

PRECISION ADCS OF A SPINNING SPACECRAFT FOR THE *MARS AERONOMY EXPLORER* MISSION

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The Mars Aeronomy Explorer (MAX) mission is a mission concept to utilize simple state-machine based spacecraft probes separated from two “smart” host spacecraft to significantly enhance the spatial extent and quantity of aeronomy science measurements made in the vicinity of Mars. A key characteristic of the mission concept is the use of two spinning spacecraft probes that must meet pointing requirements < 1 degree of orbit normal and even tighter attitude knowledge requirements while adhering to a simple state-machine based control architecture. This paper discusses a precision attitude and control technique for meeting these requirements utilizing a similar architecture that was adopted for the Laboratory of Atmospheric and Space Physics’ (LASP) SNOE (Student Nitrous Oxide Explorer) spinning spacecraft; SNOE has been operating with its ADCS architecture in low earth orbit (LEO) for over two years. In this case, an open loop control technique to precisely track the precession of the spacecraft’s orbit normal vector has been achieved by commanding an external torque device (magnetorquer) to cycle on and off periodically through a short predetermined sequence of commands uploaded once a day. Due to the lack of a significant magnetic field in the vicinity of Mars, MAX would use a propulsive technique (i.e. vacuum arc thruster or milli-newton hydrazine thruster) for applying the required external torques. The hardware requirements for the proposed technique include the thruster and fuel, horizon crossing indicators (~10cm³), an inertial sensor (i.e. accelerometer), and a state machine controller. All of these components have very small form factors and fit well within the mass/volume requirements for microsats. The pointing technique is based on the fact that major axis spinners are inherently very stable, orbit precessions rates for a 200x10,000 km Mars orbit are low (~1 deg/day), and the external torque environment for precession errors (<0.01 deg/day) and aerodynamically excited nutation (<0.0001 deg) is small. A technique for improving spinner pointing knowledge by making spin vector measurements at multiple orbit locations and using the multiple measurements in conjunction with one another (rather than only relying on ~instantaneous orbit measurements with little “memory”) is proposed. Finally, various techniques for actually implementing the attitude determination and control for MAX are discussed.

Nomenclature

\bar{h}	= spinning spacecraft inertial angular momentum
h_z, h_t	= angular momentum about the primary spin axis (z), a transverse spin axis (x,y) respectively (body)
$\bar{\tau}$	= general external torque
$\Delta\bar{h}$	= change (precession) of inertial angular momentum through application of $\bar{\tau}$
I_z, I_t	= moment of inertia about the primary spin axis (z), a transverse spin axis (x,y) respectively (body)
ω_z, ω_t	= angular spin rate about the primary spin axis (z), a transverse spin axis (x,y) respectively (body)
T	= rotational mechanical energy
t_{torque}	= time over which an external torque is applied
t_{deny}	= during denutation process, time over which h_t resides in a denutation zone
γ	= angle in inertial frame over which \bar{h} is precessed through application of $\bar{\tau}$
λ	= projected angle of h_t in transverse (xy) plane
σ	= total angle in transverse (xy) plane over which a finite time duration torque is applied

ϵ_{torque}	= efficiency of applying momentum in a desired direction when torquing over a finite angle
m_{prop}	= propellant mass consumed during a $\Delta\bar{h}$ maneuver
C_g	= spacecraft center of gravity
I_{sp}	= thruster specific impulse
d	= applied force moment arm for generating $\bar{\tau}$
\bar{g}	= earth gravitational constant
\bar{h}_{meas}	= measured angular momentum vector
$\Delta\bar{h}_{err}$	= uncertainty in angular momentum vector measurement relative to true position
v_α	= true anomaly angle over which attitude predictions are made
β_α	= angle between nadir and local horizon

I. Introduction to Spinning Spacecraft Pointing Errors

Spinning spacecraft generate an angular momentum vector that establishes a reference point for pointing of platform-based instruments. In general, one must deal with the following sources of spacecraft pointing errors: 1) Precession, 2) Nutation, 3) Wobble, 4) Instrument mounting, and 5) Sensor errors.

Precession errors are associated with external spacecraft torques (i.e. aerodynamic torques) changing the orientation (and in some cases magnitude) of the angular momentum vector in inertial space. In the case of a moving reference frame like the slow precession of an orbit LVLH (Local Vertical, Local Horizontal) frame due predominantly to a planet's J_2 perturbation, relative precession errors occur over time if the precession of the spacecraft's angular momentum vector isn't matched to the precession of the LVLH frame.

Nutation errors are associated with the dynamics of how a spacecraft spins about its angular momentum vector. In general, a stable major axis spinner is at its lowest mechanical energy state when the spin axis, angular momentum vector, and eigenvector associated with the spacecraft's maximum moment of inertia are all collinear (see Fig. 2). Any torque disturbance, in addition to precessing the angular momentum vector, also tends to excite transverse rotational energy that causes the spacecraft to appear to "wobble" about its angular momentum vector. Nutation damping or active damping is a technique to remove this residual energy.

Wobble errors are not to be confused with nutation. They are design errors associated with assuming a spin axis orientation that doesn't happen to actually coincide with a spacecraft's moment of inertia eigenvector. These errors, unlike the prior two, can be minimized by careful dynamic balancing prior to launch. Stable spin will never occur about an incorrectly assumed spin axis, and it is for this reason alone that spinners requiring precision ADC should ideally be individually balanced due to subtle variations in spacecraft mass properties (i.e. wire harnesses) between identical spacecraft. At a minimum, a bound on wobble pointing errors should be made by looking at worst case mass variations between spacecraft that have been spin-balanced and those that have not.

Mounting errors are associated with the pointing errors in instrument and sensor orientation relative to a nominal or ideal mounting. These errors are controlled by careful mechanical design, construction, and mounting.

Sensor errors are errors that occur due to such things as bias in a sensor, tolerance for estimating a reference point like a horizon, thermal variations, sensor noise, etc.

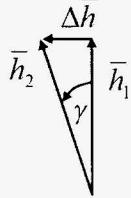
With spinning spacecraft using horizon crossing indicators, standard pointing errors are typically good to ~1 degree. Incorporating multiple measurements over the orbit as will be shown below permits improvements down to uncertainties in the external torque environment. This enhancement will be shown to improve pointing knowledge by potentially many orders of magnitude.

If wobble, mounting, and sensor errors can be designed small relative to knowledge and/or pointing requirements, and if the external torque environment and orbital precession environment are well understood in relationship to ADC requirements, than very precise open loop ADC is possible. In terms of operational robustness, if an operational torque commanding error occurs, the only result is that tracking over a small range was upset and a new set of preprogrammed torques can be uploaded to easily correct for the situation at the next communication opportunity.

To summarize, properly designed major axis spinners are inherently very stable. If the orbital and external torque environment are well understood, then precise target tracking (even open loop) with a simple set of preprogrammed torque sequences (state machine programmable) should be possible.

II. Precession Of Spacecraft Angular Momentum For Orbit Tracking:

External torques, $\bar{\tau}$ (thruster, aerodynamic, etc.), orthogonal to a spinner's angular momentum, \bar{h}_1 , reorient the inertial angular momentum vector to a new orientation, \bar{h}_2 :



$$\Delta \bar{h} = \bar{\tau} \cdot t_{torque} \quad (1)$$

where t_{torque} is the time the torque is active, $\Delta \bar{h}$, is the accumulated error angular momentum orthogonal to spin axis, and γ is the precessed angle caused by $\Delta \bar{h}$. In the case of a moving target like the precession of an orbit plane, an external actuator torque will have to be applied to keep the spacecraft angular momentum vector aligned with the precessing angular momentum vector of the orbit.

For MAX's proposed 200x10,000 km orbit, Fig. 1 illustrates the RAAN precession rate that the spacecraft momentum vector, \bar{h}_0 , would have to track. As can be seen, the tracking maneuver requires $\sim 1^\circ$ per day of actuated precession. This is consistent with the demonstrated LASP SNOE's $\sim 1^\circ$ per day precession maneuver to track the sun in a sun synchronous orbit over the course of a year. For the current MAX spinner design of 4.46 kg·m² moment of inertia about the spin axis and nominal 6 rpm spin rate, a $|\Delta \bar{h}|$ of 0.05 N·m·s of momentum would have to be applied on a daily basis for orbit tracking. Given the orbit period of 6.6 hours, $<0.35^\circ$ would have to be applied per orbit which is well within the requirements for MAX's instrument pointing requirement if, worst case, the maneuver were to only be done with horizon indicator reference over periapsis passes.

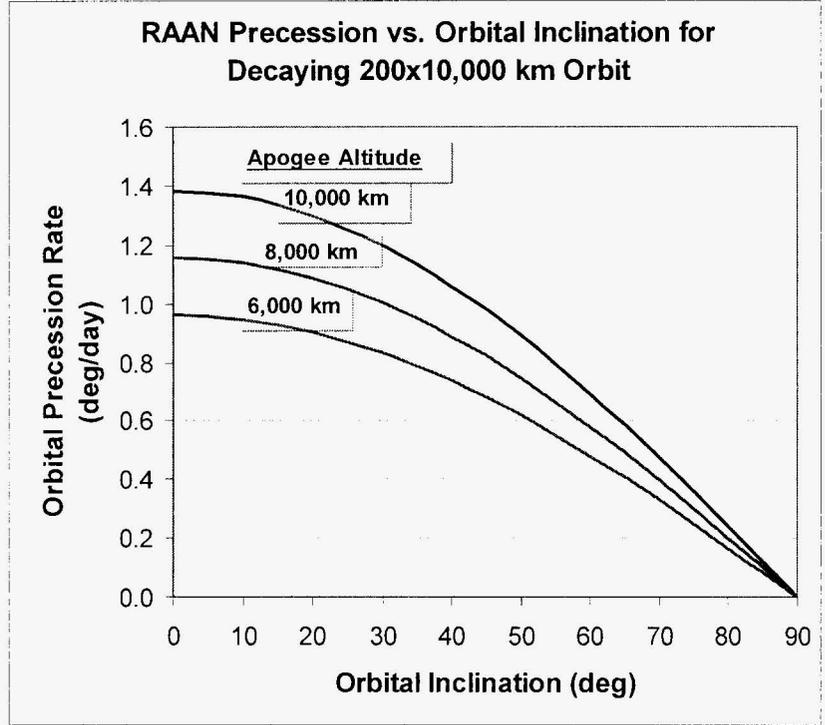


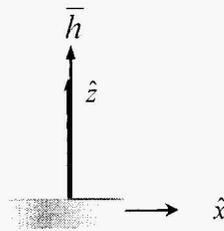
Figure 1. RAAN orbital precession for a decaying 200x10,000 km orbit at Mars as a function of orbital inclination angle.

III. Dissipating Nutation Through Active Damping

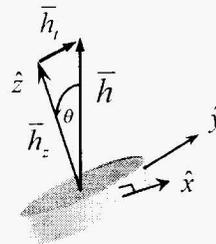
Nutation is excited anytime an external torque is applied. In the spinner's reference frame, the external torque momentarily offsets the momentum vector from the true spacecraft spin axis, \hat{z} , by a transverse component of angular momentum, \bar{h}_t , at which point natural spin dynamics (described by Euler's equations) cause the angular momentum vector to orbit \hat{z} in the body frame (Fig. 2). \bar{h}_t is directly associated with a slight increase in the overall rotational mechanical energy of the spinner. Therefore, mechanically dissipating energy is equivalent to

dissipating \bar{h}_t (while still conserving \bar{h}) which is causing the nutation. Note, I_z and I_t , and ω_z and ω_t are the spin axis and transverse moment of inertia's and spin rates respectively.

(1) Simple Spin



(2) Nutating Spin



Same Inertial Angular
Momentum Vector:

$$\bar{h} \cdot \bar{h} = I_z^2 \omega_{z1}^2 = I_t^2 \omega_t^2 + I_z^2 \omega_{z2}^2$$

Comparison of Mechanical Energy, T ,
Between Scenarios:

$$2T_1 = I_z \omega_{z1}^2$$

$$2T_2 = I_t \omega_t^2 + I_z \omega_{z2}^2$$

$$2T_1 = \frac{I_t}{I_z} I_t \omega_t^2 + I_z \omega_{z2}^2$$

For a major axis spinner $I_z > I_t$

$$\therefore T_2 > T_1$$

In the Spacecraft Body Frame, \bar{h}_t orbits \hat{z} at a rotation rate, $\dot{\lambda} \cong \frac{(I_z - I_t)\omega_z}{I_t}$:

In the case of asymmetric transverse properties, the \bar{h}_t orbit is elliptic instead of circular with a slight variation in rotation rate over its period.

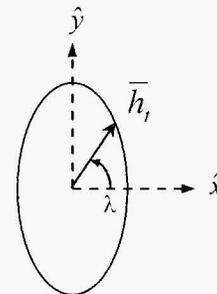


Figure 2. Nutation Dynamics of a Spinning Spacecraft

Passive nutation energy dissipation involves mechanical energy damping concepts such as fluid filled rings, eddy current dissipaters, etc. Unfortunately, these damping mechanisms for nutation angles < 1 degree are not strong and nutation after excitation (i.e. via an aerodynamic torque) can last a long time. In such a scenario active damping is preferred, and in fact, with some simple timing, the same thruster firing used to precess the angular momentum vector can also be used to damp excited nutation. The simple control criteria for this operation is that a thruster firing occurs whenever the torque vector is properly aligned (within a tolerance in the inertial frame) to the desired precession direction and the transverse component of angular momentum in the body frame is in a quadrant directly opposite the body frame thruster torque vector. If these two criteria are met, a single thruster pulse will both precess the angular momentum vector in the desired inertial direction and simultaneously dissipate residual nutation energy caused by actuated precession to a minimum \bar{h}_t of approximately one thrusting torque impulse bit. For the MAX spinner case this minimum \bar{h}_t equates to a nutation error of: ~ 0.00002 degrees for a $1\mu\text{N}$ Vacuum arc thruster, or ~ 0.0002 for the JPL milliNewton thruster. In reality, the denutation error will be limited by a sensor's ability to detect it.

The denutation portion of this technique is based on the classical approach used for active nutation damping in spacecraft and launch vehicle upper stages and is very easy to implement.¹ Figure 3 graphically illustrates the nutation damping control profile.

The duty cycle time, t_{duty} , that \bar{h}_t spends in the denutation zone (and subsequently outside of the denutation zone) can be closely estimated by Eq. (2) for axisymmetric major axis spinners:

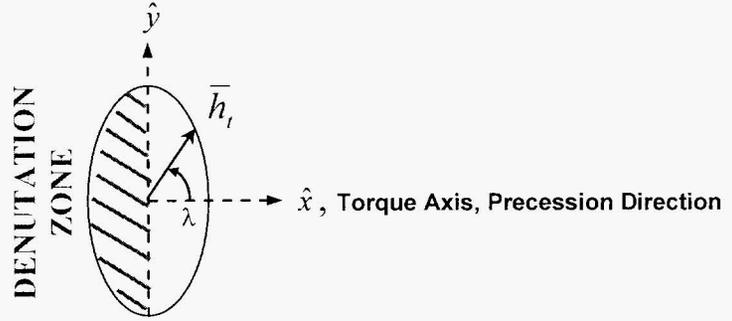


Figure 3. Active Nutation Damping Control Law in Body Frame

$$t_{duty} = \frac{\pi I_t}{(I_z - I_t)\omega_z} \quad (2)$$

where I_z is the spin axis moment of inertia ($\text{kg}\cdot\text{m}^2$), I_t is the transverse moment of inertia ($\text{kg}\cdot\text{m}^2$), and ω_z is the spin rate (rad/s). For the current MAX spinner design, t_{duty} is ~ 29 seconds compared to the spin period of 10 seconds, meaning thruster firings can occur continuously over ~ 2.5 rotations before having to stop for ~ 2.5 rotations. The denutation zone is easily detected with an inertial sensor (i.e. radially mounted accelerometer oriented along \hat{x} axis actually detecting the closely aligned ω_t) and can be hardwired into a simple control circuit that would simply detect the sign of the signal for determining one of the criteria for thruster firing.²

In summary, no extra fuel has to be budgeted for controlling nutation (to $\ll 1$ degree) caused by the orbit-tracking precessing torques since the torques are implemented in a fashion to prevent measurable nutation growth. The cost for this simple active damping architecture is the requirement to constrain thruster precession operations to a $< 50\%$ duty cycle. In reality, actual precession maneuvering will likely dictate a shorter duty cycle ($\sim 25\%$) to maintain efficient utilization of propellant (see section IV) during thrusting. This duty cycle constraint is not significant given the available thruster control authority over the course of a day (> 55 degrees/day for the vacuum arc thruster, $>> 100$ degrees/day for a milli-newton hydrazine thruster as compared to the ~ 1 degree/day orbit tracking precession requirement).

IV. Actuator and Propulsion Efficiency while Torquing over Finite Angles

To possibly take advantage of higher I_{sp} , lower torque actuator options both for precession and denutation purposes, it is worth considering the effective torque efficiency, ϵ_{torque} , defined as the integrated average percent of actuator torque, $\tau_{actuator}$, that is applied in a desired direction (inertial or body frame given the typically small nutation angles) when torquing over a finite time (and subsequently finite torquing angle). Fig. 4 illustrates ϵ_{torque} as a function of the angle over which $\tau_{actuator}$ is applied continuously. Knowing ϵ_{torque} , the total maneuvering propellant mass, m_{prop} , for a mission can be estimated:

$$m_{prop} = \frac{\Delta h}{\epsilon_{torque} I_{sp} g d} \quad (3)$$

where Δh , in this case, is the total angular momentum that must be supplied in the mission for precession and nutation; I_{sp} is the specific impulse of the propulsion system, d is the momentum arm of the thruster from the spacecraft center of mass, and g is the normalization constant of 9.81 m/s^2 (earth surface gravity).

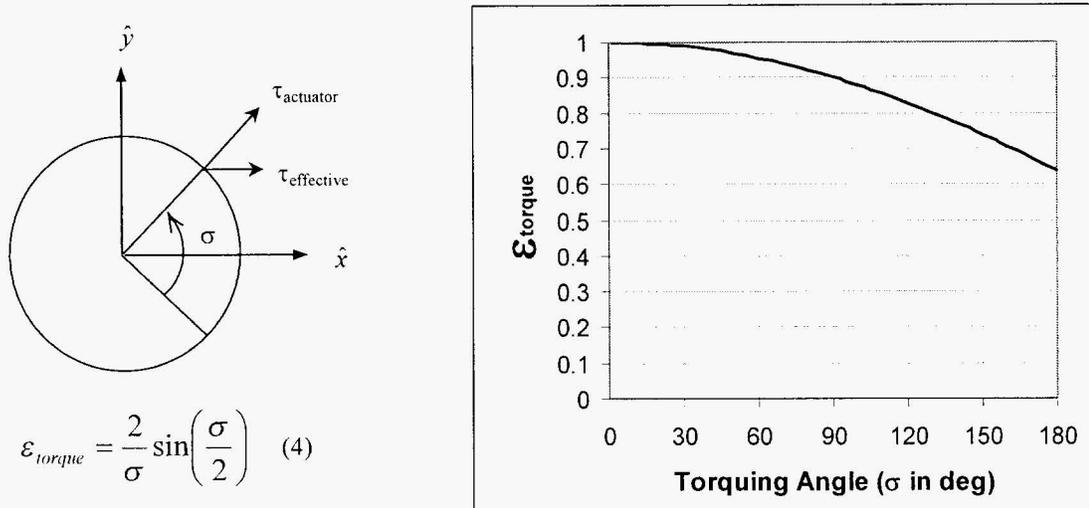


Figure 4. Effective Torque Efficiency over Large, Finite Torquing Angles

V. External Torque Disturbances - Aerodynamic Torques

External disturbance torques have two influences on the spacecraft spin dynamics and resultant pointing errors: 1) undesired precession of the spacecraft's angular momentum vector, and 2) excited nutation. For the MAX mission, Mars' most significant external disturbance torque is associated with aerodynamic drag encountered during high speed, low altitude periapse passes. Eq. (1) allows one to estimate the worst case inertial precession of \bar{h} . Figure 5 illustrates the process for estimating the maximum excited nutation angle for a ~constant inertial torque relative to the spacecraft spin rate and no consideration of nutation damping.

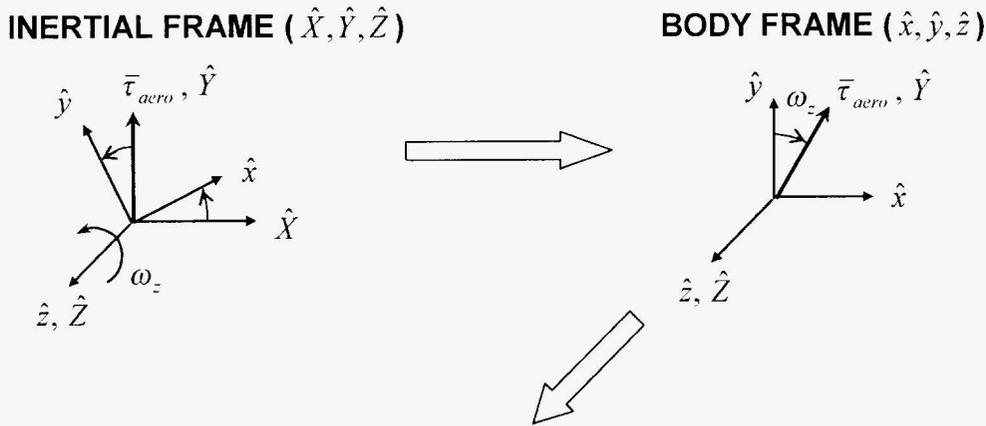
For a very large 10cm Cp/Cg offset along the spin axis, and the MAX spinner properties at 6 rpm, the maximum angle precessed for a constant aerodynamic force of $1\mu\text{N}$ over a 60 min period is 0.007° . Furthermore, due to the rotating nature of this disturbance momentum in the inertial frame, the final precessed angle after exiting the atmosphere is actually $< 0.007^\circ$. The maximum excited nutation angle over this same aerodynamic encounter is $< 0.000003^\circ$. (It's worth noting that Cp/Cg offsets in the spin plane (xy) cause negligibly small oscillations in the spin rate, but don't induce nutation or precession and hence are not included in this analysis).

The control architecture for the MAX spacecraft is dominated by orbit tracking of the orbit's RAAN precession ($\sim 1^\circ/\text{day}$). Active nutation damping during precession should easily compensate for nutation growth both during precession maneuvers and during periapse atmospheric interactions. No extra propellant is required for nutation control. Mission propellant budgeting can therefore, be calculated from eq (4) realizing $\Delta h = \gamma I_z \omega_z$ where γ , in this case, is the total orbit RAAN precession angle over the course of the mission (in radians).

VI. Precision Attitude Determination With Sensor, Mounting, And Wobble Errors

With typical spacecraft pointing requirements of $< 1^\circ$, sensor mounting errors, and wobble errors become potentially significant sources of error (as previously noted, nutation error for MAX as long as the error is not allowed to accumulate is extremely small). To give a specific example, Figure 6 illustrates a candidate miniature horizon crossing indicator (HCI) manufactured by Servo for Sandia National Labs. As shown, the F.O.V alignment to the housing is by itself responsible for a body fixed pointing error of up to 0.5° . Figure 7 illustrates a Matlab simulation showing the projected HCI ground tracks for a 1° roll angle. Figure 8 illustrates the roll angle measured vs. spin angle traverse difference between HCI 1 and HCI 2 (same configuration as shown in Fig. 7) as a function of altitude over a circular Mars (mean radius = 3397.4km).

It's important to note that similar to wobble and mounting errors, the internal errors associated with the construction of the sensor are all errors that are fixed relative to the spacecraft body (other than very small scale thermal variations that are $<< 1^\circ$). Body fixed errors can be calibrated out in flight with the technique described below. The exception to this is an instrument requiring accurate pointing. This instrument MUST be mounted well



EULER EQUATIONS

$$\dot{h}_x = \frac{(I_t - I_z)}{I_t I_z} h_y h_z + \tau_{aero} \sin \omega_z t$$

$$\dot{h}_y = \frac{(I_z - I_t)}{I_t I_z} h_x h_z + \tau_{aero} \cos \omega_z t$$

$$\dot{h}_z = 0$$

$$\text{Solution is } h_t = \frac{I_t \tau_{aero}}{I_z \omega_z} (-\cos \omega_z t \hat{x} + \sin \omega_z t \hat{y})$$



NUTATION ANGLE. θ

$$\tan \theta = \frac{|h_t|_{\max}}{h_z} = \frac{I_t \tau_{aero}}{I_z^2 \omega_z^2}$$

$$\text{Valid for spin rates } \gg \text{ precession rates } \Rightarrow \frac{1}{\omega_z} \sqrt{\frac{\tau_{aero}}{I_z}} \ll 1$$

Figure 5. Calculating Nutation Excited by Constant Inertial Torque (close approximation of ~inertially fixed aerodynamic torque relative to spin rate)

within the pointing requirement relative to the spin balanced true spin axis since there is no general way to measure instrument pointing performance after launch. It is highly recommended that this mounting requirement is tested and verified prior to flight.

Normal real time attitude determination with horizon crossing indicators tends to neglect the fact that the spacecraft angular momentum is relatively fixed in inertial space for most orbit applications. Given the extremely small external disturbances associated with aerodynamic torques that MAX can be expected to experience as described above, attitude determination may be significantly improved by including in the state estimator the known



Responsivity	15 Hz, 14-16 μ m, 1200 v/w
Field of View, full angle	4.0 $^\circ$
F.O.V alignment to housing	+/- 0.5 $^\circ$
Frequency Response	Rolls off @20dB/dec. at 250 Hz.
Noise voltage	(25,15,1) \leq 500 nV/ \sqrt Hz
Operating Voltage	3-18Vdc, 9Vdc typical
Operating Current	10 μ Amps typical
Operating Temperature	-40 to 85 $^\circ$ C

Figure 6. Example Miniature Horizon Crossing Indicator Specifications

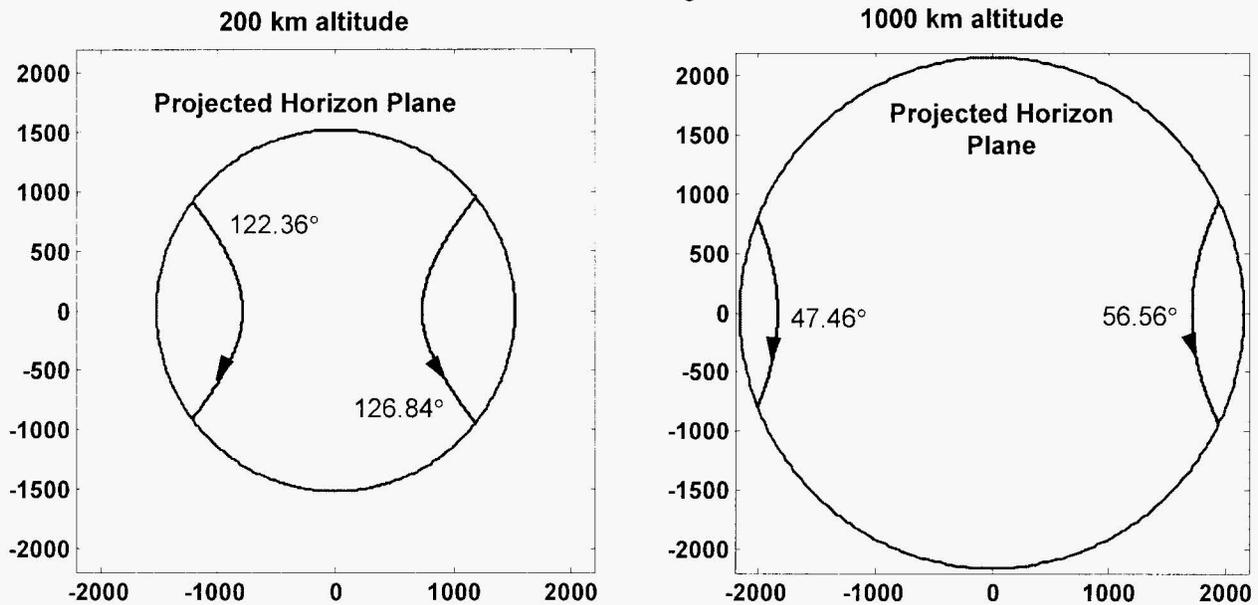
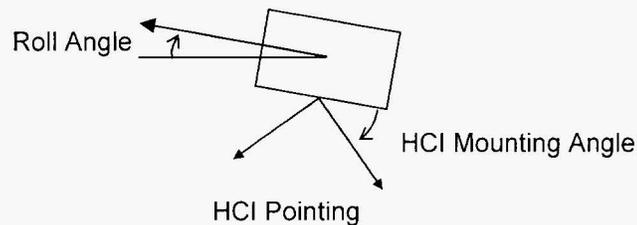


Figure 7. HCI Spin Angle Measurements for 1 $^\circ$ Roll Angle Pointing Error as a Function of Altitude for a 45 $^\circ$ HCI Mounting Angle.

dynamics of the precessing orbit and position in orbit where the measurement is being made while neglecting the very small and negligible effects associated with aerodynamic disturbances. With these assumptions the potentially much larger constant error sources (relative to body) associated with wobble, sensor, and mounting, $\Delta \bar{h}_{err}$, may be significantly filtered out. Figure 9 illustrates conceptually this attitude determination scheme for a non-precessing circular orbit. Figure 10 extends this scheme to include the effects of orbit eccentricity and orbit precession.

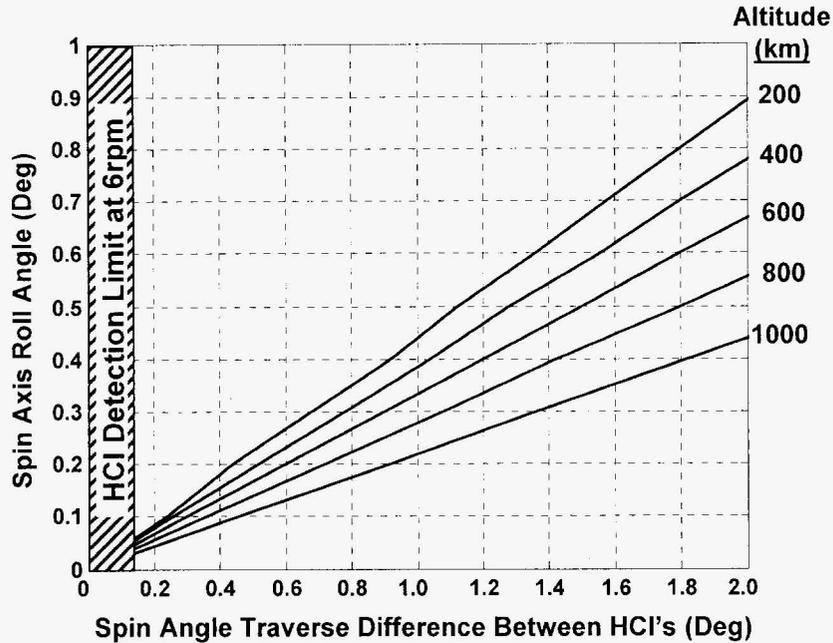


Figure 8. Spin Axis Roll Angle Vs. Spin Angle Traverse Distance Between HCI's (with HCI Frequency Response Cutoff at 6rpm)

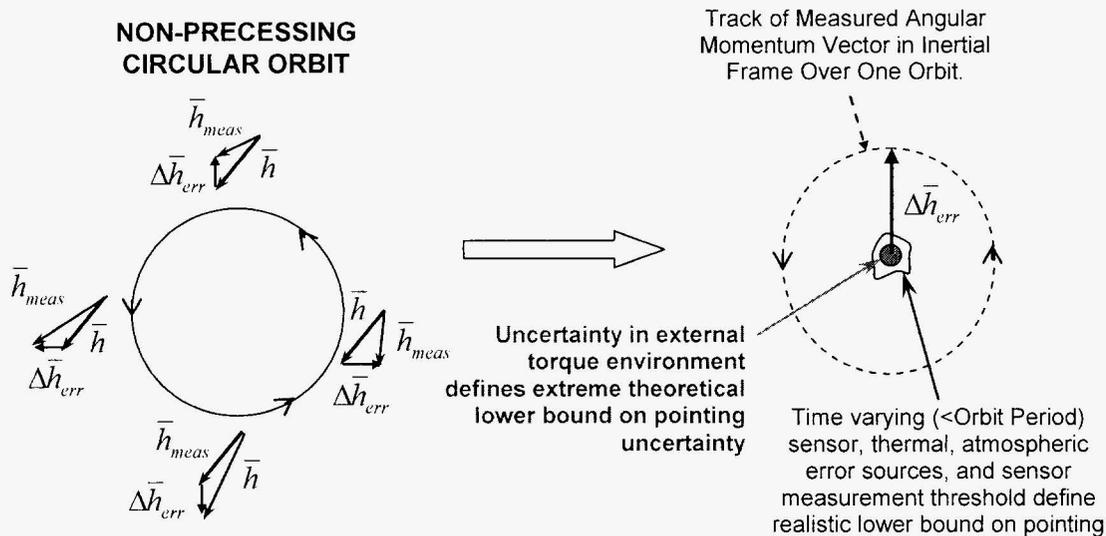
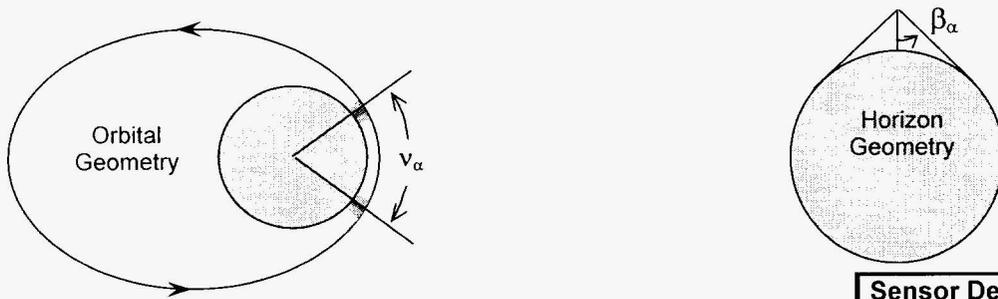


Figure 9. Precision Attitude Determination and Body-Fixed Error Source Filtering for a Non-Precessing Circular Orbit Incorporating Measurements over Multiple Locations in Orbit

Included in Figure 10 is an estimated optimal band of attitude estimating traverse arcs shown in shades of blue. The optimal arc appears to be $\sim 80^\circ$ - 100° based on the number of statistically significant attitude measurements (>100) that can be made with a 6rpm spinner, the angular spread for being able to triangulate the position of the true angular momentum vector, and a minimal variation in the angle of the horizon from the LVLH horizontal plane (based on altitude changes over the pass) for purposes of choosing a horizon crossing sensor and design mount. Given the relatively short periapse pass (~ 20 minutes), the amount of orbit precession (for 1° per day RAAN precession) is $\sim 0.02^\circ$ which tends to look small relative to estimating the actual position of \bar{h} (although this effect can easily be included to increase pointing resolution). Based on these results, to first order it appears very reasonable to be able to make attitude measurements better than the required pointing resolution.



						Sensor Design Constraint
Attitude Estimation Arc (v_a)	Arc Traverse Time	Number of Attitude Measurements at 6rpm	1deg/day RAAN Precession over Arc	Peak Altitude	Nadir to Horizon Angle (β_a)	Max Variation in β_a over Arc
(deg)	(min)		(deg)	(km)	(deg)	(deg)
0.0	0.0	0	0.000	200.0	70.8	0.0
10.0	2.4	15	0.002	205.0	70.6	0.2
20.0	4.8	29	0.003	220.1	69.9	0.9
30.0	7.3	44	0.005	245.4	68.8	2.0
40.0	9.8	59	0.007	281.1	67.5	3.4
50.0	12.4	74	0.009	327.6	65.8	5.0
60.0	15.0	90	0.010	385.4	63.9	6.9
70.0	17.7	106	0.012	454.8	61.9	8.9
110.0	29.9	179	0.021	864.7	52.9	17.9
120.0	33.4	200	0.023	1005.1	50.5	20.3
130.0	37.1	223	0.026	1163.0	48.2	22.6
140.0	41.2	247	0.029	1340.1	45.8	25.0
150.0	45.5	273	0.032	1537.9	43.5	27.3
160.0	50.3	302	0.035	1758.2	41.2	29.6
170.0	55.5	333	0.039	2003.0	39.0	31.8
180.0	61.2	367	0.042	2274.4	36.8	34.0

Example Measured Angular Momentum Track In Inertial Space over a Periapse Pass with Constant Error Sources (Relative to Body)

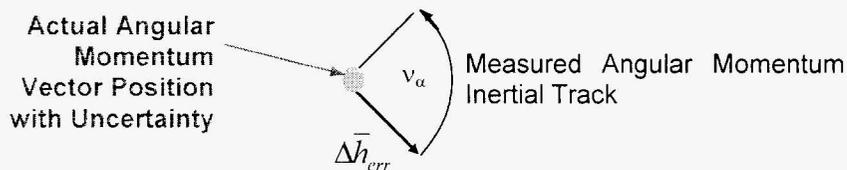


Figure 10. Precision Horizon-based Attitude Determination and Error Source Filtering for a Precessing Elliptic Orbit Incorporating Measurements over a Periapse Pass for a Mars 200x10,000 km orbit (circular planet)

VII. Implementation Strategy

Based on the fact that the pointing requirement is approximately equal to the required precession angle per day to track the orbital RAAN precession, it would be very reasonable to control pointing of the spacecraft open loop since the precision for tracking can be relatively low on a daily basis and easily updated for traverse errors in following days (see Figure 11.) Alternatively, this maneuver can also be done potentially in a state machine controller with occasional ephemeris/or clock updates (~1 second accuracy). Either method would require the following control parameters:

1. Time for the beginning of Pulse Train based on a position in orbit (or clock time)
2. Individual actuated pulse timing based on spin cycle period, spacecraft mass properties, nutation control criteria with inertial sensor (i.e. accelerometer), and occasional inertial updates (~once per several days). This entire process doesn't need to be highly accurate given Fig. 4, Fig. 11, and an individual thruster pulse precision of $<0.001^\circ$.
3. Pulse Width vs. Pulse Train Duration based on Fig. 4 for propellant budgeting .

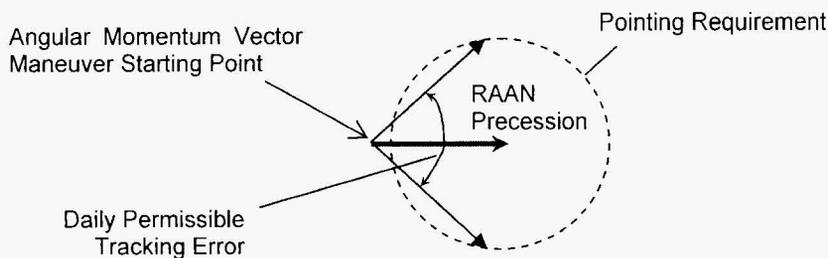


Figure 11. Daily Permissible Tracking Error with Required RAAN Precession and Pointing Requirement

VIII. Conclusions

This study supports the conclusion that a spinning spacecraft for the MAX mission can relatively easily be pointed to within a 1 degree pointing requirement given the extremely small external disturbances and the fact that body-fixed sensor errors can be averaged out so that attitude estimates consistent with the pointing requirement can be made. The study is incomplete in terms of bounding absolute attitude knowledge capability for such a spacecraft using HCI's. A first order look suggests that attitude knowledge limited by the HCI's cutoff frequency will be no better than $\sim 0.15^\circ$. This error is proportional to the spin rate so that it can be improved by spinning slower. Given the extremely small external torque environment, it is reasonable to spin significantly slower before precession and nutation errors become issues. The trade of spin rate vs. instrument measurements and navigation updates over a precession pass is worth considering in the future.

The bound on attitude knowledge uncertainty associated with Mars' physical oblateness is yet to be determined. However, modeling this uncertainty based on the recent MOLA data to fit a Mars ellipsoid³ is a recommended next step to complete this work.

Acknowledgements

The research described in this paper was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration. The authors would also like to specifically thank Frank Jordan, Stu Kerridge, and Bob Easter of the Mars Program Office for supporting this work.

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