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Thrust Vector Control Algorithm Design for the Cassini Spacecraft

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A preliminary design of the Cassini Thrust Vector Control algorithm which controls the spacecraft attitude during main engine burns is described. The discussion includes software architecture, sensor/actuator characteristics, and vehicle dynamics, as well as controller design and margin analysis via classical methods, and performance evaluation via simulation. Special attention is paid to actuator modeling, propellant dynamics, and to potential controller interactions with structural flexibility and propellant slosh.

Introduction

The Cassini spacecraft (Fig. 1) is designed for a four-year orbital tour of the Saturn system, with delivery of the Huygens Probe (ESA) into the atmosphere of Saturn’s largest moon, Titan. Cassini is scheduled for an October 1997 launch on a Titan IV with the Centaur upper stage. The mission design includes gravity-assist flybys of Venus, Earth, and Jupiter, and eventual rendezvous with Saturn in June 2004.

The interplanetary cruise trajectory requires a large maneuver between the Venus flybys, and as many as twenty smaller trajectory correction maneuvers (TCMs) for navigation. The orbital phase begins with the long Saturn-orbit insertion burn just after periapse, followed by a periapsis raising maneuver at apoapse, which establishes the initial tour orbit. The probe is released on a Titan impact trajectory, and after performing a small deflection maneuver, the orbiter records the probe data as it enters Titan’s atmosphere.

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The orbital tour includes many small TCMs, targeting the orbiter for flybys of Titan and the icy satellites Rhea, Dione, Iapetus, and Enceladus.  

Almost all of the required $\Delta v$ is delivered by the (selected) main engine, a 490 N bipropellant thruster, which burns MM11 and $N_2O_4$. (There are two identical main engines for redundancy.) At launch, the total bipropellant load is 3000 kg, which is 60% of the spacecraft's launch mass. The separate Reaction Control System (RCS) consists of sixteen 0.65 N monopropellant thrusters (eight redundant pairs) fed by a single tank which contains 132 kg of hydrazine at launch. The RCS is used for cruise attitude control and for momentum management during the orbital phase when the reaction wheels are in use. The system can also be used for small $\Delta v$ maneuvers. Each main engine is mounted to the "bottom" of the spacecraft in a gimbaling system, allowing two axes of articulation for thrust vector control (TVC) during burns. The third axis (roll about the thrust vector) is controlled using the RCS thrusters.

A variety of propulsive systems have been used for J11 interplanetary orbiters. Recently, the Magellan spacecraft was inserted into Venus orbit using a STARR-48 with 100 lbf hydrazine thrusters for attitude control during the burn. Mars Observer will use two (of four) fixed 110 lbf biprop engines, attitude control being provided by smaller biprops. Finally, the dual-spin Galileo spacecraft will use its single 400 N biprop for Jupiter-orbit insertion, relying on spin stabilization at 10 rpm. The heritage of the current design goes all the way back to Mariner 9, which used a gimbaled 300 lbf engine for insertion into Martian orbit. (The Viking Orbiter design was similar.) For a three-axis stabilized spacecraft, the primary advantage of the gimbaled engine design is that it eliminates the need for intermediate-sized thrusters dedicated to attitude control during the burn. The block-redundant Cassini design also provides single-fault tolerance for the engine and its gimbal actuators, which must be capable of performing as many as 200 burn over the 11-year mission.

This paper describes a preliminary design of the Cassini TVC algorithm. Although the control logic itself is ultimately distilled into a few lines of code scattered over several software modules, the analysis of the "algorithm" as a functional unit necessitates the definition of the context in which the controller must perform. The bulk of the work involves modeling of sensors, actuators, and vehicle dynamics, understanding software characteristics, especially timing issues and I/O functions, and formulating a concise definition of what the algorithm is supposed to do, and how accurately it needs to do it. (This can be more difficult than it sounds.)

First, the overall software "architecture" is described, at least the subset which supports the TVC algorithm. This is followed by a discussion of hardware modeling and vehicle dynamics. After control law design and margin analysis via classical methods, simulation results are presented and evaluated in the context of the maneuver error or requirements.

**TVC Architecture**

Fig. 2 is a data flow diagram for the TVC algorithm. The shaded blocks represent software "objects," while the white blocks are functions in the attitude controller object that comprise the TVC algorithm. Starting from the top left, the attitude commander provides the commanded spacecraft quaternion $q^c$ which aligns the selected main engine with the desired $\Delta v$ at the initial engine articulation. (This articulation will have been calculated to point the thrust vector at the center-of-mass.) The commanded rates $\omega^c$ are zero except during the insertion burn where a slow turn is used to minimize finite-burn $10$ sSCS. At the bottom-left the attitude estimator uses star tracker and gyro data to propagate the spacecraft attitude with respect to the celestial reference frame. (During burns star tracker data may not be available, in which case the attitude is propagated inertially.) The estimator provides the quaternion $q^e$, and the body rates $\omega^b$, which the attitude controller compares to the commanded attitude and rate. The attitude error is decomposed into orthogonal small -angle errors and transformed to a coordinate system which has its 3-axis along the centerline of the selected main engine at the initial articulation. Let $C$ be the direction cosine matrix of this frame with respect to the spacecraft, so that the transformed position and rate error vectors $p$ and $\tau$ become:

$$p = 2C^T \xi$$

$$\tau = C^T (\omega^c - \omega^b)$$

where $\xi$ is the error rotation vector computed from the error quaternion:

![Figure 2 TVC Data Flow](image-url)
\[ \xi_i = e_q^c_i e_q^c_4 \quad (i = 1,3) \]  

(3)

with \(e_q^c = (q^c)^* q^c\). (The asterisk denotes conjugation, and juxtaposition denotes quaternion multiplication, i.e., successive rotation.)

The 1- and 2-components of the position and rate errors drive two single-axis TVC controllers, which output acceleration requests \(a = [a_1 a_2]^T\). (The design of the controller is discussed at length below.) The decoupler transforms these requests into engine rotation commands \(\beta = [\beta_1 \beta_2]^T\), using spacecraft mass properties and engine characteristics:

\[ \beta = \frac{f^a}{r} \]  

(4)

where \(f^a\) is the upper left 2x2 partition of the inertia matrix transformed into engine coordinates, \(F\) is the main engine force, and \(r\) is the distance from the main engine to the spacecraft center-of-mass. The rotation commands are sent to the EGA manager, which computes the necessary extension commands for the linear Engine Gimbal Actuators. These angles are also fed back to the guidance, loops of the TVC controllers (discussed below). The 3-component position and rate errors drive the, roll control logic, which schedules RCS thruster activity with the thruster manager.

### Sensor/Actuator Models

The strap-down integrating gyro package is mounted to the upper equipment module (Fig. 1). Every computational cycle (125 ms) the gyro manager reads the Linlc-tagged angular increments accumulated since the previous read. (One pulse corresponds to 8 rad.) The attitude estimator propagates the spacecraft quaternion using a second-order expansion of the quaternion kinematic equation, and derives rate using the first-order difference. The dynamics of the gyro rebalance loop arc modeled as a damped second-order system with a 5 Hz bandwidth, and an integral pole at 0.5 Hz. While noise inputs in acceleration and rate integrate to rate and position random walks. Although the simulation model includes the pulse-generation logic, the linearized “design model” ignores the quantization altogether, and being negligibly small in the TVC environment. The transfer function representation is as follows:

\[ H_{gyro}(s) \cdot Z(s) = \frac{-b s + a}{s^2 + c s + d + b s + a} \]  

with \(a = 3102. \text{rad/s}^3\), \(b = 1118 \text{rad/s}^2\), and \(c = 89.2 \text{rad/s}\). (Note that star tracker data is available during the burn. It will be used to update the inertially-propagated attitude, with a probable one-second update frequency. Closed loop performance with filtered celestial updates has not yet been simulated.)

Figure 3 depicts the Main Engine Assembly (MEA), with the two independently-gimbaled 490 N engines each controlled by two linear Engine Gimbal Actuators (EGA’s). An EGA is a ballscrew assembly driven by a brushed DC motor, and each is mounted to the thrust plate via a universal joint. The output shaft is connected by a pin joint to the engine above the gimbal plane, so that extension or retraction of the actuator causes the engine to rotate in its gimbal system.

**Figure 3**  
**Main Engine Assembly**

The exact relationship between actuator extensions \(\varepsilon_1\) and \(\varepsilon_2\), and the gimbal angles \(\delta_1\) and \(\delta_2\), is as follows:

\[ \varepsilon_i = \sqrt{d_{2}^2 - 2 d_{2} \varepsilon_i - d_{2}} \quad (i = 1,2) \]  

(6)

with

\[ q_1 = \frac{d_{1}^2 (1 - \cos \delta_2) + d_{1} d_{3} \sin \delta_2 \cos \delta_1}{d_{2}} + \frac{d_{1} \sin \delta_1 \cos \delta_2}{d_{2}} \]  

(7a)

\[ q_2 = \frac{d_{1} \sin \delta_1}{d_{2}} + \frac{d_{1} \sin \delta_1 \sin \delta_2}{d_{2}} - \frac{d_{1} d_{3} \sin \delta_1 (1 - \cos \delta_2) - d_{2} d_{3} \cos \delta_1 \sin \delta_2}{d_{2}^2} + \frac{d_{3}^2 (1 - \cos \delta_1 \cos \delta_2)}{d_{2}^2} \]  

(7b)

where \(d_{1}\) is the radial offset of the EGA pin joint from the engine centerline, \(d_{2}\) is the actuator null extension, and \(d_{3}\) is the “lever arm,” i.e., the distance from the pin
joint to the gimbal plane. The forward kinematics (6-7) can be approximated as follows:

\[ \varepsilon_1 = \frac{d_3 \delta_1 + \frac{d_1^2}{2d_2} \delta_2^2}{2} \quad (8a) \]

\[ \varepsilon_2 = \frac{-d_3 \delta_2 + \frac{d_1^2}{2d_2} \delta_1^2 + d_1 \delta_1 \delta_2}{2} \quad (8b) \]

Fig. 4 shows the contours of constant cone angle from the null orientation mapped to EGA extension space by the exact (solid) and approximate (dashed) forward kinematics. (The ±0.6 inch bounds are the extension limits of the actuators.) Note the significant distortion (from perfect circles) caused by the fairly large radial offset \( d_1 \), and also try the attachment of the outer-gimbal actuator to the engine rather than to the gimbal ring.

![Figure 4](image_url)

**Figure 4**

EGA Extensions for Constant Cone-Angle

Contours of 1°, 3°, 5°, etc.

Every computational cycle, the EGA manager receives engine rotation commands (8) from TVC, and converts to gimbal angles. (This involves a twist of approximately 45° for the current MEA layout.) The gimbal angles are then converted to extensions using the approximate relations (8), and 12 bit extension words are written to the Engine Gimbal Electronics (EGE). The EGE includes a fast digital loop which controls actuator extension using feedback from an LVDT which is integral to the actuator ballscrew. This is a loud-compensated high-gain system with forced saturation, and includes an integral-like term to hold the engine against reflected forces from wind-up of the propellant flex lines. The motors are driven by a binary rate modulator (BRM), which converts the 7 bit torque word into a pulse train. The sample time is 800 μsec, with a minimum BRM pulse-width of 4 μsec, allowing the EGE/EGA servo to be modeled as a continuous-time process for TVC. (The simulation model is designed to run with a 5 ms time step for a fourth-order Runge-Kutta integrator, and is not intended to capture dynamics much beyond 20 Hz.)

The actuator dynamics model is essentially a double integrator with a rather elaborate set of enhancements, including back-EMF, viscous friction, a Dahl model for ball-screw friction, a hysteretic model of the reflected propellant flex line forces, and a model of tile compliance and backlash of the "softmount" which connects the actuator to the thrust plate and provides isolation from launch loads. The actuator dynamics are integrated separately, with tile output extensions and extension rates being converted to gimbal angles and angular rates for the spacecraft dynamics model, where the engine articulation degrees of freedom are treated as prescribed motion. This approach captures the one-way coupling from the dynamics of the actuator/engine system to the spacecraft dynamics, both through the articulating force vector, and also through inertial torques. (The coupling in the other direction is negligible.) Note that the conversion of extensions (and rates) to gimbal angles (and rates) requires an inversion of the kinematic relationships (6-7) in the simulation software. This is accomplished using a few Newton iterations on an initial guess extracted from an approximate inversion of (8).

Fig. 5 shows the response of the system (the EGA Manager, the EGE/EGA servo, and the MEA kinematics) to a 5° step command in the outer gimbal. The response is rate-limited (primarily due to back-EMF). Note the small hang-off due to the softmount compliance, and also the "cross-axis" response on the other gimbal due to coupling. This system is fast enough to be almost transparent to the TVC algorithm, which generates command profiles that are very smooth on this kind of time scale. For the linear
design model, the system was represented by a first order lag, whose phase shift was matched to the simulated response to a 11 Hz sinusoid with an amplitude of 1.\(^\text{\textdegree}\). The phase lag, which was taken as the phase shift of the fundamental Fourier component, was 12.7\(\text{\textdegree}\), which prompted the following transfer function representation:

\[
H_{EG\alpha}(s) = \frac{1}{T_{a}s + 1} \tag{8}
\]

with \(T_{a} = 0.036\) s (4.4 Hz break).

The thrust profile of the main engine is modeled as an exponential rise to the steady-state thrust level with a 40 ms rise time. The RCS thrusters are modeled similarly with a rise time of 90 ms, and a decay time of 0.50 s.

**Spacecraft Dynamics**

The spacecraft structure is designed around the propulsion module, which houses the two bipropellant tanks (cylindrical with hemispheric heads) in a stacked configuration. Each tank includes a central-sponge type propellant management device (PMD), which exploits surface-tension forces to control the propellant center-of-mass under quiescent conditions. The upper and lower equipment modules attach to the propulsion module (Fig. 1), forming the central structure of the spacecraft, which can be considered rigid in this analysis. Most of the longeron-supported equipment is also very stiff, with resonant frequencies typically 10 Hz and above. Exceptions are the 10-meter magnetometer boom, whose first bending mode lies near 0.7 Hz, and the light RPWS antennas, with bending modes as low as 0.08 Hz.

For the current study, it is adequate to model the spacecraft as a rigid "basebody" with spring-restrained appendages for the mag boom and RPWS antennas. The spring constants are chosen to match available finite-element data. Pendulums are attached to the base body (along the tank centerlines) to model the propellant slosh under the main engine acceleration. (When the main engine is not on, a different type of propellant model is used, which mimics the "centering" effect of the PMD in the low-g environment.) The pendulum mass, length, and attachment point are computed using a contractor-supplied program which solves the linearized fluid-dynamic equations using an eigenfunction expansion technique, and derives the mechanical-analogy model parameters for the first lateral mode. Under the main engine acceleration, the bipropellant pendulums oscillate near 0.1 Hz. The selected main engine is also modeled as an appendage whose articulation is dictated by the EGA dynamics model discussed above. For simulation, the system equations of motion were generated by a symbolic manipulation code. In future studies, a higher-fidelity model will be used, which incorporates the mag boom and RPWS flexible modes (as well as any other significant flex modes) into a central flexible basebody.

For TVC design, the X-axis dynamics can be idealized by the planar system depicted in Fig. 6. Attached to the (shaded) basebody are the spring-restrained magnetometer boom and two pendulums for the sloshing bipropellant (The RPWS antennas have been truncated on account of their lightness. The RCS monopropellant is discussed below.) The mag boom bending frequency is almost a decade above the propellant slosh frequencies, insuring that its dynamics can be considered independently. Locking up the propellant degrees of freedom and linearizing the two-body equations of motion results in the following transfer function from engine gimbal angle, \(\gamma\), to the rotation angle of the basebody, \(\phi\):

\[
0(s) = \frac{\omega^2}{\omega_b^2 + 1} s^2 + 1 \tag{10}
\]

\[
\gamma(s) = \frac{\omega^2}{\omega_b^2 + 1} \frac{1}{(1+\eta)\omega_b^2} \tag{11}
\]

where \(\eta = \frac{R}{1-R}\), with the inertia ratio \(R\) defined as:

\[
R = \frac{I^0 + \mu(rb+cb)^2}{I^0 + \mu rb^2} \tag{11}
\]

Here \(I^0\) is the inertia of the boom about its own mass center, \(cb\) is the \(Y\) separation of the boom hinge from the basebody center of mass, \(rb\) is distance from the boom hinge to the boom center of mass, and the reduced mass \(\mu = \frac{m_b}{m_0 + m_b}\), where \(m_0\) is the basebody mass and \(m_b\) is the boom mass. Finally, \(\omega_b\) is
natural frequency of the appendage with the basebody fixed:

\[ \omega_n^2 = \frac{k_b}{l_b + m_br_b^2} \]  

Note that the pole of the transfer function (10) is above the zero (K is small and positive), indicating stable interaction for an “infinite bandwidth” PD (no actuator dynamics). To match finite element data, \((1 + \eta)\omega_b\) was set to 0.7 Hz. Viscous damping of 0.25% was added, which is thought to be rather conservative.

The dynamics of the bi-propellant pendulums is considered next. Looking up the the mag boom, the linearized three-body equations yield the following transfer function:

\[ \begin{align*}
\theta(s) &= F_r \frac{\begin{bmatrix} s^2 & \pm 1 & \pm 1 \\ \pm 1 & s^2 & \pm 1 \\ \pm 1 & \pm 1 & s^2 \end{bmatrix}}{s^2} \\
\chi(s) &= \frac{s}{l_s^2} \begin{bmatrix} \frac{s^2}{p_1^2} & + 1 \\ \pm 1 & \pm \frac{s^2}{p_2^2} \end{bmatrix}
\end{align*} \]

The zeroes \(z_1\) and \(z_2\), are given by:

\[ \begin{align*}
z_1^2 &= \frac{m_0}{m_1} (1 - b_1 r_0) + \frac{m_2}{m_0} (1 - b_2 r_0) \\
z_2^2 &= \frac{1}{\frac{m_0}{m_1} + \frac{m_2}{m_0}} \\
\omega_p^2 &= \frac{F_r}{m_{r_p}} \end{align*} \]

where \(m_0\) is the basebody mass, \(r_0\) is the distance from the engine gimbal to the basebody center of mass, \(m_1\) and \(m_2\) are the pendulum masses, and \(b_1\) and \(b_2\) are the distances from the basebody center of mass to the pendulum hinges. The normalizing frequency \(\omega_p\) is the usual pendulum frequency under the main engine acceleration:

where \(l_0\) is the inertia of the basebody. For Cassini mass properties and geometry the ordering of the singularities is as follows: \(z_1 < \rho_1 < z_2 < \rho_2\). As was the case with the spring-restrained appendage, this pattern implies stable interaction with the controller. This is remarkable since the top tank taken in isolation would be unstably interacting due to its “forward” location. (This type of stabilizing phenomenon is multi-tank situations, and also the approximation leading to (17) and (18) were discussed by Greensite.)

In terms of disturbances to the basebody, the worst-case propellant fill is in the 50-60% range, where the pendulum masses approach 300 kg (MMH) and 500 kg (NTO). Generating the pendulum models and evaluating (13)-(18), it is found that the poles and zeroes are clustered around 0.1 Hz. Although extremely low damping levels have been observed in “clean” propellant tanks, it is thought that the presence of the PMD will enhance the damping level. Viscous damping of 1% was added to the pendulum models.

Although the RCS tank holds a significant amount of hydrazine, the design includes an elastomeric diaphragm for bubble-free expulsion in zero-g. It is expected that the presence of this diaphragm will result in greatly reduced motion, with frequencies and damping levels well above those that would be predicted by a clean-tank pendulum model. As a placeholder, such a pendulum was included in the simulation model. Its frequency is just above the bi-propellant slosh modes, although it is much smaller, and it is stably-interacting.

Finally, inertial effects of the articulation of the main engine need to be considered. For a rigid spacecraft, this is known to introduce a pair of imaginary zeroes at the frequency where inertial forces of the articulating engine exactly cancel the thrust vector moment (usually referred to as the “tail-wags-dog” zero). The frequency is given by:

\[ \omega_d^2 = \frac{F_r}{l_c + m_{e OTG}} \]

where \(l_c\) is the moment of inertia of the engine about the gimbal axis, \(m_e\) is the engine mass, and \(r_e\) is the distance from the engine center of mass to the gimbal point. The 5 cm engine balancing requirement keeps the TWD zero above 4.4 Hz for current Cassini mass properties.

Control Design and Analysis

Fig. 7 shows the complete linearized X-axis model which was used for TVC design. As depicted the output of the TVC block is a gimbal angle command, which is delayed, held, and passed through the EGA first-order lag to result in the engine gimbal angle. Obviously this is a gross simplification of the data.
flow between TVC and the E/G manager previously depicted in Fig. 2. Here the gimbal angle is relative to the initial thrust vector orientation, and drives the spacecraft dynamics model after being augmented by the "pre-aim" error \( \theta_p \), which is the angle between the initial engine thrust vector and the vector from the engine gimbal through the system center of mass. (Due to mass property uncertainties and various misalignments, this can be as large as 1° for the first burn.) The spacecraft dynamics model is a cascade of the transfer functions derived above, evaluated at current mass property estimates for a 50% bipropellant fill. Included are the lightly damped mag boom bending mode, the coupled bipropellant slosh modes, and the rigid body mode which has been combined with the TWD zero. The output of the double-integrator is the basebody angle, which feeds the 3rd-order gyro model. At this point the sensed angle is sampled and sent to the controller, along with the back-differenced rate estimate. This is a simplification of the process which occurs in the attitude estimator, the attitude command, and the error decomposition block of Fig. 2, which is made possible by the single-axis reduction and the restriction to small errors. Note that it has been assumed here that the commanded position and rate are both zero.

There is a clear desire to maintain the controller bandwidth as large as possible, not only to minimize the maneuver errors, but also to keep spacecraft rates to an acceptable level during the transient that follows ignition. (This is a problem particularly for short burns, where even a moderate rate following engine cut-off may be difficult for the low-authority RCS thrusters to cope with.) Sensors and actuators are flat out to a few Hz, and the Nyquist frequency is at 4 Hz, suggesting that the system is equipped to operate in the 0.5 Hz range without difficulty. This is complicated, however, by the presence of the lightly-damped magnetometer boom bending mode near 0.7 Hz. As mentioned above, the zero-pole type mode is benign in the sense that it interacts stably with an infinite-bandwidth PID controller. But for the system at hand, the phase situation starts to degrade rapidly around 1 Hz, with significant losses resulting from sensor/actuator dynamics, the need to drive rate by back-differencing position, and the proximity of the Nyquist. This is exacerbated by the computational delay of 125 ms (an entire sampling period), which at 1 Hz already contributes 45° of phase lag. The desire to accommodate a one-sided requirement on mag boom frequency \( (\omega_b > 0.7 \text{ Hz}) \) prompted a gain-stabilization approach for the preliminary design, with a bandwidth of 0.07 Hz. If the boom frequency comes in high, or if damping estimates improve, it would become feasible to design in the 0.1-0.2 Hz vicinity; however, simulation results presented below indicate that even the slower design meets the performance requirements.

As is evident from Fig. 7, the pre-aim error \( \theta_p \) comes in as a constant disturbance torque, and the system response to this dc. serves some attention. The physics of the problem dictates that the steady-state situation has the thrust vector pointing through the system center of mass, i.e. \( \gamma_{ss} = -\theta_p \). The necessary basebody pointing error to drive this offset would be \( \theta_{ss} = \theta_p / k \), where \( k \) is the position-to-gimbal gain, and the resulting thrust vector would be rotated by \( (1 + 1/k)\theta_p \) with respect to its initial orientation. This error is unacceptable, and is remedied by positive feedback of the commanded gimbal angle with a gain of \( 1+k \), as shown in Fig. 8, which makes the steady-state gain from \( \theta_c \) to \( y \) equal 10-1. Lag is required in the feedback loop to abate its destabilizing effects on the main loop. This design was adapted from the analog Mariner 9 design, where it was termed "path-guidance," it being viewed as a rather degenerate form of the more general "guidance loop" associated with missile autopilots. Some block-diagram algebra reveals that this compensation can be rewritten as a combination of outer-loop gimbal feedback with

\[
\begin{align*}
11188s^2 + 3102 \quad & \text{gyro} \\
0.052s^2 + 0.002s + 1 & \text{mag boom} \\
0.045s^3 + 0.002s^2 + 1 & \\
2.74s^2 + 0.033s + 1 & \\
& \text{rigid body / TWD}
\end{align*}
\]

**Figure 7**

TVC X-axis Block Diagram
integral control in the forward path, similar to the design recently outlined by Wic.\textsuperscript{10}

The complete controller with a lead compensation time constant of 6 seconds and a guidance time constant of 10 seconds is shown in Fig. 8. The forward gain (basebody rotation to gimbal angle) is 0.7. The roll-off filter is second order, with poles near 0.3 Hz. The symbol $Z(s)$ denotes a discrete equivalent via bilinear transform with pre-warping.

![Figure 8](image)

**Figure 8**

**TVC Controller**

The open-loop gain-phase plot of the system is shown in Fig. 9. The lower and upper gain margins are -10 dB and +7 dB. The zero-dll crossing is just above 0.07 Hz with 30° of phase margin. The bimpropellant pendulums are phase-stable with frequency-variation margins better than a factor of two in either direction. The mag boom bending mode is gain-stabilized by 9 dB at the assumed lower frequency and damping bounds. The system step response is shown in Fig. 10. The overshot is just a few percent, but there is a moderate undershoot.

![Figure 9](image)

**Figure 9**

**Open-Loop Gain-Phase Plot**

The Y-axis dynamics are quite similar to the X-axis dynamics, except for the lack of the mag boom bending mode. (The torsional mode is insignificant.) The TVC\textsubscript{2} block of Fig. 2 was designed by re-scaling the gains from TVC\textsubscript{1} for the Y-axis inertia, although it can be argued that a higher bandwidth design may be possible for this axis. The roll control logic which drives the RCS thrusters was a simple bang-bang scheme, executing at the 125 ms computational cycle. The rate-to-position gain was 2, and a relatively wide deadzone of ±1° was used to prevent thruster response to transient disturbances coupling in from the TVC axes, which have little secular content.

**Simulation and Performance**

The three-axis algorithm performance was evaluated in a simulation environment built around the 12-body spacecraft dynamics model. Included were the full nonlinear models of the gyros, F\textsubscript{2}GAs, and thrusters, supported by preliminary versions of the "hardware manager" software blocks.

Fig. 11 shows the response to a pre-aim error of 10 arc per axis. The initial attitude error was just 1 mrad per axis, which is a typical deadband for the RCS attitude controller which orients the spacecraft for the maneuver. The bipropellant pendulums were initialized each 30° from the Z-axis, an upper bound which was derived from the requirement that the PMD control the propellant center of mass to 5 cm by the end of a short "quiescent" period immediately prior to main engine ignition.

The first plot of Fig. 11 shows the history of the X-axis orientation angle, which was initialized at 6.2° to align the pre-aimed main engine with the Av target at the reference inertial coordinates [0 0 -1]. The large disturbance from the sloshing bipropellant is
superimposed on the slow guidance loop response, which slew the spacecraft attitude 1° to compensate for the pre-aim error. The second plot shows the spacecraft rate, which peaks out at 0.5 O/s. The mag boom response to the translational acceleration from the main engine is also apparent in this plot. The third plot shows the engine deflection angle, referenced to the pre-aim orientation. The 1° shift points the engine, thrust vector through the equilibrium system mass center. The maximum deflection is 3° from the initial orientation.

The maneuver Av pointing error is considered 10 be the sum of 2 distinct contributions. The first of these, referred to as proportional pointing error, is the steady-state misalignment of the thrust vector from the Av target. It is required to be less than 4 mrad. (This is actually a sub-allocation from a maneuver error budget which contains various other contributions, including structural misalignments and attitude determination errors.) The second contribution is referred to as the fixed pointing error, and is defined as the lateral ΔV component which results from the transient response due to the pre-aim error. It is required to be less than 100 mm/s. (This is for the uncalibrated case.) These errors are assumed to be independent, and are RSSed to form a single requirement on the total Av pointing error, φ, which is defined as the angle between the actual Av vector and the Av target vector:

\[ \phi < \sqrt{(0.004)^2 + (0.001)^2} \]
The last plot of Fig. 11 shows the Av pointing error as a function of maneuver termination time, with upper and lower bounds representing the requirement (20). After the first main engine burn, the pm-aim error can be corrected by analyzing telemetered EGA data. It is clear from Fig. 11 that a burn of at least a minute or so (around 6 m/s) is required, in order to average the data over several cycles of the propellant slosh disturbance. With the recalculated pre-aim, the next main engine burn should have a fixed pointing error well below the calibrated requirement, which is 50 mm/s.

A final note concerning the post-maneuver situation is in order. It is evident from the rate plot in Fig. 11 that short main engine burns can terminate with residual rates as high as 0.5 O/s. The 0.65 N thrusters of the RCS have to fire continuously for 30 seconds to kill this rate, with the spacecraft drifting some 15° from the maneuver orientation. This may aggravate time-line problems for early TCMS, where the time spent near the maneuver attitude is restricted to minimize the exposure time of certain surfaces to the sun.

Future Work

Further work will focus on controller refinement and optimization as better data become available for the spacecraft structural dynamics, propellant models, and sensor/actuator characteristics. An additional area of interest is the performance during SOI, which has a small rate bias (<1°/min) to minimize gravity loss.

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References