Scan-by-Nutation: A Spacecraft Control and Scan Motion Concept

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A spacecraft attitude control system is described which causes the instantaneous line of sight of a body-fixed payload sensor to scan through a large spatial volume while simultaneously maintaining the spacecraft to an earth centered orbit plane reference. Simulation results are shown, as are results of a demonstration test on an air bearing table. The desired reciprocating line of sight scan through the earth limb does not require expenditure of electrical energy, stored gas or fuel for scan reversal, depending instead on the undamped roll-yaw nutational motion of a deliberately disturbed momentum-bias spacecraft for scan control. The only energy required is to counteract external torques and to replenish internally dissipated energy.

Nomenclature

a,b,c,d,A,B = substitute constant functions of inertias and rates
F = thruster force level
h = momentum of momentum wheel
I_x,I_y,I_z = moments of inertia about X, Y and Z axes
$\mathcal{I}_z$ = yaw thruster impulse
ℓ = center of mass unbalance on test satellite
M_x,M_y,M_z = disturbance torques about x, y, z
M_{xc},M_{yc},M_{zc} = control thruster moment arm
R_z = yaw thruster moment arm
s = Laplace operator
t = time
t_M = thruster on-time for roll maneuver
T = scan period
w,x,y,z = substitute constant functions of inertias and rates
W = weight of test satellite and air bearing table
ζ = damping ratio of roll motion
φ_b = roll bias voltage
$\varphi_{MAX},\Delta\varphi$ = roll maneuver amplitude
φ,θ,ψ = Euler angles roll, pitch and yaw about x, y, z
$\varphi_0,\theta_0,\psi_0$ = initial condition Euler angles
$\dot\varphi_0,\dot\theta_0,\dot\psi_0$ = initial condition Euler angle rates
σ = standard deviation
ω_o = orbit rate
ω_I = nutation frequency

Introduction

A SPACECRAFT attitude control system is described which causes the instantaneous line of sight (LOS) of a body-fixed payload sensor to scan through a large spatial volume while simultaneously maintaining the spacecraft to an earth-centered orbit plane reference. The scan-by-nutation concept was developed to meet a specific set of scan requirements for an earth limb radiometer and is not applicable to a wide range of scan missions. It is presented here as an example of the use of innovative ideas to yield simple solutions for specific spacecraft missions. Exploitation of the relationship between natural motion of a spinning body (or one containing a bias momentum) and its inertial properties has been proposed in the past to stabilize or modify the motion.[1-5] The spacecraft concept described herein utilizes the spin momentum of the spacecraft momentum wheel primarily to maintain gyroscopic yaw reference. Energy added to the system is added to maintain nutation rather than to damp it as in the usual case.

The 1.5×6.0 mrad horizontally oriented instantaneous field of view (FOV) of the payload is required to be scanned vertically through the Earth's limb, on the side of the orbit facing away from the sun, from local horizontal to a point tangent to the Earth's crust (i.e., tangent altitudes from 602 km to zero). Minimum spacing between touchdowns (tangent altitude zero) must be 926 km to satisfy coverage requirements, and the maximum rate of scan of 10 mrad/sec is imposed by the payload sensor design. Rotation of the FOV about the LOS should be minimized to maintain vertical resolution, and horizontal dispersion should be minimized to ensure that each vertical profile corresponds to a given geographical area. The 20- to 30-day mission would enable gathering of data over the entire globe, from pole to pole, over conditions of varying local times. The payload is cooled by a stored cryogen which is vented to space after it sublimes and cools the detector. In terms of spacecraft axes, for a sensor viewing to the side (plus or minus Y axis), roll corresponds to tangent height, yaw to horizontal dispersion, and pitch to rotation of the FOV. The critical attitude determination requirement of 194 μrad in roll imposes the requirement for a precision attitude determination system for that axis.

Five conceptually different scan-producing control systems were considered which would produce acceptable scan motions, and the resulting scan patterns compared, as were the system complexities and impacts on other spacecraft subsystems. These techniques are compared in Table 1 on the basis of all scans constructed with 926 km between touchdowns to the earth's crust. Four of the approaches incorporate a body-mounted payload sensor. The other configuration calls for a three-axis stabilized spacecraft with a single-axis gimbal.

Approach 5 is the scan-by-nutation spacecraft described in detail in this paper. The selected approach is a pitch momentum-bias system with the payload principal ray depressed 12.1° from the spacecraft pitch axis. The spacecraft's pitch

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Index category: Spacecraft Attitude Dynamics and Control.

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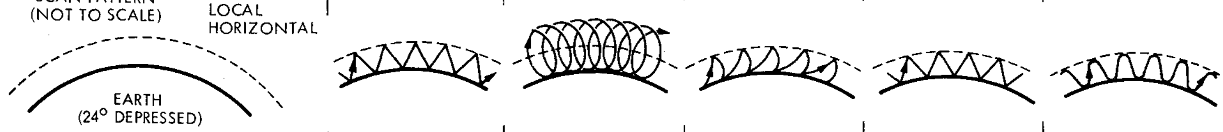
Table 1 Scan mechanization trade study

APPROACH		NO. 1 GIMBALED SENSOR	NO. 2 PITCH SPIN	NO. 3 DUAL SPIN	NO. 4 SAWTOOTH SCAN	NO. 5 NUTATING SCAN
SCAN PATTERN (NOT TO SCALE) LOCAL HORIZONTAL EARTH (24° DEPRESSED)						
SCAN PARAMETERS FOR 500 N MI BETWEEN TOUCHDOWNS	DATA DUTY CYCLE 0-150 NA MI PCT	28	14	45	28	36
	FOV ORIENTATION DEG	±1	±24 (0-150 N MI)	±24 (0-150 N MI)	±1	±2
	MAXIMUM ANGULAR SCAN RATE DEC/SEC	0.41	1.18	0.66	0.41	0.56
	TANGENT HEIGHT SCAN RATE (AT 100 NA MI) N MI/SEC	9.2	15.4	8.5	9.2	8.6
SPACECRAFT DESCRIPTION	POTENTIAL SPACECRAFT CONFIGURATION	3 AXIS STABILIZED EITHER ZERO MOMENTUM OR DUAL SPIN	SPIN-STABILIZED, SPIN VECTOR NORMAL TO ORBIT PLANE	DUAL SPIN, SPIN VECTOR NORMAL TO ORBIT PLANE	3-AXIS STABILIZED, EITHER ZERO MOMENTUM OR PITCH MOMENTUM BIAS WITH INTERAXIS MOMENTUM TRANSFER	PITCH MOMENTUM BIAS
	PAYLOAD ORIENTATION	ON SINGLE AXIS GIMBAL	OFFSET 24 DEG. FROM SPIN AXIS	OFFSET 24 DEG. FROM SPIN AXIS	OFFSET 12 DEG. FROM PITCH AXIS	OFFSET 12 DEG. FROM PITCH AXIS
	SCAN GENERATION	GIMBAL TORQUER	NONE (VEHICLE MOTION)	WHEEL TORQUE ABOUT PITCH AXIS	REACTION WHEEL OR CMG TORQUE ABOUT ROLL AXIS	NONE (VEHICLE MOTION)
SUBSYSTEM IMPACT	ATTITUDE CONTROL	COMPLEX 3 AXIS CONTROL REQUIRES TORQUING OF PAYLOAD AND JOINT FOR COOLANT	SIMPLE – PASSIVE CONTROL, NO TORQUING, NUTATION DAMPER	SIMPLE-TO-MODERATE, NUTATION DAMPER, 1 ACTIVE AXIS WITH ACTIVE TORQUING	COMPLEX NUTATION DAMPING WITH PITCH MOMENTUM BIAS VERSION, 3 ACTIVE AXES WITH ACTIVE ROLL TORQUING, REQUIRES GYRO FOR YAW	SIMPLE TO MODERATE PASSIVE CONTROL, NO TORQUING, NUTATION DAMPING IMPLICIT
	ATTITUDE DETERMINATION	STAR TRACKER HORIZON SCANNER GIMBAL PICKOFF	STAR SCANNER AND HORIZON SCANNER, SIMPLE HARDWARE, COMPLEX SOFTWARE	STAR TRACKER HORIZON SCANNER	STAR TRACKER HORIZON SENSOR	STAR TRACKER HORIZON SENSOR
	STRUCTURAL AND THERMAL	RIGID STRUCTURE BETWEEN ACS COMPONENTS AND P/L, TRANSFER OF CRYOGENIC FLUID, VIBRATION. MOST EXPENSIVE APPROACH	REQ. DYNAMIC BALANCE	MECHANICAL DESPIN BEARING VIBRATION, DYNAMIC BALANCE, CONTAMINATION POTENTIAL OF EXPOSED JOINT	3-4 WHEELS MAY REQUIRE STRUCTURAL DAMPING OR ISOLATION	1 WHEEL MAY REQUIRE STRUCTURAL DAMPING OR ISOLATION
	ELECTRICAL	80 SQ FT ARRAY	106 SQ FT ARRAY	80 SQ FT ARRAY	80 SQ FT ARRAY	80 SQ FT ARRAY
	PAYLOAD COOLANT	COMPLEX COOLANT TRANSFER ACROSS GIMBAL	HIGH APERTURE HEAT LOAD. WHEN $-25° < \beta < +25°$ COOLANT LOAD INCREASES 38.5%	+25% OVER MINIMUM	MINIMUM COOLANT	1% OVER MINIMUM
	TT&C	FIXED ANTENNA 10 WATT TRANSMITTER	REQUIRES EITHER DESPUN ANTENNA OR HIGH POWER TRANSMITTER (40 WATT-LARGER SOLAR ARRAY)	FIXED ANTENNA 10 WATT TRANSMITTER	FIXED ANTENNA 10 WATT TRANSMITTER	FIXED ANTENNA 10 WATT TRANSMITTER
RELATIVE COSTS-TOTAL 2 SPACECRAFT		3.41	0.82	1.10	1.14	1.00

axis is set into a controlled nutation about the orbit normal. The scan normally extends ±12.1° about the nominal 12.1° angle. An elliptic motion determined by the spacecraft inertial properties is described by the nutation. This desired nutation motion (which is maintained, not damped) produces a scanning motion with acceptable scan rate characteristics and with a small rotation of the instantaneous FOV. The control system itself is relatively simple, depending, as does the pure spinner, on its inherent inertial stability. The momentum wheel is of fixed speed, and no instrument gyroscope reference is required to generate the scan pattern or establish the spacecraft yaw reference. The motion is controlled about a natural nutation oscillation of the system dynamics. Control correction is required only to compensate for effects tending to disturb, or deviate from, the basic free oscillation.

Figure 1 is a plot of LOS tangent point altitudes (naut miles) vs in-track distance (naut miles) for the nutating system. The circled figures along the horizontal line depict the spacecraft position relative to the corresponding numbers on the trace. The nutation period is 125 sec. A momentum wheel was selected (40.7 N-m-s) that approximates the design goal while employing available, qualified wheels.

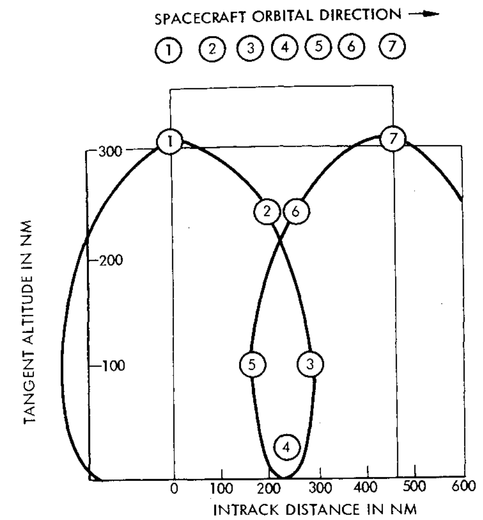


Fig. 1 Tangent altitude for nutating scan.

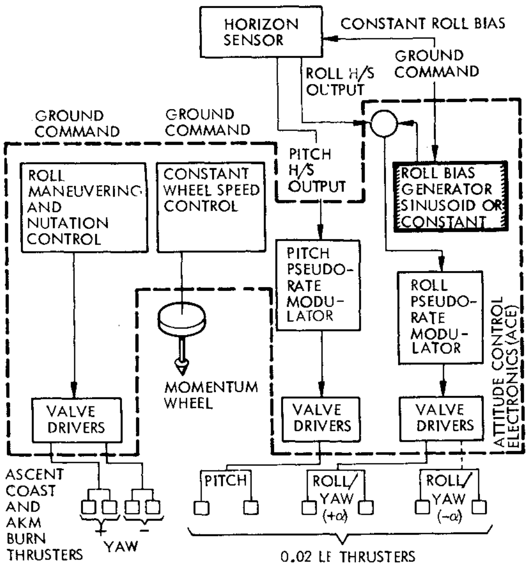


Fig. 2 Nutating scan attitude control electronics.

The selected configuration is inherently more flexible and forgiving with respect to possible changes in payload or operational requirements. As shown, it has an effective scan, which can be further optimized at a small cost. It has no exposed bearings and no new technology, and it depends on a completely passive stable gyroscopic motion to produce the scan.

Scan-by-Nutation Implementation

The attitude control system used to implement the scan-by-nutation concept (Fig. 2) consists of a fixed-speed wheel whose angular momentum is aligned with the vehicle negative pitch axis (Fig. 3), a horizon sensor to provide pitch and roll attitude information, and control electronics utilizing pseudorate modulators to drive a set of cold-gas thrusters. Except for the use of a sinusoidal roll attitude bias, this system is identical to the WHECON (wheel control) system studied by Dougherty, Rodden, and Scott.[6]

In a WHECON control system, pitch and roll control is implemented by the generation of control torques from the thrusters in response to error voltages derived from differencing horizon sensor outputs and a bias level. The bias can be commandable and represents the desired vehicle attitude.

Yaw control is achieved without the use of a yaw sensor by utilizing as a control torque the gyroscopic restoring torque

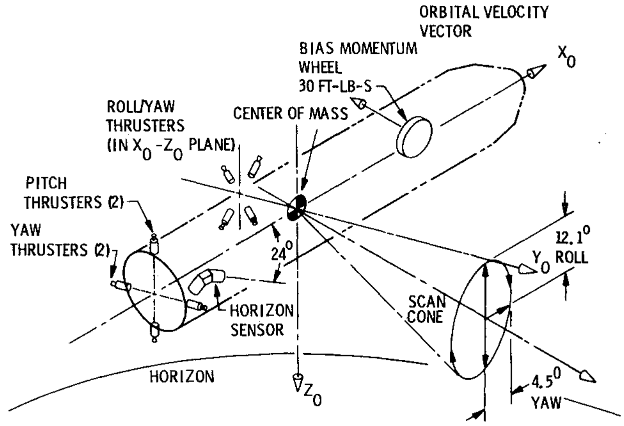


Fig. 3 Control system for nutating scan.

resulting from an offset of the wheel angular momentum. Since roll-yaw motion is coupled, yaw controller damping may be introduced by offsetting the roll thrusters to produce both roll and yaw torques. Yaw motion sensed by the roll horizon sensor activates the roll gas valve, and damps the yaw gyroscopic torque.

Following the derivation of Ref. 6, the dynamics of the WHECON system are developed. Using small angle approximations, the equations of motion expressed in the Laplace domain are

$$\begin{bmatrix} I_x s^2 + a + h\omega_0 & 0 & (b+h)s \\ 0 & I_y s^2 + d & 0 \\ -(b+h)s & 0 & I_z s^2 + c + \omega_0 h \end{bmatrix} \begin{bmatrix} \varphi(s) \\ \theta(s) \\ \psi(s) \end{bmatrix}$$

$$= \begin{bmatrix} I_x(\dot{\varphi}_0 + s\varphi_0) + (b+h)\psi_0 \\ I_y(\dot{\theta}_0 + s\theta_0) \\ I_z(\dot{\psi}_0 + s\psi_0) - (b+h)\varphi_0 \end{bmatrix} + \begin{bmatrix} M_x + M_{xc} \\ M_y + M_{yc} \\ M_z + M_{zc} \end{bmatrix} \tag{1}$$

$$a = 4\omega_0^2(I_y - I_z) \qquad c = \omega_0^2(I_y - I_x)$$

$$b = -\omega_o(I_x - I_y + I_z) \quad d = 3\omega_0^2(I_x - I_z)$$

For the case of the scan-by-nutation application, the wheels' angular momentum is made to dominate over the gravity-gradient and inertia gyroscopic coupling effects. For this case, in which $a \ll \omega_o h$, $c \ll \omega_o h$, $b \ll h$, the coupled roll-yaw equations of motion are

$$\begin{bmatrix} I_x s^2 + h\omega_0 & hs \\ -hs & I_z s^2 + \omega_0 h \end{bmatrix} \begin{bmatrix} \varphi(s) \\ \psi(s) \end{bmatrix}$$

$$= \begin{bmatrix} I_x(\dot{\varphi}_0 + s\varphi_0) + h\psi_0 \\ I_z(\dot{\psi}_0 + s\psi_0) - h\varphi_0 \end{bmatrix} + \begin{bmatrix} M_x + M_{xc} \\ M_z + M_{zc} \end{bmatrix} \tag{2}$$

For the scaling investigated, the roots of the characteristic equation of the coupled roll-yaw system are approximately ω_o (the orbit frequency) and $\omega_l \approx h/(I_x I_z)^{1/2}$ (the nutation frequency), which are separated with $\omega_0 \ll \omega_l$. This solution of Eq. (2) for roll and yaw responses to initial conditions becomes

$$\varphi(t) = \varphi_o\left(1 + \frac{\omega_o I_x}{h}\right)\cos\omega_o t - \varphi_o \frac{\omega_o I_x}{h}\cos\omega_l t$$

$$+ \psi_o \sin\omega_o t - \psi_o \frac{\omega_o}{\omega_l}\sin\omega_l t + \dot{\varphi}_o \frac{I_x}{h}\sin\omega_o t \tag{3}$$

$$+\dot{\varphi}_o[(1-\frac{\omega_o I_x}{h})/\omega_I]\sin\omega_I t-\dot{\psi}_o\frac{I_z}{h}\cos\omega_o t$$

$$+\dot{\psi}_o\frac{I_z}{h}\cos\omega_I t$$

$$\psi(t)=-\varphi_o\sin\omega_0 t+\varphi_o\frac{\omega_o}{\omega_I}\sin\omega_I t+\psi_o(1+\frac{\omega_o I_{\hat{z}}}{h})\cos\omega_o t$$

$$-\psi_o\frac{\omega_o I_{\hat{z}}}{h}\cos\omega_I t+\dot{\varphi}_o\frac{I_x}{h}\cos\omega_o t-\dot{\varphi}_o\frac{I_x}{h}\cos\omega_I t$$

Equations (3) and (4) show the important result for the scanning technique that the short term roll and yaw responses to initial rates are sinusoidal at frequency ω_I. This sinusoidal motion is the basis of the scan-by-nutation concept. By proper selection of wheel angular momentum and initial conditions, the desired amplitude and frequncy of sinusoidal scan motion about the roll axis can be induced. Once started this motion is self-sustaining in that no energy need be expended to continue the scan, except for that required to compensate for internal damping or external torques.

In an application where continuous scanning over one or more orbits is required, the initial conditions and angular momentum are adjusted to insure adequate coverage over an entire orbit. That is, the terms containing ω_o must also be considered.

For the particular requirements of the scanning earth limb radiometer, the sensor LOS is aligned in the plane formed by the vehicle $Y-$ and Z-axes and is depressed 12.1° below the-Y-axis (Fig. 3). The spacecraft is steered to −12.1° in roll to align the LOS with the local horizontal. A 24.2°, peak-to-peak roll sinusoidal motion at the nutation frequency is then initiated. Any deviation from the desired sinusoidal profile due to external torques or internal energy dissipation will be controlled by the normal response of the WHECON roll control channel. The reference roll bias is a sinusoid which matches the desired roll scan motion. Deviations from the bias result in control signals from the horizon sensor.

Both roll maneuvering and scan motions are initiated by firing a yaw thruster to generate a yaw rate initial condition. For roll maneuvering, a second yaw thruster firing occurs 1/2 nutation cycle later to null the yaw rate when roll is at the peak of its attitude motion. By utilizing the nutation motion for steering, there is no residual nutation after a maneuver. Yaw torque requirements to steer in roll or initiate nutation are a function of the desired roll motion amplitude. In Eq. (3) for $\varphi_o=\psi_o=\dot{\varphi}_o=0$ and $\dot{\psi}_o=\mathcal{J}_z/I_z$

$$\varphi(t)=-(\mathcal{J}_z/h)(\cos\omega_o t-\cos\omega_I t) \tag{5}$$

For time intervals on the order of the nutation period and with nutation frequency ω_I much greater than orbital angular velocity ω_0, it may be assumed that $\cos\omega_0 t\approx 1$. The desired value t_M of thruster on-time to initiate a peak-to-peak roll angle φ_{MAX} may be established through the approximation $\varphi_{\mathrm{MAX}}\approx -2\mathcal{J}_z/h$ as

$$t_M=h\,\varphi_{\mathrm{MAX}}/2FR_z \tag{6}$$

where F is yaw thrust level, and R_z is the thruster moment arm. Selection of the wheel angular momentum is based on the required scan period, T sec: $h=(2\pi/T)\,(I_xI_z)^{1/2}$.

To minimize control gas usage, the sinusoidal roll bias is matched as closely as possible to the undriven roll nutation motion. Generally, all of the relevant terms of Eq. (3) must be used in specifying the roll bias signal. For the common case where the yaw angle and roll rate initial conditions are zero, the roll bias $\phi_b(t)$, is

$$\varphi_b(t)=A\cos\omega_o t+B\cos\omega_I t$$

where

$$A=\varphi_o(1+\frac{\omega_o I_x}{h})-\dot{\psi}_o\frac{I_z}{h} \tag{7a}$$

$$B=-\varphi_o(\frac{\omega_o I_x}{h})+\dot{\psi}_o\frac{I_z}{h} \tag{7b}$$

Further matching of nutation motion and roll bias requires inclusion of the various terms eliminated as small in the derivation of Eqs. (3) and (4). As scan amplitudes increase, the small angle approximation will become invalid so that the scan motion must be solved from non-linear equations.

One nonlinear problem encountered in this study was the coupling of roll-yaw motion into pitch. This coupling results in a sinusoidal pitch motion and an initial condition pitch rate at the time of nutation initiation. The pitch attitude control channel (identical to roll except that the reference bias is zero) will remove the initial rate condition and then allow the sinusoidal motion inside the controller attitude deadbands. External disturbance torques will also be compensated by the pitch controller.

Since the scan-by-nutation concept allows for the roll-yaw induced pitch motion within the pseudorate deadbands, nutation damping effects of pitch control torques[5,7] will be minimized. The spacecraft design must also reflect the potential perturbations to the desired roll-yaw nutation which can result from products of inertia, fuel slosh, etc. The design problem in this case is opposite to that usually encountered in dual-spin spacecraft, i.e., the elimination of nutation damping instead of its maximization. However, the extensive literature on nutation damping may be used to aid in parameter selection and design to enable the maintenance of nutation motion.

Additional Scan Capability

Two additional scan modes are inherently provided by the scan-by-nutation control system: 1) Constant tangent altitude scan; and 2) Scanning ahead or behind the satellite. The first mode requires a constant roll attitude offset and utilizes spacecraft motion to accomplish altitude scan. Any roll offset within the roll-bias range can be maintained by proper setting of the bias voltage. The WHECON control system will null the attitude to the bias level (within the controller deadbands).

Maneuvering to the required attitude is accomplished in exactly the same manner as described to position the spacecraft prior to nutation scan initiation. The yaw pulse on-

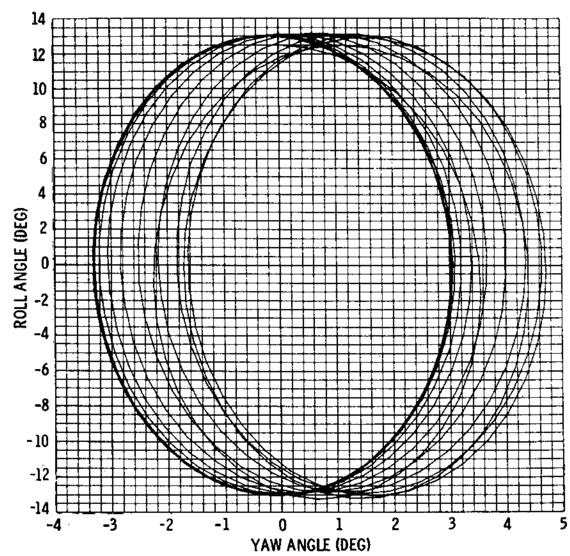


Fig. 4 Nutation control with frequency mismatch.

time is found from Eq. (6) with φ_{MAX} replaced by $\Delta\varphi$, the change in roll attitude desired.

Maintaining the roll attitude will require additional control gas expenditure to counter the gyroscopic torque produced by offsetting the wheel angular momentum in roll. For the particular vehicle studied, the gas requirement was 0.09 kg per orbit to scan a constant 100 km tangent altitude (9.8° roll offset).

In a manner completely analogous to roll maneuvering, yaw may also be offset. However, the yaw attitude cannot be maintained, so the sensor LOS will sweep horizontally back to zero position. This yaw scan can be done either alone or in conjunction with a roll-yaw nutation scan.

One requirement for the originally designed scan-by-nutation system was to be able to scan either to the "right" or "left" of the orbit plane. To provide stable control with the WHECON system, an additional set of offset roll thrusters had to be provided as shown in Fig. 3. A 180° yaw maneuver was used to move the sensor LOS from one side of the orbit plane to the other, and a wheel reversal provided stable operation in the new attitude.

Simulation Results

Both linear and nonlinear dynamic models were used to simulate the roll-yaw nutation used to produce the sensor scan. A linear model was used to investigate the operation of the system with mismatches in roll bias amplitude and frequency. Figure 4 shows the roll and yaw motion for a 10% mismatch in nutation and roll bias frequencies. After 2600 sec the spacecraft axial motion is entrained with the roll bias motion even though its natural frequency is not matched. The penalty of this situation is that control gas is being expended to force the motion to conform to the bias.

Figure 5 shows roll and yaw motion (linear dynamics) with the roll bias amplitude 30% lower than the nutation amplitude. Again, the spacecraft is forced to follow the programed bias even though the initial conditions were not proper for the case. For both mismatch cases, frequency and amplitude, the nutation was forced to follow the roll bias in less than one orbit.

For the nominal case of roll bias matched to the nutation and with a nonlinear dynamics simulation, Fig. 6 shows the roll, pitch, and yaw motion from 1500 to 2000 sec after initiating the nutation. The pitch deadband is set to ±1.0° and the nutation period is 150 sec. Pitch motion exhibits the deadband-to-deadband motion resulting from damping the initial rate. No external torques were simulated in these runs. The harmonics of the nutation frequency are clearly evident in the pitch response. Stable nutation motion results after 1500 sec. These nonlinear dynamic simulations were run for approximately half an orbit to confirm the stability of the nutation.

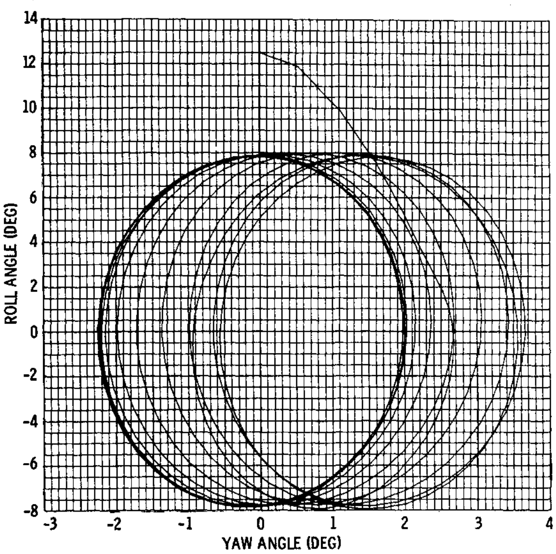


Fig. 5 Nutation control with amplitude mismatch.

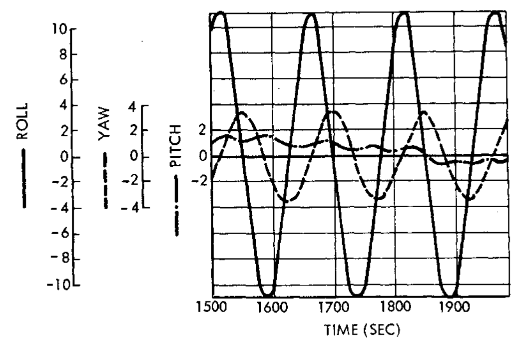


Fig. 6 Roll, pitch and yaw attitude time histories (deg).

Attitude Determination

The described attitude determination system using a star tracker and a horizon sensor achieves the desired attitude determination accuracy during the selected roll-yaw scan motion of the spacecraft.

The undisturbed attitude motion of a nutating spacecraft in roll and yaw is characterized by sinusoidal oscillation at two frequencies, i.e., the nutation frequency and the orbital frequency [Eqs. (3) and (4)]. A fixed phase relation exists between roll and yaw, with yaw lagging roll by 90°. For given inertial characteristics (moments of inertia) of a spacecraft and a specified nutation frequency (926 km vertical sampling interval) the required nutation characteristics are uniquely defined, yielding a maximum scan rate of 9.8 mrad/sec occurring at 450 km tangent altitude, with lower rates elsewhere (7.3 mrad/sec at 150 km). Once set in motion, the spacecraft scan motion is smooth, with no disturbances expected which could cause any high-frequency attitude changes. The absence of high-torque devices precludes their generation of large disturbance torques. The control thrusters are sized at 0.09 N, to further minimize the spacecraft disturbance environment, and the telemetered thruster firings are used in the attitude determination program to reinitialize the nominal motion estimate, thereby accounting for their effect on the attitude motion before fitting takes place using star data. The assumed technique for continuous attitude determination does not require controlled rates or even knowledge of spacecraft rates. It is a curve-fitting technique utilizing a continuously tracking star tracker with a sensitivity adequate to ensure almost continuous availability of stars, together with a model of spacecraft nutation motion.

Attitude Determination Sensors

The inputs to the payload data processing consist of continuous star tracker and horizon sensor measured attitudes. The star tracker giving star position readout in two axes is mounted with its LOS approximately along the spacecraft pitch axis, providing an output corresponding to the roll and yaw motions of the spacecraft. By utilizing position readout of stars up to magnitudes of +6.0, continuous attitude measurement is obtained at the telemetry sampling period. New data would be unobtainable only during transition between stars, not to exceed 5 sec each. The attitude determination algorithm, however, provides attitude information continuously. The assumed star tracker, the Ball Brothers Research Corp. CT-401, was manufactured for use on Small Astronomy Satellite-C (Explorer 53), has the capability of tracking stars from +2 to +6 magnitudes. This provides essentially continuous star tracking capability for any orbit, with only a few brief (20 sec) periods without a star for even the worst orbits. The assumed attitude determination

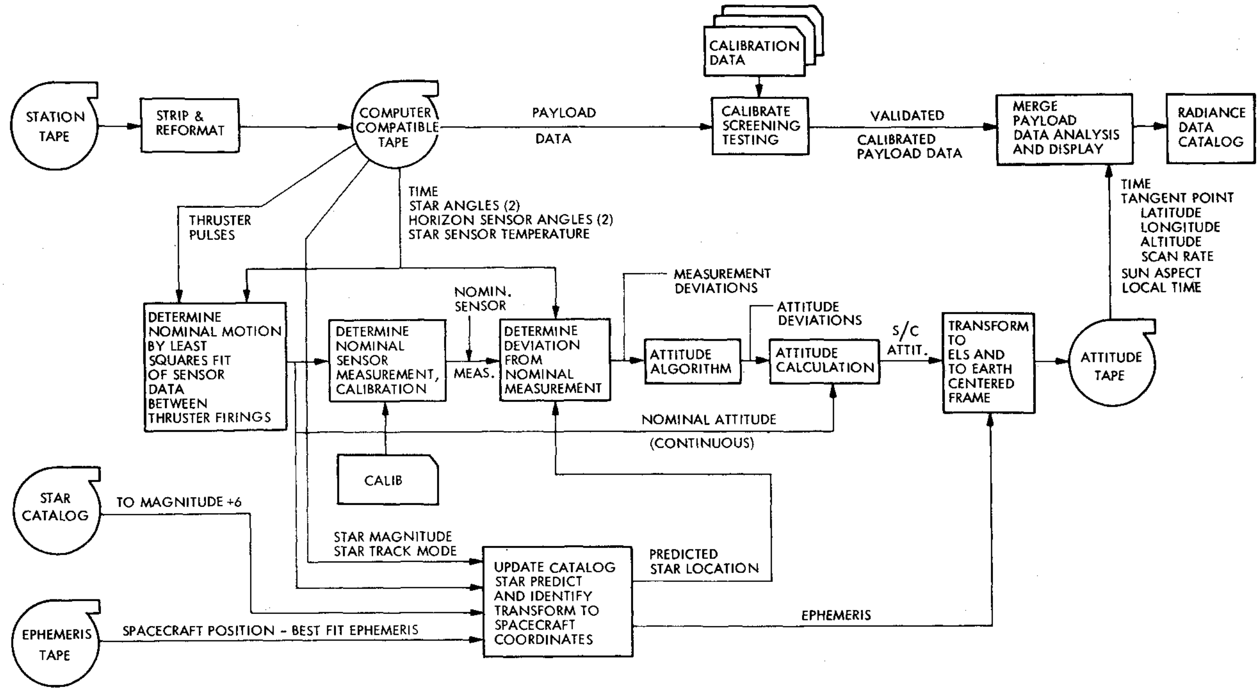


Fig. 7 Assumed form of the attitude determination process.

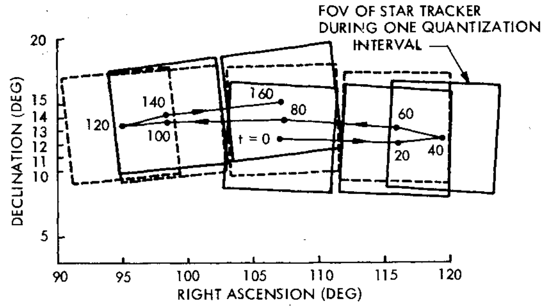


Fig. 8 Instantaneous field of view of star tracker during one scan cycle.

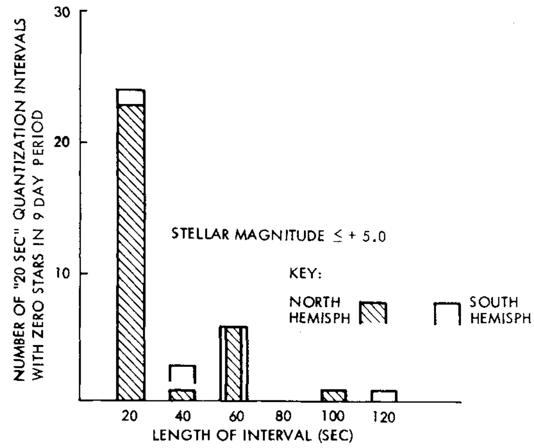
algorithm involves the matching of arc-segments of measured data with an analytical expression of the dynamic nutating motion and estimating a nominal nutation curve about which perturbations are taken. The star tracker provides accurate star position readout within ± 97 μrad (1σ) in roll and yaw, and is not degraded by scan rates up to 10.5 mrad/sec.

The horizon sensor assumed for this application is the Barnes Engineering Co. Model 13-166 with fields of view operating on the fore and aft horizons. In this configuration the roll output linear range is $\pm 15°$ to accommodate the $\pm 12.1°$ nutation. The pitch linear range of $\pm 10°$ adequately handles the $\pm 1°$ pitch motion. Operated in this mode, pitch and roll accuracies are expected to be within ± 1.2 mrad (1σ). The pitch outputs of the horizon sensor are used as a direct measurement of pitch attitude, as well as to augment the pitch measurement equation in the attitude determination software.

Attitude Determination Processing

To discuss the effects of nutation scan motion on spacecraft attitude determination, it is necessary to describe a typical attitude determination proceedure based on a linear perturbation about the nominal spacecraft motion. Figure 7 depicts the elements of the attitude determination process as assumed for the following discussion.

The form of the nominal motion is known apriori to any data gathering by the attitude sensors, Eqs. (3) and (4), and may be expressed as a set of three equations in sinusoidal terms in ω_o and ω_l with constant coefficients [e.g., Eq. (7)].



	% OF TIME W/O STARS	AVE PERIOD W/O STAR (SEC)	AVE PERIOD WITH STAR (SEC)	LONGEST PERIOD W/O STAR (SEC)
NORTHERN HEMISPHERE	16.5	31	157	100
SOUTHERN HEMISPHERE	18.5	32	140	120

Fig. 9 Star availability statistics for typical orbit.

Expressions for the equation coefficients are known in terms of vehicle parameters and external torques. Since there may be errors involved in the determination of the coefficients, they are adjusted from the measurements via a least-square fit over several nutation cycles. Once the nominal motion is determined, an additional pass through the data will be necessary to determine deviations from the nominal motion. This last pass results in highly accurate high-bandwidth attitude information in roll and yaw (the pitch information being supplied by the horizon sensor).

An advantage of using this procedure is that in the few regions of no star measurements the nominal motion will be propagated enabling a continuous attitude estimate. These regions should be almost nonexistent with the proposed sensor. This accurate nominal motion will be useful in simplifying the star identification algorithm.

In the regions where star data is available, attitude perturbations from the nominal are computed from

Table 2 Effect of tracker sensitivity on star availability

Star magnitude	Percent of time without star in FOV	
	Low star density region	High star density region
+3	72	55
+4	51	34
+5	23	6
+6	2	0

measurement perturbations defined as the difference between the actual and nominal measurements. It is at this point in the data processing that calibration corrections can be made. Hence, the definition of the nominal sensor measurement can be expanded to be that measurement that would result if the spacecraft were following the motion, and the measurements are corrected by calibrations.

The two primary calibrations to be applied to the raw star sensor data relate to the nonlinear scale factor of the star sensor and the phase shift in the measurement induced by the time constant of the star sensor interacting with the nominal motion of the spacecraft. Calibrations for temperature and magnetic field can be applied in an analogous manner.

The procedure previously described is a deterministic measurement calculation and should be at least as good as the fundamental accuracies of the sensors. Further improvement may be gained by using a dynamic filter.

Star Availability Analysis

Computer simulations were performed to determine the number of stars within the 8° square FOV of the star tracker during its scan. A star count was taken every 20 sec (quantization interval for this illustrative analysis only—the flight system tracks stars continuously) as depicted in Fig. 8.

Table 2 summarizes the results of the star availability studies in terms of the percent of time during any given orbit with no trackable stars in the tracker field of view for star tracker sensitivities of from +3 to +6, and for sparse and high density regions of the celestial sphere. Another statistic of interest in error analyses was the distribution of the times with and without stars in the instantaneous field of view. A typical orbit was chosen to illustrate the operation with a tracker threshold sensitivity of +5. The average period when no stars are within the FOV of the sensor (Fig. 9) is shorter than a nutation period. Attitude extrapolation could be perfomed over these periods since no high frequency torque sources exist which could change spacecraft attitude in an unpredictable manner.

Test Configurations

The final step in the demonstration of the scan-by-nutation concept for the orbiting payload called for a test demonstration reproducing the orbital dynamics to the extent possible in a 1 *g* environment. A Development Test Satellite (DTS) that was being tested on a 3-axis air bearing was made available for this purpose.

The test satellite contained a gimballed momentum wheel of 81.3 N-m-s that could be operated as a rigidly mounted momentum bias wheel by a mechanical clutching arrangement. Design studies yielded a required momentum value of 40.7 N-m-s. The DTS had a 81.3 N-m-s wheel and, since orbital rates, inertias and momenta had to be scaled for the nutation demonstration tests, no attempt was made to reduce the momentum to the design value. The larger momentum provided adequate stiffness in the presence of external disturbances. The existing DTS control electronics could not readily be modified to provide the required closed loop control of the nutating scan; therefore, the test was configured to demonstrate the basic concept without any control in roll or yaw (open loop). The satellite was controlled closed-loop in pitch to track an infrared target that simulated the inertially

Table 3 Measured noise characteristics of the Marconi horizon sensor

	Roll (μrad)	Pitch (μrad)
Maximum noise	908	559
Mean	87	35
Variance	7	2

rotating earth vertical vector. Spacecraft attitude motion with respect to the earth target center was measured in two axes by a horizon sensor. Provision for measuring yaw attitude motion were not available. Figure 10 depicts the test configuration. the angular velocity ω_o represents the orbital rate. The magnitude of ω_o was set at 4.66×10^{-3} rad/sec. This represented about 60 times the orbital rate at the satellite altitude considered. The advantages of this high rate are that various extraneous torques due to table imbalance are minimized and the test speeded up.

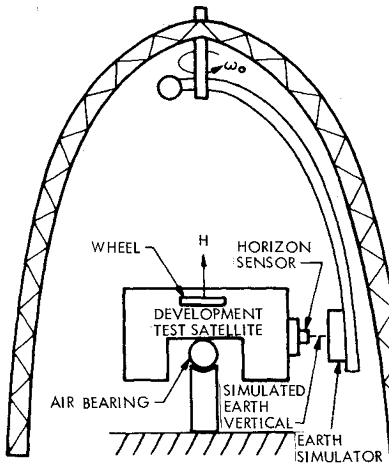


Fig. 10 Test facility schematic.

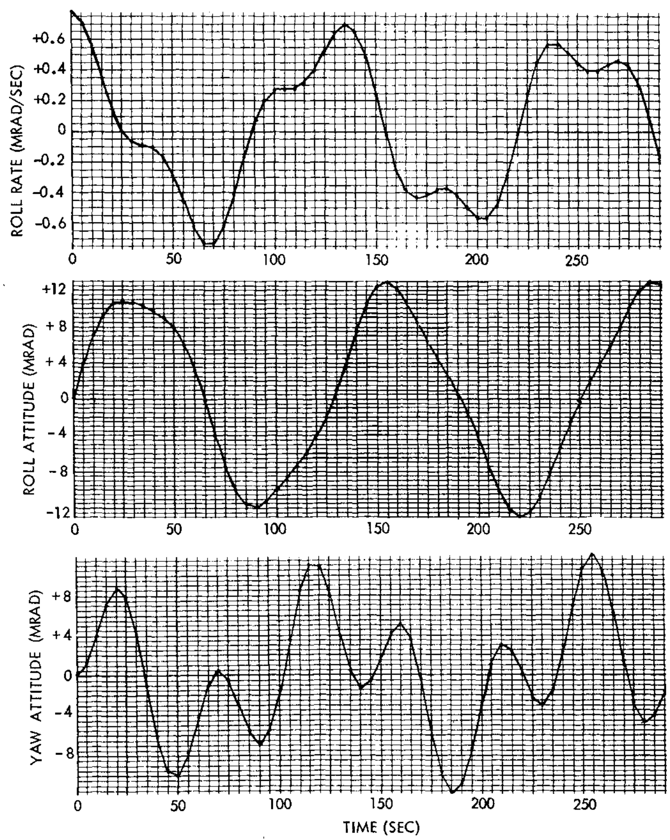


Fig. 11 Simulated roll and yaw motion of the DTS.

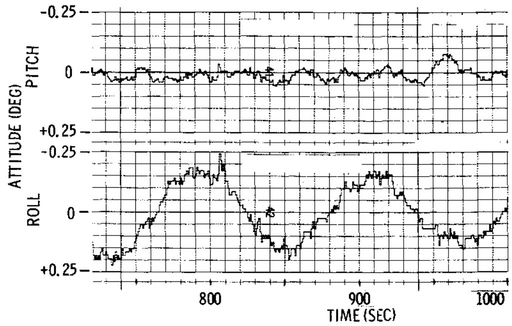


Fig. 12 Actual roll and pitch motion of the DTS.

Test Hardware

The air bearing table fixture consists of a 25.4-cm diam stainless steel sphere suspended in a 120° air bearing pad. The sphere has a 30.5 cm by 30.5 cm attachment surface for mounting of the test vehicle. The electrical and pneumatic hardline connections are routed through the center of the pedestal.

The variable speed momentum wheel of 81.3 N-m-s nominal angular momentum was built by Teldix GmbH. The wheel was mounted in a two-degree-of-freedom flexure pivot (roll and yaw) and positioned by linear stepper actuators built by the Kearfott Division of Singer. As stated previously, the actuators were locked for this test.

An earth horizon sensor built by Marconi Space and Defense Systems, Ltd., was used for attitude sensing with respect to the simulated earth target. The sensor uses a flexure mounted mirror to scan the FOV of a thermistor bolometer across the earth. Measured accuracy of this unit in the test set-up is shown in Table 3.

The earth simulator was built by Lockheed. The face is a 73.7-cm diam aluminum dish, behind which are strips of conductive heater tape. The face of the simulator is painted with 100 Series 3M Black Velvet with an emissivity of approximately 0.9. The attitude error measured by the horizon sensor was telemetered to the control room adjacent to the test area for display. The linear range for the pitch and roll channels was set at $\pm 0.25°$.

Air Bearing Balance

A spacecraft executes attitude motion about its center-of-mass. Faithful simulation of this attitude motion requires that the pivot point in the air bearing coincides with the center-of-mass of the spacecraft. In practice it is desirable to keep the center-of-mass slightly below the pivot to provide a stable restoring torque in case the test fixture is tilted about either the roll or yaw axis. It had been found during previous tests with the DTS that small temperature changes inside the test area or even changes in mass distribution of the batteries with level of charge would produce a vertical shift in the center-of-mass approaching 0.5 mm. This placed a lower limit on the practical level of balance which would assure at least a neutrally stable test configuration.

The initial balance that was achieved in this set was computed as 0.69 mm, based on a measured pendulosity period in roll (zero wheel speed) of 114 sec. The moments of inertia of the 1361-kg satellite were 3041 and 667 kg-m^2 for roll and yaw, respectively.

The theoretical nutation period as computed from $1/\omega_l = (I_x I_z)^{1/2}/h$ with a 81.3 N-m-s wheel angular momentum is 110 sec. This presented a test situation which ruled out direct observation of nutation. With the close proximity between the pendulosity and nutation modes, it is evident that nutation would be only observed indirectly through the coupling of the two modes as controlled by the linear torque-free equations

$$\text{Roll:} \quad \ddot{\varphi} + \left[\frac{\omega_0 h}{I_x} + \frac{W\ell}{I_x} \right] \varphi + \frac{h}{I_x} \dot{\psi} = 0 \tag{8}$$

$$\text{Yaw:} \quad \ddot{\psi} + \left[\frac{\omega_0 h}{I_z} + \frac{W\ell}{I_z} \right] \psi - \frac{h}{I_z} \dot{\varphi} = 0 \tag{9}$$

With ω_0 set at 4.66×10^{-3} rad/sec, an order-of-magnitude analysis shows that $\omega_0 h \ll W\ell$. Thus the orbital terms appearing in the roll and yaw equation of a free satellite have been replaced by pendulosity of the DTS.

The coupled equation of this test set-up showed two modes with the periods of 46.9 and 127.5 sec. Computer simulation of the linear equations with an initial roll rate of 0.775×10^{-3} rad/sec (based on an impulse imparted to the DTS) is shown in Fig. 11. Roll attitude motion as measured in the test is shown in Fig. 12. The roll motion clearly indicates a 120 sec mode. The damping observed in the test was computed as an equivalent $\zeta = 0.02$ (second-order system). This is sufficiently low to be attributed to aerodynamic damping and dissipation within the structure of the DTS. Pitch simulations were not run since the uncoupled equations for roll-yaw were the only ones required to support the DTS demonstration. The agreement between theory and experiment is considered quite good. It definitely demonstrated the presence of free nutation as it coupled with the pendulosity dynamics.

Conclusions

The requirement for repetitive scanning of a payload sensor over large angles for long periods in orbit has been filled by exploiting the stable dynamic nutation resulting from the perturbation of a body possessing angular momentum. This stable motion is preserved, rather than damped, by action of the attitude control system. The resulting system is found to be considerably simpler than those employing alternative methods of creating the required scan motion. The major features of the control system are demonstrated on an air bearing test stand using a full size momentum-bias communication satellite development model.

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