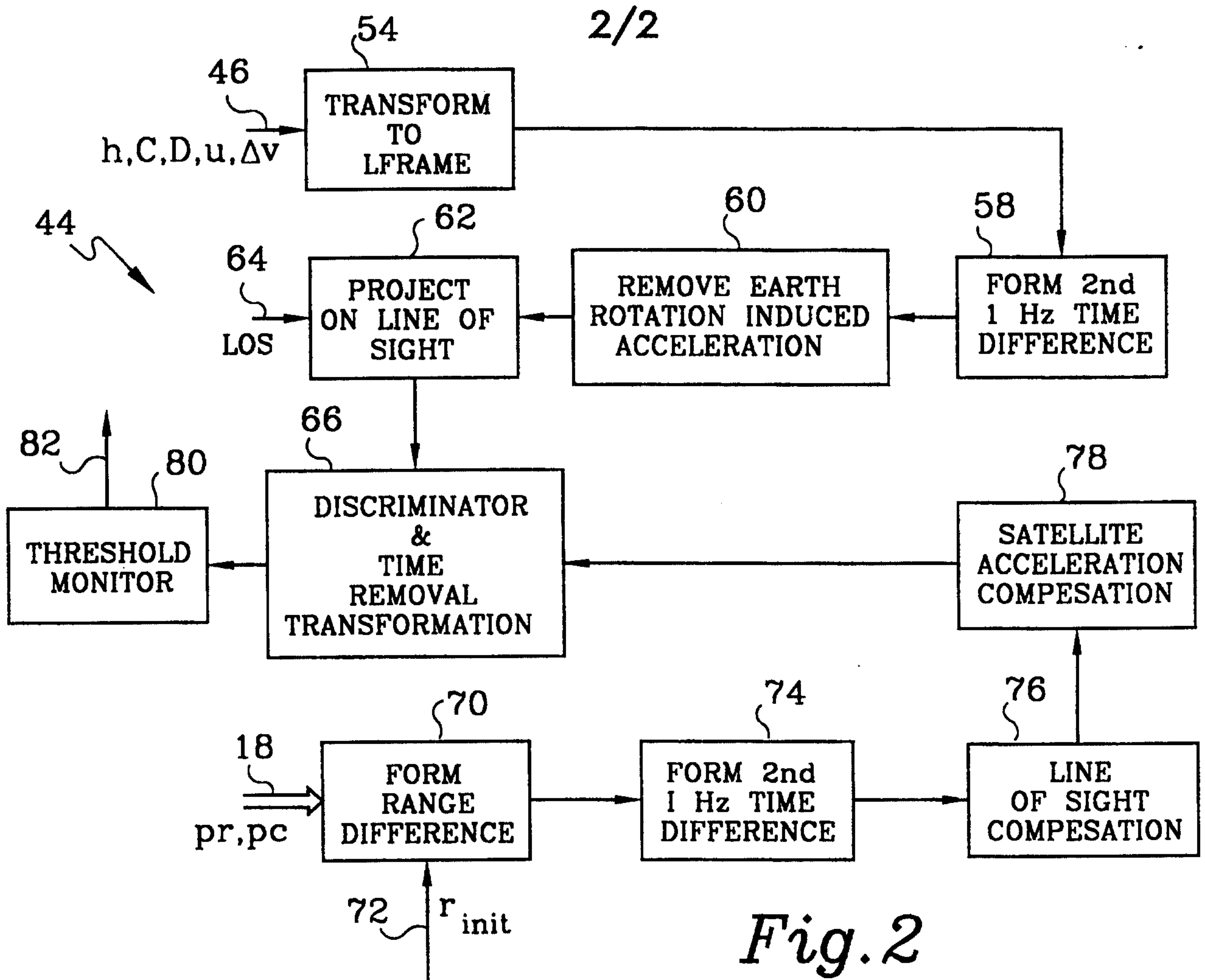


Fig. 2

