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ABSTRACT

The real-time acquisition and tracking program used on board the NASA tracking ship, the USNS VANGUARD, is described in this paper. The computer program uses a variety of filtering algorithms including an extended Kalman filter to derive real-time orbit determination (position-velocity state vectors) from shipboard tracking and navigation data. Results from Apollo missions are given to show that orbital parameters can be estimated quickly and accurately using these methods.

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THE REAL-TIME ACQUISITION AND TRACKING PROGRAM FOR THE USNS VANGUARD

INTRODUCTION

To extend the tracking, telemetry, and command coverage of the fixed land-based stations of NASA's Spaceflight Tracking and Data Network (STDN), NASA has developed a tracking ship, the USNS VANGUARD. The VANGUARD is a 600 foot ocean-going vessel which carries two major tracking systems and a variety of navigation and communication systems which make it a self-contained instrumentation site capable of supporting a variety of space missions and geophysical experiments. The ship can be located wherever necessary to provide coverage of key spaceflight events such as orbital insertion, staging, and other critical maneuvers. Originally developed as one of five such ships for Apollo support, it is currently the only such ship operated by NASA. In addition to Apollo, the VANGUARD has provided tracking support for the Pioneer and Mariner probes, the Earth Resources Technology Satellite, the Radio Attenuation Measurement Series, and other important missions in the past. Currently, the ship is being used for SKYLAB telemetry and command. Future requirements now being planned include such missions are GEOS, Atmospheric Explorer, the Global Atmospheric Research Project, VIKING, and the Apollo-Soyuz Test Project.

This paper will discuss the real-time orbit determination computer program used in the Central Data Processor System (See Figure 1) onboard the ship. The computer used is a UNIVAC 1230 with 32K memory and a 2 micro-second cycle time. Originally designed to provide sufficient data for a GO/NO-GO decision for Apollo launches, this program is a real-time, multi-task level program which accepts data from the operator console, both tracking systems (C-Band Radar and Unified S-Band), the navigation systems, and communication systems. The program stabilizes and designates both tracking systems using a variety of acquisition data sources and data from the navigation systems. Once the antenna systems acquire the space vehicle and go into auto-track, the program uses the data from the tracking systems (range, elevation, and bearing), the navigation systems (latitude, longitude, roll, pitch, heading, velocity), and ship's flexure systems as input to a powered-flight filter (polynomial filter) or a free-flight filter (extended Kalman filter) to perform independent orbit determinations (C-Band and USB data are processed independently). The output from these filters (position and velocity vectors in an earth-centered inertial coordinate system) is used for onboard displays and is transmitted back to the mission control centers at Goddard Space Flight Center and Johnson Spacecraft Center.

PRELIMINARY DATA FILTERS

Data filtering is performed in the program at several levels. Raw radar, flexure, and navigation data are sampled at a rate of ten samples per second and smoothed over one-second intervals. This processing is performed using a combination of editing and smoothing digital filters that have been designed to meet the timing requirements and to be responsive to the power spectra of the various data sources. The smoothed one-second samples are processed by various coordinate transformation routines which produce range and angles referenced to a local tangent coordinate system centered at a fixed nominal ship's position determined for each mission. This smoothed data is processed by either the powered-flight polynomial filter which produces a new position-velocity vector once per second or the free-flight extended Kalman filter which produces a new position-velocity vector once per second or the free-flight extended Kalman filter which produces a new position-velocity vector once every two seconds.

In the past, the first raw data filter has been of the classical "α - β" filter form [1]. This filter is given by the equations:

$$X_{n+1/n} = X_{n/n} + \dot{X}_{n/n}(\Delta t)$$

$$\dot{X}_{n+1/n} = \dot{X}_{n/n}$$

$$X_{n+1/n+1} = X_{n+1/n} + \alpha(X_{n+1} - X_{n+1/n})$$

$$\dot{X}_{n+1/n+1} = \dot{X}_{n+1/n} + \frac{\beta}{\Delta t}(X_{n+1} - X_{n+1/n})$$

where X_{n+1} is the raw sample at time $n+1$, $X_{n+1/n}$ is the predicted scalar parameter value at time $n+1$ based on measurements up to time n, $\dot{X}_{n+1/n}$ is the predicted parameter rate $X_{n+1/n+1}$ is the filtered parameter estimate at time $n+1$ based on measurements up to time $n+1$, $\dot{X}_{n+1/n+1}$ is the filtered parameter rate estimate, and α and β are constants chosen to optimize filter performance with respect to the data being processed. This filter is formally equivalent to a Kalman filter with a linear dynamical model and a constant gain given by the parameters α and β.

The raw data is input through interface buffers [Figure 1]. The program decodes the binary input and the data validity flags. The same filter subroutine is used for processing all tracking and navigation data streams. However, each data stream is processed with different α and β and a different editing limit. The filter contains provision for real-time editing and reinitialization based on limiting the size of $|X_{n+1} - X_{n+1/n}|$. Present limits for this value are included for each data stream processed by the filter. If at any time the limit is exceeded, the raw value is set equal to the predicted value. If too many edits occur during a time interval, the filters are reset. The filters operate at a fundamental rate of ten samples per second (Δt = 100 milliseconds) although predictions of up to $3\Delta t$ are required for certain asynchronous devices.

Recently, a second order polynomial raw data filter has been used in place of the above $\alpha - \beta$ filter. This filter uses a weighting coefficient, β, to control the amount of smoothing performed by the filter. The same filter subroutine is used to process all tracking, flexure, and attitude data but each data stream uses a value of β chosen to provide appropriate filter performance. Filters of this type are described in [5, Ch. 13].

The filter has the form:

$$X_{n+1/n} = X_{n/n} + \dot{X}_{n/n}(\Delta t) + 1/2\, \ddot{X}_{n/n}(\Delta t)^2$$

$$\dot{X}_{n+1/n} = \dot{X}_{n/n} + \ddot{X}_{n/n}(\Delta t)$$

$$\ddot{X}_{n+1/n} = \ddot{X}_{n/n}$$

$$X_{n+1/n+1} = X_{n+1/n} + (1 - \beta^3)\,(X_{n+1} - X_{n+1/n})$$

$$\dot{X}_{n+1/n+1} = \dot{X}_{n+1/n} + \frac{3}{2(\Delta t)}\,(1 + \beta)\,(1 - \beta)^2\,(X_{n+1} - X_{n+1/n})$$

$$\ddot{X}_{n+1/n+1} = \ddot{X}_{n+1/n} + \frac{(1 - \beta)^3}{(\Delta t)^2}\,(X_{n+1} - X_{n+1/n})$$

where $0 \leq \beta \leq 1$. It should be noted that this filter is also structurally a Kalman filter with a quadratic dynamical model and a constant gain and that small values of β give a large gain hence a highly responsive filter while large values of β provide a heavy smoothing of the data. Real-time editing and reinitialization features as described above are also included in this filter.

The next data filter is a second order least-squares mid-point smoother which operates on the 100 millisecond data to produce smoothed navigation and radar data at a rate of one sample per second. These filters have the form:

$$X_{mp} = \sum_{1}^{11} a_i \bar{X}_i$$

where the $\overline{X}_i$ are the outputs of the first raw data filters, the a_i are weighting constants derived from the normal equations of the second-order least-squares formulation, and X_{mp} is the smoothed mid-point value. The filters are synchronized so that the smoothed values are calculated each second on the second. A validity bit is defined based upon the validity bits of the processed raw data. The C-Band metric data validity bits are combined (logical and) to determine a measurement set (range and angles) validity bit. Identical processing occurs for the USB data.

COORDINATE SYSTEMS

Before further processing can be done the metric data must be transformed to different coordinate systems. Four different coordinate systems are used in the program. The first coordinate system is called the pedestal coordinate system. A position vector in this coordinate system is defined in terms of range, elevation, and bearing relative to the antenna pedestal. The second coordinate system is called the deck coordinate system. It is a similar coordinate system to the pedestal system, but it is centered at the ship's inertial navigation system rather than at the antenna pedestal. Next we have the local stable or local tangent coordinate system in which positions are referenced to a system tangent to the geoid of the earth. To transform to this system the attitude of the ship (roll, pitch, and heading) must be known, and it is at this point that the smoothed navigation data is used. Finally, we have the Apollo or inertial coordinate system. This is an earth-centered non-rotating coordinate system with the X-axis through the Greenwich Meridian at the equator at midnight of launch day (GMT = 0) the Y-axis 90° East in the equatorial plane, and the Z-axis through the north pole. The validity bit is carried through the coordinate transformations to provide validity bits for the inertial position vectors.

The next data filter is the powered flight polynomial smoother. It is selected whenever the spacecraft is thrusting or when other non-orbital tracking is being conducted (e.g., aircraft or weather balloons). This filter operates on the smoothed one-second navigation and metric data to produce position and velocity vector estimates in the inertial coordinate system at a one second rate. The smoothed one-second metric data is transformed from the pedestal system to the deck system and, using the smoothed navigation data, to the local stable and finally to the inertial system. The output of these transformations is an array of one-second position vectors in the inertial system. A second order polynomial is fit to the data over a ten-second arc (eleven points). The mid-point and the derivative at the midpoint are used to obtain the position-velocity vector. If at least eight of the eleven points are valid, the filter output is marked valid. A sliding midpoint fit is used so that a new position-velocity vector is obtained each second on the second.

The output of the powered flight filter is also used to obtain the initial state vector and state covariance matrix for the free-flight filter which will now be described.

EXTENDED KALMAN FILTER

The free-flight filter is an extended Kalman filter [3] which uses the spacecraft position-velocity vector (earth-centered inertial coordinate system) as the state vector and the smoothed one second range and angles (output from the above mentioned digital filters) as measurement data. The dynamical model consists of a simplified earth's gravitational field using the central term and the oblateness term. Therefore, the state vector

$$X = \begin{pmatrix} X_p \\ X_v \end{pmatrix}$$

satisfies the differential equation:

$$\dot{X} = \begin{pmatrix} X_v \\ G(X_p) \end{pmatrix} \text{with } G(X_p) = \begin{pmatrix} G_1 \\ G_2 \\ G_3 \end{pmatrix}$$

where

$$G_1(X_p) = -\frac{\mu X_{p1}}{R^3}\left(1 + J\frac{a^2}{R^2}\left(1 - 5\frac{X_{p3}^2}{R^2}\right)\right),$$

$$G_2(X_p) = -\frac{\mu X_{p2}}{R^3}\left(1 + J\frac{a^2}{R^2}\left(1 - 5\frac{X_{p3}^2}{R^2}\right)\right),$$

$$G_3(X_p) = -\frac{\mu X_{p3}}{R^3}\left(1 + J\frac{a^2}{R^2}\left(3 - 5\frac{X_{p3}^2}{R^2}\right)\right),$$

$R = \| X_p \|$, J is the oblateness coefficient, a is the earth's semi-major axis, and μ is the gravitational constant.

This is made into a discrete time model with a fourth-order Runge-Kutta integration algorithm. We shall denote numerical integration by $F(\cdot)$. Thus, if we denote by $X_{j/j}$ the estimate of the state vector at time t_j based on data taken up to t_j, then the predicted state vector at t_{j+1} is given by $X_{j+1/j} = F(X_{j/j})$.

As mentioned previously, the initial state vector and state covariance matrix are obtained from the output of the powered flight polynomial filter. The selection of which filter is to be used is currently a manual function controlled by a button on the operator's control panel. When the operator is advised of thrust cutoff, he releases the thrust button and free-flight filter initialization begins. The powered flight filter processes the next ten second data arc as described previously to obtain a position-velocity vector estimate. The mean square error of the fit is used to obtain the magnitude of the estimated variance of the velocity fit. The initial state covariance matrix is diagonal with preset position variances and velocity covariances obtained as above (but not less than (50 ft./sec.)^2).

The use of actual measurement data to perform filter initialization has the effect of assuring that the initial state lies in the linear range of the filter. Although processing data by the Kalman filter is delayed somewhat by this initialization procedure, we have found that the more accurate initialization obtained from processing real data as compared with a premission nominal initialization more than compensates for the delay. As will be shown later, the filter converged to accurate solutions for the Apollo trajectories within data spans of one to three minutes.

The prediction of the covariance matrix requires a state transition matrix. The state transition matrix used in this program is a analytic matrix based on a truncated power series expansion using a method due to Danby [2]. The matrix $\Phi(t, t_0)$ is partitioned into four sub-blocks:

$$\Phi = \begin{pmatrix} \Phi_{11} & \Phi_{12} \\ \Phi_{21} & \Phi_{22} \end{pmatrix}$$

The sub-blocks are given by:

$$\Phi_{11} = I + 1/2\, G(t - t_0)^2$$

$$\Phi_{12} = I(t - t_0) + 1/6\, G(t - t_0)^3$$

$$\Phi_{21} = G(t - t_0)$$

$$\Phi_{22} = I + 1/2\, G(t - t_0)^2$$

where I is the identity matrix, G is given by:

$$G = -\frac{\mu}{|R|^3} I + \frac{3\mu}{|R|^5} X_p X_p^T,$$

μ is the gravitational constant, X_p is the three dimensional position vector and $R = ||X_p||$ evaluated at t_0.

The covariance matrix $P_{j/j}$ is predicted to $P_{j+1/j}$ by the equation:

$$P_{j+1/j} = \alpha \Phi(t_{j+1}, t_j) P_{j/j} \Phi^T(t_{j-1}, t_j).$$

Note that there is no state noise model which would add another term to this equation, and that there is an extra age-weighting factor, α, which can enlarge the predicted covariance. We have used this factor as a divergence prevention device with values for α between 1.0 and 1.4. Most divergence prevention devices focus on keeping the state covariance matrix realistically large. The extended Kalman filter, because of neglected nonlinearities and modeling errors, often tends to produce optimistically small state covariances and hence will tend to become unresponsive unless some device is used to enlarge the state covariance or the gain matrix. This has been a subject of considerable research [3].

The measurement vector consists of range, elevation, and bearing referenced to the ship's position. We shall denote the measurement vector at time j by Y_j. The measurement model function which maps the state vector to the measurement space will be denoted by $h(\cdot)$. Thus, the predicted measurement at t_{j+1} given measurements up to t_j shall be written as $\bar{Y}_{j+1} = h(X_{j+1/j})$. If the measurement vector is invalid, the filter update is not performed.

We assume that the measurement data is corrupted by Gaussian white noise with mean zero and covariance matrix γR which is given by $R = \text{diag}(\sigma_R^2, \sigma_E^2, \sigma_B^2)$ where σ_R = 30 ft., $\sigma_E = \sigma_B$ = .25 deg, and γ is an elevation dependent factor defined by $\gamma = \max(1, 85/(218.5 \times EL-2))$ where EL = max (elevation angle in radiances, .04). For low elevations γ is about 12, and it decreases to 1 at 22°. This factor accounts for the increased measurement uncertainty due to atmospheric refraction. The stochastic process:

The stochastic process:

$$\epsilon_{j+1} = Y_{j+1} - h(X_{j+1/j})$$

is called the innovations process, and, if the usual linearization assumptions are valid, this process is approximately Gaussian with mean zero and covariance given by:

$$E(\epsilon_{j+1}\epsilon^T_{j+1}) = H(X_{j+1/j})\, P_{j+1/j}\, H^T(X_{j+1/j}) + \gamma R$$

The matrix H is the jacobian matrix of the coordinate transformation from state to measurement coordinates. If any measurement value is such that any component of ϵ_{j+1} exceeds (in absolute value) three times the square root of the corresponding diagonal element of the above covariance matrix, then the measurement is rejected. If a preset limit on the number of such rejections is exceeded, the filter will reinitialize. This device has proved useful in preventing bad data, erroneously marked valid by the measurement systems, from being processed and causing the filter to diverge.

As a further aid to preventing filter divergence we do not allow the filtered standard deviations of the state velocity components to go below three feet per second. This limit is often reached during orbital support and is required to assure that the filter remains responsive to the data.

To summarize, this filter structure is given by the following equations:

- Initialization
  1. $X_{0/0}$ is obtained from the output of the powered flight filter.
  2. $P_{0/0}$ is diagonal with present position covariances and velocity covariances obtained from the powered flight filter.

- Prediction
  1. State prediction is accomplished using a Runge-Kutta integrator:

$$X_{j+1/j} = F(X_{j/j})$$

  2. Covariance prediction is performed using the state transition matrix and the age-weighting factor α.

$$P_{j+1/j} = \alpha\Phi(t_{j+1}, t_j)\, P_{j/j}\, \Phi^T(t_{j+1}, t_j)$$

  3. Measurement prediction is performed using the coordinate transformations which take on earth-centered vector to the measurement coordinates:

$$\bar{Y}_{j+1} = h(X_{j+1/j})$$

- Filtering
  1. Perform edit test by comparing the components of $\epsilon_{j+1} = Y_{j+1} - \bar{Y}_{j+1}$ with the appropriate elements of the covariance matrix $H_{j+1}\, P_{j+1}\, H_{j+1} + \gamma_{j+1} R$. If the number of edits exceeds a preset limit the filter is reinitialized.

2. The Kalman gain is computed from the equation:

$$K_{j+1} = P_{j+1/j} H_{j+1}^T (H_{j+1} P_{j+1/j} H_{j+1}^T + \gamma_{j+1} R)^{-1}$$

3. The filtered state is computed from the equation:

$$X_{j+1/j+1} = X_{j+1/j} + K_{j+1} \epsilon_{j+1}$$

4. The filtered covariance is given by:

$$P_{j+1/j+1} = (I - K_{j+1} H_{j+1}) P_{j+1/j}$$

If the diagonal elements corresponding to the velocity components of the state vector are less than $(3 \text{ft./sec.})^2$, the elements are replaced by $(3 \text{ ft./sec.})^2$. This addition retains both the symmetry and positive definiteness of the covariance matrix.

This filter produces a new filtered state vector every two seconds. The data is time-tagged with the total seconds since the beginning of the year, and the output is synchronized to occur every other second on the even second.

APOLLO MISSION RESULTS

Some results of orbit determination performed by the free-flight filter during the last three Apollo missions are presented in Tables 1, 2, and 3. The results of the VANGUARD orbit determinations are compared to the results of the orbit determination made at Houston from the selected source for orbit initialization and to the post-flight trajectories developed at Marshall Space Flight Center and the Goddard Space Flight Center. The comparison is done in terms of spacecraft velocity (V), flight path angle (γ), and altitude (h) since these were the parameters of interest for the GO/NO-GO decision at Houston. In evaluating the results the following factors should be recalled. Unlike a land-site, the location of the ship is rarely known to more accuracy than a few hundred meters. The ship attitude data is also subject to error. Finally, the ship must produce an orbital estimate based on only a few minutes of tracking data.

Despite these difficulties, the errors in the above orbital parameters were usually (after transients of one minute) within one meter/sec. in velocity, a few hundredths of a degree in flight path angle, and a kilometer in altitude. These figures are well within the accuracy requirements set by Houston. These requirements were $|\Delta V| \leq 4.88$ meters/sec., $|\Delta\gamma| < .16$ degrees, and $|\Delta h| < 4.45$ kilometers to be reached within one minute of insertion. Versions of this program have operated in mission support since 1968 [4], and the success of this program demonstrates the feasibility and practicality of real-time filtering for orbit determination when quick solutions are required.

For Apollo support, the VANGUARD was positioned in the Atlantic so as to provide coverage of thrust cutoff and the subsequent insertion into earth orbit. The VANGUARD was usually able to track the second earth orbit (Rev. 2) before translunar injection. The selected source and VANGUARD solutions were real-time solutions based only on the data indicated while the post-mission trajectories were based on data from many sources processed by a weighted least-squares smoother. On each figure the lift-off time and parking orbit insertion time are given. The source selection was made at Houston between one and two minutes after insertion.

During Apollo 16, the C-Band radar provided an excellent solution at source selection time, but it developed servo problem in the auto-track loop shortly thereafter. During Apollo 17, a problem in the USB antenna feed occurred which prevented the acquisition of much valid data. Due to the delay in the launch and the consequent change in launch azimuth, there was little data for Rev. 2.

CONCLUSIONS

Based on the experience gained from the Apollo missions and others, the following conclusions can be drawn.

- Extended Kalman filters can be made to work in real-data situations. This conclusion may seem obvious to some, but it is the author's experience that many people in the aerospace field do not believe this. This skepticism results from too much money spent for programs that work well in simulations but not at all on real data.
- The extended Kalman filtering process is statistically robust. This term means that the filter performs adquately on non-Gaussian data and under the handicap of modeling errors.
- For this application at least, the filter structure need not be overly complex. There is no attempt at a simultaneous estimation of parameters other than the state vector. Divergence is prevented by the simple methods discussed above.
- In high data rate situations measurement data, preprocessing is critical. Due to computer restraints, it may not be possible to operate the Kalman filter at the raw data rate. Hence, preliminary digital filters may be required. The improvement of the high rate digital filters using modern spectral estimation techniques is a current area of program development.

- For data processing at remote tracking sites or perhaps even on-board space vehicles where small computers are used, a recursive data processing formulation is required. Such a method can save considerable core storage and is also ideal for real-time application. Moreover, studies being conducted at GSFC indicate that for many orbit determination problems the filtered solution is as good as the solution produced by a weighted least-squares batch processor.

- The development of an extended Kalman filter to process real data involves a certain amount of what is called "fine tuning." This involves the proper selection of the parameters defining the probability density of the initial state vector, state and measurement noise covariances, and any other parameters required by modifications to the extended Kalman filter algorithm. For this application, the principal fine tuning involved was the selection of the [illegible] parameter in the raw data filter for each data stream using spectral analysis techniques. The R-matrix was taken from the manufacturer's specification and modified by a scalar factor to account for additions unmodeled atmospheric refraction effects. The Q-matrix was set equal to zero, and the parameter α, after much experimentation which consisted of comparing results with post-flight trajectories, was set nearly equal to one since larger values contributed to instability. As mentioned above, the initialization using real data rather than using an artifically large initial covariance matrix resulted in quicker filter convergence. The lower limit on the velocity component variances reflects a slightly conservative estimate of the filter's capability based on actual performance.

Therefore, for this application, some effort was required for tuning the filter, but not so much as to weaken the conclusions above.

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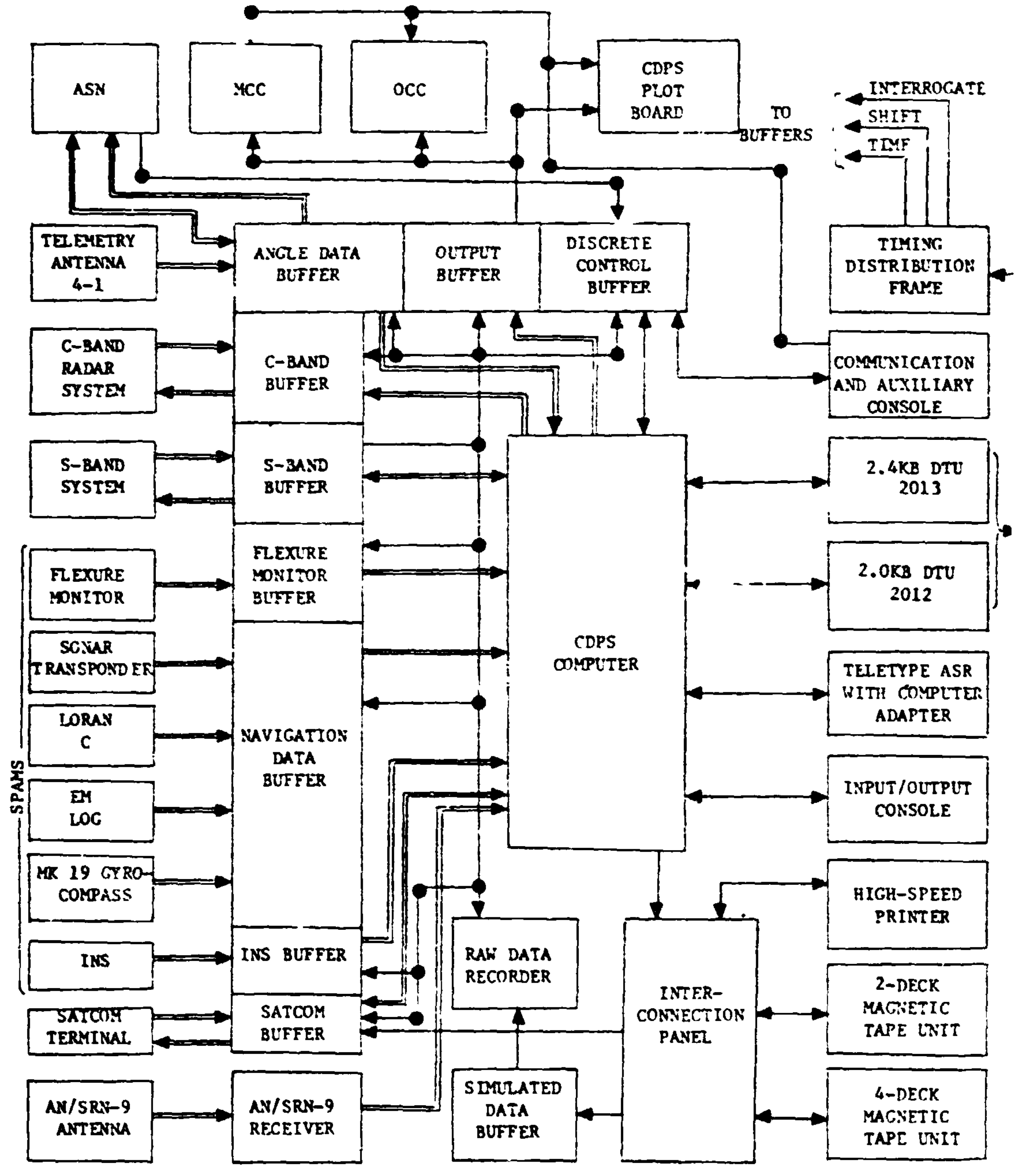

Figure 1. USNS Vanguard System

Table 1
Apollo 15 Results (July 26, 1971)

	Lift-Off Time	13/34/00 GMT		
	Parking Orbit Insertion	13/45/44.67 GMT		
	Source Selection at Houston	13/47/00 GMT		
Time	**Source**	**V(Meters/Sec.)**	**γ(Degrees)**	**H(Km.)**
Orbital Parameters (Launch)				
13/47/00	IU*	7804.09	-.000	171.5
	MSFC (Post-Flight)	7804.06	.01	172.6
	GSFC (Post-Flight)	7804.00	.01	172.8
	VAN C-Band	7802.24	-.005	172.1
	VAN USB	7802.11	.042	171.5
13/49/00	MSFC	7804.31	.01	172.4
	GSFC (Post-Flight)	7804.89	.015	172.6
	VAN C-Band	7803.89	-.018	171.3
	VAN USB	7803.48	-.001	170.9
Orbital Parameters (Rev. 2)				
15/18/28	MSFC	7801.38	-.00	177.8
(AOS +	GSFC	7801.69	.01	178.4
2 Min.)	VAN C-Band	7801.13	.08	177.4
	VAN USB	7800.71	.11	176.9
15/19/28	MSFC	7801.69	-.00	177.4
(AOS +	GSFC	7803.21	.02	178.2
3 Min.)	VAN C-Band	7802.63	.06	177.6
	VAN USB	7802.11	.09	177.4

*IU - Inertial Unit Onboard Apollo Vehicle

Table 2

Apollo 16 Results (April 16, 1972)

Lift-Off Time	17/54/00	GMT
Parking Orbit Insertion	18/05/56.21	GMT
Source Selection	18/07/30	GMT

Time	Source	V(Meters/Sec.)	γ (Degrees)	H(Km.)
Orbital Parameters (Launch)				
18/07/30	IPR*	7805.71	-.039	171.1
	MSFC	7805.04	-.00	173.0
	GSFC	7805.34	.00	173.2
	VAN C-Band	7804.73	-.004	172.2
	VAN USB	7804.61	.050	171.7
18/09/30	MSFC	7805.34	.00	172.8
	GSFC	7805.34	.00	172.4
	VAN C-Band	(Invalid Data – Servo Problem)		
	VAN USB	7804.43	.03	172.2
Orbital Parameters (Rev. 2)				
19/38/46 (AOS + 2 Min. for C-Band AOS + 16 Sec. for USB)	MSFC	7802.91	-.01	177.8
	GSFC	7802.60	.00	177.6
	VAN C-Band	7801.41	.05	177.2
	VAN USB	7808.33	.08	177.0
19/39/46	MSFC	7803.15	-.01	177.4
	GSFC	7803.03	.00	177.2
	VAN C-Band	7802.42	.05	177.2
	VAN USB	7802.94	.05	177.1

*Raw tracking data from the Atlantic Range

Table 3
Apollo 17 Results (December 7, 1972

Lift-Off Time	5/33/00 GMT			
Parking Orbit Insertion	5/44/52.65 GMT			
Source Selection	5/46/30 GMT			
Time	Source	V(Meters/Sec.)	· (Degrees)	H(Km.)
Orbital Parameters (Launch)				
5/46/30	IU	7804.12	-.001	171.5
	MSFC	7805.04	-.000	169.6
	GSFC	7805.04	.00	172.6
	VAN C-Band	7803.52	-.005	170.2
	VAN USB	(Invalid Data – Antenna Feed Problem)		
5/48/30	MSFC	7805.77	-.01	168.9
	GSFC	7804.83	.00	171.5
	VAN C-Band	7805.44	-.01	169.3
Orbital Parameters (Rev. 2)				
Due to delay in the launch, the van was not in position to see much of Rev. 2 (Maximum elevation 2.5 degrees)				

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