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ORBIT MAINTENANCE FOR SMALL SATELLITES USING ELECTRIC PROPULSION

Hassan S.D.*, Argoun M.B., Elbayoumi G.***, Omer O.M.******

ABSTRACT

Controlling a small satellite position using low thrust electric propulsion requires continues control. Continues control means that the orbit control system will command the thrusters every small period of time so that the error will not accumulate. This is due to the fact that low thrust level thrusters are unable to correct an accumulated – relatively – big error. In this paper a control algorithm is developed for controlling a small satellite with Field Emission Electric Propulsion (FEEP) thruster. This algorithm takes into account the errors induced in the GPS receiver and an available thrust level of 1mN. in the satellite velocity direction only. Controller gains are chosen so as to optimize the system equivalent noise bandwidth. This approach is based on estimating the states based on the measurements and controlling the satellite position using the estimated states.

This algorithm uses the modulation capability of the thruster to produce a proportional thrust for disturbances compensation.

A state estimation technique is used to estimate satellite position from GPS receiver. Batched least square estimator is used. The estimated states are used with a PD discrete controller to correct the satellite position. The results pointed out the potential of using state estimation and showed that the satellite position could be kept in orbit with an error in altitude less than one meter using a single 1mN FEEP thruster.

Key words

Satellite, Orbit control, Low thrust level, Orbit maintenance, Orbit parameters estimation, Equivalent noise bandwidth.

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* Professor, Dpt. Of Aeronautical Engineering, Cairo University, Giza, Egypt.
** Head of Space division, National Authority for Remote Sensing and Space Science.
*** Assistant Professor, Dpt. Of Aeronautical Engineering, Cairo University, Giza, Egypt.
**** Researcher Assistant, Space division, National Authority for Remote Sensing and Space Science.
e-mail: ossama_omer@hotmail.com

1 INTRODUCTION

To keep the satellite orbiting in the required orbit suitable for its mission, a thrust force is required for compensating the disturbances affecting the satellite motion. These disturbances for small low earth orbit satellites are in general the aerodynamic forces and solar radiation. These external disturbances cause the satellite to decay. The role of the thrust is to compensate for these disturbance forces. This function is called *Orbit Maintenance*. Small satellites usually do not have propulsion system because of weight considerations and complexities added to the system. Recently, a new technology was developed using electric propulsion, which resulted in small weight propulsion systems. Electric propulsion which is less in weight and easier to mount on the satellite bus is then easier to use it in small satellites for orbit maintenance. Of these, FEEP, Field Emission Electric Propulsion, is used to control the satellite position in this development. FEEP thrust level is less than 1mN and can operate in a modulated form.

To perform the function of orbit maintenance, a suitable control system is designed. This algorithm is applied to the orbital plane motion only, i.e., we do not correct for the inclination errors. This is because inclination errors are usually small and require very long maneuvers if we try to correct using low thrust level thrusters, and in that case the only way to perform such maneuver is to open thrusters at full thrust. The algorithm, developed to achieve orbit maintenance, tries to correct the difference between the satellite position and a target position at which the satellite is required to be. So, to generate the error signal we build two orbit propagators; the first propagates the actual satellite position and the second propagates the ideal satellite position. This algorithm is referred to as "Target Position Control of Orbital Plane Motion". In this method of control, modulation capabilities of the thrusters are required to produce thrust at any level lower than the maximum thrust available.

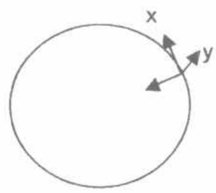
The results of this algorithm depend mainly on the measurements errors and the method by which these errors are filtered.

To get rid of measurements errors effects, state estimation technique was used. A batched least square estimator was used since it is simple and also suitable for the slow dynamic orbit control problem. The state vector is chosen to be the system state vector which is a four elements vector containing the satellite in plane position and velocity. This vector is also the measurements vector, since GPS output is the satellite position and velocity. However, the observation vector is chosen to be the thrust vector containing the thrust values at the selected points. This is because thrust is a function in the four measurements and so it is a representative single observation. Design of the estimator showed a convergent solution. Simulations resulted in a very good system performance. The error in satellite altitude will be less than 1m with a required thrust level less than 1mN.

To build a controller for the satellite orbit, we need first to design the controller in order to get the controller gains. Then we simulate the resultant controller together with the satellite dynamics and the external disturbances to test and modify the controller gains. In the design step, linearized equations of motion will be used. Orbit control problem will be divided into two separate problems: Orbital plane motion control and Normal to orbit plane errors control. Only the first problem is considered here.

2 SYSTEM MODEL:

The control method discussed here is characterized by controlling only the satellite position in two directions: the velocity direction (x direction) and the earth pointing direction (z direction). The linearized equations of motion derived by Arthur E. Bryson, JR. [1] are used. The final form is:



$$\dot{\bar{x}} = F\bar{x} + Gu$$

$$\bar{x} = \begin{bmatrix} \delta u \\ \delta w \\ \delta x \\ \delta z \end{bmatrix} \quad ; F = \begin{bmatrix} 0 & n & -n^2 & 0 \\ -n & 0 & 0 & 2n^2 \\ 1 & 0 & 0 & n \\ 0 & 1 & -n & 0 \end{bmatrix} \quad ; G = \begin{bmatrix} 1/\mathrm{m} & 0 \\ 0 & 1/\mathrm{m} \\ 0 & 0 \\ 0 & 0 \end{bmatrix}$$

$$u = \begin{Bmatrix} T_{dx} \\ T_{dz} \end{Bmatrix}$$

where, δu, δw are the components of the velocity deviation vector in the orbital axes.

δx is the position error in x direction of the orbital axes, δz is the position error in z direction of the orbital axes, n Orbital angular velocity, m Satellite mass, T_{dx} ,T_{dz} are the thrust forces in x and z directions respectively.

The above equation can be rewritten in a normalized form by using the following units: time in *1/n*, *(δu, δw)* in *nR*, *(δx, δz)* in *R*, *(T_x, T_z)* in *mg*. The new matrices are:

$$F = \begin{bmatrix} 0 & 1 & -1 & 0 \\ -1 & 0 & 0 & 2 \\ 1 & 0 & 0 & 1 \\ 0 & 1 & -1 & 0 \end{bmatrix} \qquad G = \begin{bmatrix} 1 & 0 \\ 0 & 1 \\ 0 & 0 \\ 0 & 0 \end{bmatrix}$$

The transfer functions are:

$$\frac{\delta z}{T_{dx}} = \frac{-2}{s(s^2+1)} \qquad \frac{\delta x}{T_{dx}} = \frac{s^2-3}{s^2(s^2+1)}$$

3 FULL STATE FEEDBACK CONTROLLER:

It can be proved that feedback with either δz or δx does not provide stability. To stabilize the satellite motion full state feedback is used. $T_x = -Kr\,x$.

Linear quadratic regulator (LQR) design method is used to get the feedback gains with the following performance of index:

$$J = \int_0^\infty \bar{x}^T A \bar{x}\, dt + \bar{T} B \bar{T} \qquad (1)$$

where ,

x is the estimated state vector, T is the control thrust vector, only T_{cx} in this case.

A, B are weighting matrices.

The above performance of index will optimize the thrust value and also the error in the forward displacement δx and error in altitude δz.

The control system used to maintain the satellite position is shown in Fig.2.

4 REDUCING THE EFFECT OF MEASUREMENTS ERRORS

Measurements errors have an effect on the controller action. Reducing this effect is the objective of the development in this section. This reduction is accomplished by lowering the system Equivalent Noise Bandwidth (ENB). This can be performed only by changing the controller gains. It is possible to calculate the controller gains that minimize the system ENB and hence minimize the effect of measurements errors on the system. However, the required control thrust will not be practically accepted in terms of that the required thrust will be in two directions; this means two thrusters are required to be mounted on the satellite and hence critical power consumption. So, the objective is to select the controller gains so as to lower the system ENB with an accepted thrust profile. This can be done by modeling a thrust ENB and check its value. Thrust ENB can be minimized by changing the controller gains. This will be on the cost of system performance. A compromise between thrust ENB and system ENB is done to select the appropriate controller gains.

4.1 Review:

In orbit determination using GPS, the receiver receives a signal that determines the position of the satellite. This signal is affected by a superimposed error. Hence, we receive a signal with certain rate and each time we receive a signal there is a random error value added to the signal. So, we can model this as a white noise added to the received signal. This white noise can be considered as an input to the system.

Recall the system block diagram:

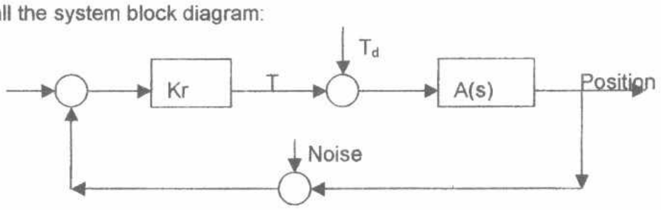


Figure 1 Typical system block diagram

Where, A(s) is the system transfer function. Force to position, Kr are the controller gains, T is the calculated control thrust, T_d is the disturbances on the system.

If we assume that the spectrum of this white noise can be defined as:

ϕ_a = Ka^2 m^2/rad/s., where Ka is constant.

If the system transfer function is: $F(s) = \frac{b(s)}{a(s)}$

Then, the mean square output is:

$$\sigma^2 = k_a^2 \int_0^\infty \frac{|b(jw)|^2}{|a(jw)|^2} dw$$

The above Integral is the Equivalent noise bandwidth, *ENB*, which we aim to reduce the noise effect.

4.2 Minimizing ENB:

The only way to change the value of the integral representing the Equivalent noise bandwidth, in this problem, is to change the controller gains. We can change the gains by changing the weighting matrices used to calculate the gains in LQR technique.

A point that must be mentioned is that noise has effect on both the system performance and the required control thrust. Both of these variables must be checked, so the *ENB* for the system performance will be studied and also the *ENB* for the control thrust. Let δZ, altitude error, be considered as a measure for the system performance.

Now, gains will be changed and for each set of gains, the *ENB* for δZ and the *ENB* for thrust will be calculated. Results are plotted in Fig. 5.

From Fig. 5, we can see that by lowering the gains we lower the thrust and the *ENB* of the thrust while the *ENB* of δZ increases. By increasing the gains, the *ENB* of thrust increases while *ENB* of δZ decreases. So, we have to choose a point which produces an acceptable δZ with a thrust that is feasible. A feasible thrust is a thrust that is only in the positive direction with a value that is lower than 1mN.

Several simulations are performed, and resulted in that we cannot achieve a positive only thrust. So, we will try to minimize the negative portion of the thrust, and force this negative portion to zero then we test the new system via simulations. Fig.6 and Fig.7 show simulation results for the following cases:

Noise free: system performance where there is no noise added to the GPS output.

Case 1: Kr=[3.325 – 3.2653 2.2653 – 5.17] *ENB* for δZ= 1.552 w_{orb}

Case 2: Kr=[3.39 – 3.39 2.346 – 5.36] *ENB* for δZ= 1.472 w_{orb}

Case 3: Kr= [3.46 – 3.56 2.44 – 5.59] ; *ENB* for δZ= 1.397 w_{orb}

Case 4: Kr= [4.98 – 7.78 4.616 – 11.9] ; *ENB* = 0.5592 w_{orb}

Lowering the *ENB* of the δZ improved the system performance on the cost of a worse thrust profile as shown in Fig.6 and Fig.7.

Fig.8 shows the effect of lowering the *ENB* of thrust on system performance and the required control thrust.

Case 1: E.N.B. of thrust = 44.2 worb

Case 2: Kr= [2.121 – 1.283 0.966 – 2.399] E.N.B. of thrust = 12.551 worb.

Case 3: Kr= [1.204 – 0.4126 0.3126 – 1.135] E.N.B of thrust= 3.0885 worb.

Case 4: Kr= [0.5974 – 0.105 0.0734 – 0.5404] E.N.B = 0.6948 worb.

A degradation in the system performance occures by lowering the *ENB* of the thrust. However; the thrust is improved, as expected, as shown in Fig.9.

Fig.10 and Fig.11 show the system performance after we force the negative portion of the thrust to be zero.

Case 8: Kr= [0.0821 – 0.0022 0.0012 – 0.0799] ; E.N.B. of thrust = 0.0152 worb.

Case 9: Kr= [0.0249 – 0.0002 0.001 – 0.0247] ; E.N.B. of thrust = 0.00133 worb.

Case 10: Kr= [0.0139 – 0.0001 0 – 0.0138] ; E.N.B. of thrust = 0.00022 worb.

Case 11: Kr=[0.0078,0,0,-0.0078] ; E.N.B. of thrust = 9.438 e-s worb.

From the above results, It can be seen that controller gains are changed so as to reduce the system and thrust *ENB*. From the above plots, case 9 is the best case. We conclude that by lowering the *ENB* of the thrust we can achieve good performance with a thrust only in the positive direction. The maximum error is about 1 meter in two orbits. In a longer simulation, the maximum error is about 5 meters per day. This is a well-accepted performance. Thrust value is almost under 0.2 mN, which is suitable.

5 BATCH LEAST SQUARE ESTIMATOR (BLSE):

First we need to determine the state vector, observation vector, and observation model vector. The state vector is the vector containing the states of the system. The observation vector contains the observations; an observation is any quantity calculated based on measurements. Observation model vector is a vector generated by a model for the observations. The basic idea is that we wish to minimize the difference between the observations, i.e. measurements, and the observation model vector [11].

consider the state vector to be:

$$x = \begin{bmatrix} \delta u & \delta w & \delta x & \delta z \end{bmatrix}^T \qquad (2)$$

The state vector x is a function in time and x°; where x° is the state vector at the start time of the measurements period; called reference time t_o. The batch least square estimator estimates this state vector x°; this estimate is denoted by x°.

In the process we collect measurements over a certain period of time. For the moment Let us collect measurements on a period of 100 seconds. We have a measurement every one second. So we have 100 set of measurements. For each measurements set, the observations associated with these measurements will be calculated. The measurements are the states of the system: δu, δw, δx and δz.

Since the thrust is calculated directly from the measurements, thrust, T, may be considered as the only observation in this model. Let the observations vector be:

$$y = \begin{bmatrix} y_1 & y_2 & \ldots\ldots & y_n \end{bmatrix}^T \tag{3}$$

where, y_i is the thrust value T_i; $y_i = -[k]\, x$.

[k] is the controller gains vector.

The observation model vector ,z, is an n-dimensional vector composed of predicted values of the observational vector based on estimated values of the state vector elements.

$$z = g(x(t)) = -[k]x(t) \tag{4}$$

To determine the state vector x, we assume that y equals the observation model vector , g(x(t)), based on the mathematical model of the observations plus additive random noise v. Thus for each element of y,

$$y_i = g_i(x(t_i)) + v \tag{5}$$

Now, we wish to minimize the loss function:

$$J = \frac{1}{2}\rho^T W \rho + \frac{1}{2}\left[x^o - \bar{x}_A^o\right]^T S_o \left[x^o - \bar{x}_A^o\right] \tag{6}$$

where, S_o is the state weight matrix. If the elements of S_o are zero, no weight is assigned to the a priori estimate.

ρ is the observation residual vector, defined by: $\rho = y - g$.

W is a symmetric, nonnegative definite matrix chosen to weight the relative combination of each observation, according to its expected accuracy or importance. In the simplest case *W* is the identity matrix indicating that equal weight is given to all observations.

For *J* to be a minimum with respect to x°, ∂J/∂x° must be zero. Therefore the value of x° which minimizes *J* is a root of the equation:

$$\frac{\partial J}{\partial x^o} = -\rho^T W G + [x^o - \bar{x}_A^o]^T S_o = 0^T \tag{7}$$

$$G = \frac{\partial g}{\partial x^o} = \begin{bmatrix} \frac{\partial g_1}{\partial x_1^o} & \frac{\partial g_1}{\partial x_2^o} & \frac{\partial g_1}{\partial x_3^o} & \frac{\partial g_1}{\partial x_4^o} \\ \frac{\partial g_2}{\partial x_1^o} & \frac{\partial g_2}{\partial x_2^o} & \frac{\partial g_2}{\partial x_3^o} & \frac{\partial g_2}{\partial x_4^o} \\ \ldots & \ldots & \ldots & \ldots \\ \frac{\partial g_{100}}{\partial x_1^o} & \frac{\partial g_{100}}{\partial x_2^o} & \frac{\partial g_{100}}{\partial x_3^o} & \frac{\partial g_{100}}{\partial x_4^o} \end{bmatrix} \tag{8}$$

Values for $\partial g_i / \partial x^o$ can be calculated from:

$$\frac{\partial g_i}{\partial x^o} = \frac{\partial g_i}{\partial x}(t_i) \cdot \frac{\partial x}{\partial x^o}(t_i) \qquad (9)$$

Values for $\partial g_i / \partial x$ can be computed analytically from the observation model:

$$g_i = -[k]x_i = \begin{bmatrix} -k_1 & -k_2 & -k_3 & -k_4 \end{bmatrix} x_i$$

$$\frac{\partial g_i}{\partial x_i} = \begin{bmatrix} -k_1 & -k_2 & -k_3 & -k_4 \end{bmatrix} \qquad (10)$$

In order to calculate $\partial x_i / \partial x^o$, recall the linear model of the system:

$$\frac{d}{dt} x = Ax + Bu \qquad (11)$$

$$A = \begin{bmatrix} 0 & n & -n^2 & 0 \\ -n & 0 & 0 & 2n^2 \\ 1 & 0 & 0 & n \\ 0 & 1 & -n & 0 \end{bmatrix} \qquad B = \begin{bmatrix} 1/m & 0 \\ 0 & 1/m \\ 0 & 0 \\ 0 & 0 \end{bmatrix} \qquad (12)$$

where, n is the orbital angular velocity, m is the satellite mass.

The solution to the above equation is:

$$x(t) = e^{A(t-t_o)} x(t_o) + \int_{t_o}^{t} e^{A(t-\tau)} Bu(\tau) d\tau = e^{A(t-t_o)} x^o + \int_{t_o}^{t} e^{A(t-\tau)} Bu(\tau) d\tau \qquad (13)$$

Form the above equation, we can generate the matrix G.

Now, For J to be a minimum with respect to x^o, $\partial J / \partial x^o$ must be zero. Therefore the value of x^o which minimizes J is a root of the equation:

$$\therefore \frac{\partial x_i}{\partial x^o} = e^{A(t_i - t_o)} \qquad (14)$$

$$\therefore \frac{\partial g_i}{\partial x^o} = \begin{bmatrix} -k_1 & -k_2 & -k_3 & -k_4 \end{bmatrix} e^{A(t_i - t_o)} \qquad (15)$$

$$\therefore \bar{x}^o = \bar{x}^o_A + S_o^{-1} G^T W \rho \qquad (16)$$

$$\therefore \bar{x}^{o} = \bar{x}^{o}_{A} + S^{-1}_{o} G^{T} W (y - g) \tag{17}$$

The above equation estimates the state vector at the reference time if we know the vector g. But g is a function in x°, So the above equation will be used in an iterative procedure to get an estimate for the state vector at the reference time. We begin with apriori estimate for the state vector x^{o}_{A}, Then we calculate the observation model vector g on the whole period of measurements using this apriori estimate for the initial state vector, g_A. We also generate the matrix G as explained above, then we can calculate the estimated value for the state vector at the reference time.

$$\bar{x}^{o}_{k+1} = \bar{x}^{o}_{k} + S^{-1}_{o} G^{T} W (y - g_{k}) \tag{18}$$

To examine the convergence of the above equation; Let us rewrite the above equation in the form:

$$\begin{aligned}
&\bar{x}^{o}_{k+1} = \bar{x}_{k} + S^{-1}_{o} G^{T} W y - S^{-1}_{o} G^{T} W g_{k} \\
&g_{ki} = -[k] x_{ki} \\
&U = \begin{bmatrix} -[k]_{1} \\ -[k]_{2} \\ \dots\dots \\ -[k]_{100} \end{bmatrix} \\
&\therefore \bar{x}^{o}_{k+1} = \bar{x}_{k} + S^{-1}_{o} G^{T} W y - S^{-1}_{o} G^{T} W U \bar{x}_{k} \\
&\therefore \bar{x}^{o}_{k+1} = (I - S^{-1}_{o} G^{T} W U) \bar{x}_{k} + S^{-1}_{o} G^{T} W y
\end{aligned} \tag{19}$$

So, To examine the convergence, the eigenvalues of the matrix T should be checked:

$$T = (I - S^{-1}_{o} G^{T} W U) \tag{20}$$

To do that, assume that S_o is the identity matrix and W is:

$$W = w \begin{bmatrix} 1 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 \\ \dots & \dots & \dots & \dots \\ 0 & 0 & 0 & 1 \end{bmatrix}$$

The condition of convergence is that all eigenvalues are within the unit circle. This condition is not satisfied for any positive value for w. So this procedure must be modified. this will be done, but for the moment, if we start with a good apriori estimate x_A and use the above equation for a single iteration only we can get good results. Now, using the estimated values for the initial state vector, we propagate this initial condition over the whole period to get the states. Since, the propagation period is only 100 seconds, linear propagator will be used without affecting the accuracy:

$$x(t_i) = e^{A(t_i - t_o)} \bar{x}^o + \int_{t_o}^{t} e^{A(t-\tau)} B u(\tau) d\tau \tag{21}$$

Finally, propagated values are used for the state vector at the end of the measurements period to calculate the required control thrust.

Now, after implementing the above calculations in the simulation environment, a simulation is run for three orbital periods to examine this way in estimating the states.

Let for the moment, the weighting matrices be as follow:

$$S_o = \begin{bmatrix} 1 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 \\ 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 1 \end{bmatrix}$$

$$W = 0.01 \times \begin{bmatrix} 1 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 \\ 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 1 \end{bmatrix} \tag{22}$$

Now we use another way to solve equation(6) since the previous procedure was not convergent. We can linearize g about a reference state vector, and expand each element of g in a Taylor Series of the reference state vector. Let the reference state vector be x_A^o.

$$g_i = g_i(x_A^o) + \frac{\partial g_i}{\partial x^o}(x_A^o)\left[x^o - x_A^o\right] \tag{23}$$

$$\therefore\ g = g_A + G_A x^o - G_A x_A^o \tag{24}$$

Now, substitute in equation(6),

$$[S_o + G_A^T W G_A] x^o = S_o x_A^o + G_A^T W [y - g_A + G_A x_A^o] \tag{25}$$

$$\therefore \bar{x}^o = x_A^o + [S_o + G_A^T W G_A]^{-1} G_A^T W (y - g_A) \tag{26}$$

If the correction from the above equation is not small we can use it in an iterative way:

Let:

$$M = F^{-1}S_o \qquad F = S_o + G_k^T W G_k \qquad N = F^{-1} G_k^T W$$

$$\therefore \bar{x}^o_{k+1} = \bar{x}^o_k + F^{-1} G_k^T W(y - g_k) + F^{-1} S_o \bar{x}^o_A - F^{-1} S_o \bar{x}^o_k \tag{27}$$

$$\therefore \bar{x}^o_{k+1} = \bar{x}^o_k + F^{-1}\left[G_k^T W(y - g_k) + S_o\left(\bar{x}^o_A - \bar{x}^o_k\right)\right] \tag{28}$$

$$\therefore \bar{x}^o_{k+1} = \bar{x}^o_k - M\bar{x}^o_k + M\bar{x}^o_A + N(y - g_k) \tag{29}$$

In order to check the convergence of this approach, we check the eigenvalues for the matrix:

$$T = (I - M - NU)$$

This matrix is convergent for values of *w* from 0.01 to 1.99.

This way of solving equation(6) is implemented. These results show that we could achieve very good performance and thrust level. The thrust level is lowered to be less than 0.6 mN but there is negative control thrust. This means that this estimator does not provide the required accuracy to achieve only positive control thrust. We can improve the estimation either by increasing the number of measurements processed in each time more than 200 or by using a recursive estimator; either a least square recursive estimator or a kalman filter. Since we have already a kalman filter for the attitude estimation, we can use it to estimate also the satellite position. So increasing the size of the state vector to include also the four states of the orbit control system will result in better estimation for the satellite position and it is expected to get a positive only thrust using states estimated from a kalman filter.

Another way to achieve 'only positive' thrust required for control is to lower the gains of the controller. Recalculate the gains for two cases:

case 1: the weighting matrices: K_r = [0.371, -0.224, 0.000184, -0.000457];

case 2: the weighting matrices K_r = [0.31755, -0.164, 0.000136, -0.000365];

Simulations are performed for both cases using a controller that produces a control command every 100 seconds. The results of simulations are plotted in Fig.3.

As expected, the thrust is lowered and the negative portion started to disappear. The degradation in performance is not a matter at all; the error in altitude is less than 0.3m.

Now, we can completely get rid of the negative thrust by saturating any negative thrust to zero without affecting the performance so much. A simulation is performed for 25 orbital periods and a control command every 100 seconds. the results are plotted in Fig 4.

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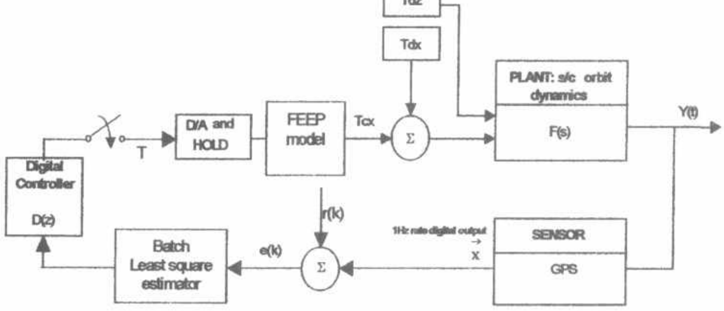

Figure 2 Block diagram for orbit control system.

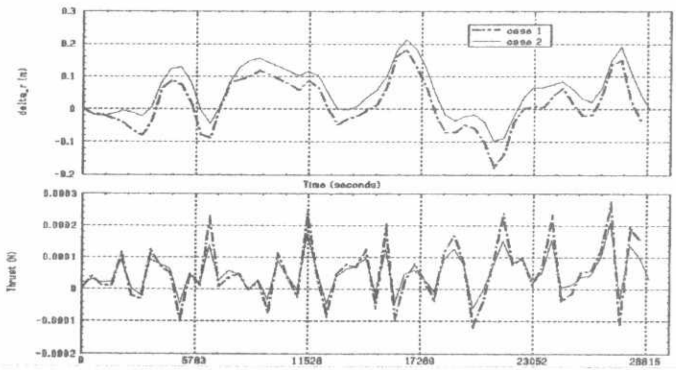

Figure 3 System performance variations vs. gains variations using BLSE.

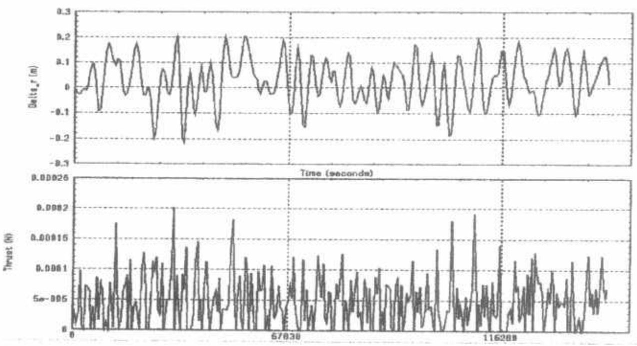

Figure 4System performance using BLSE with only positive thrust.

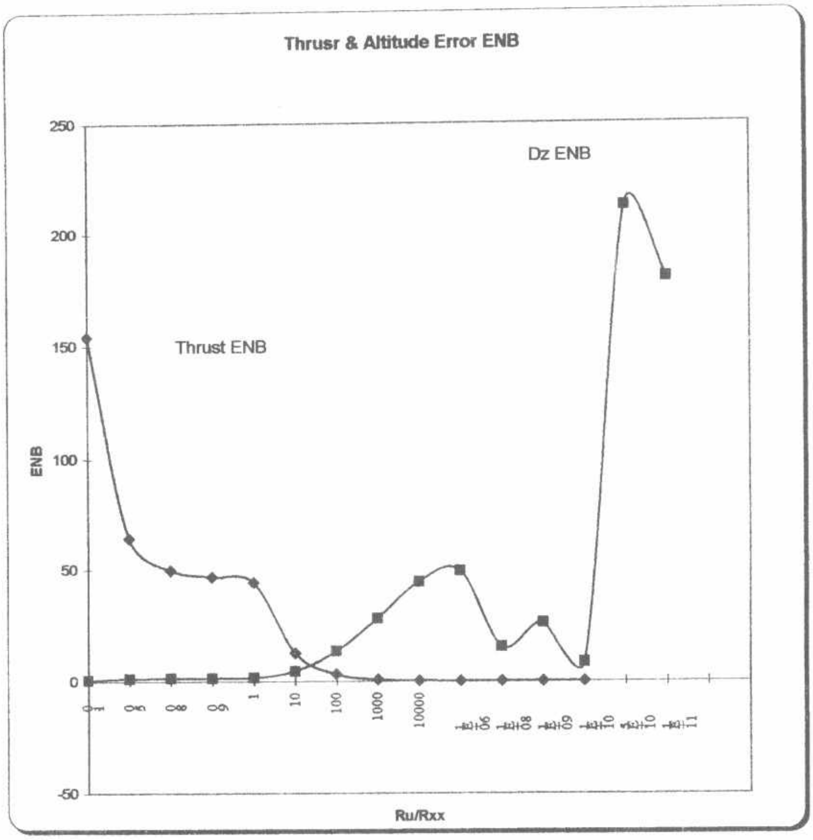

Figure 5 ENB values for Thrust & δZ vs. Weighting matrices

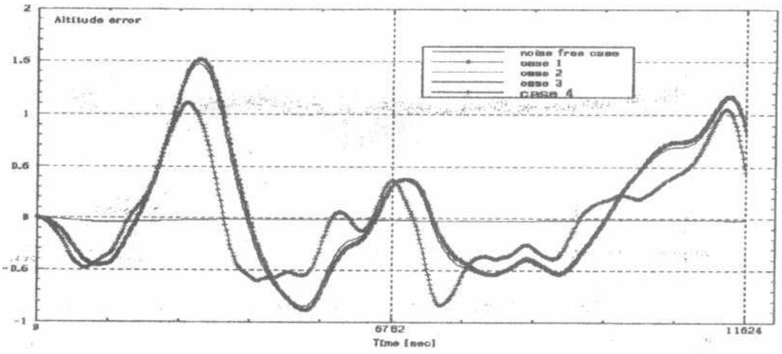

Figure 6 System Performance variation after lowering the ENB of the δZ.

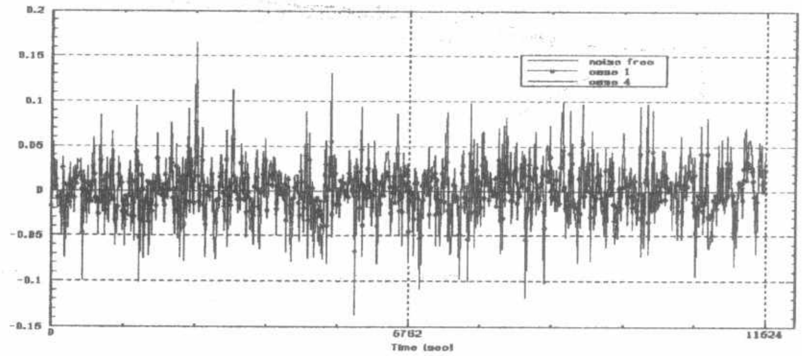

Figure 7 Thrust variation after lowering the ENB of the δZ

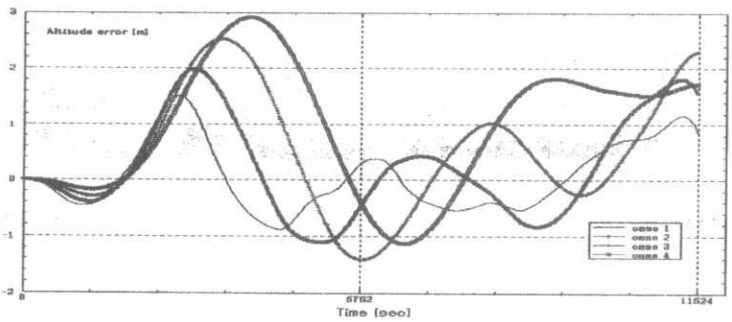

Figure 8 System Performance variation after lowering the ENB of the thrust

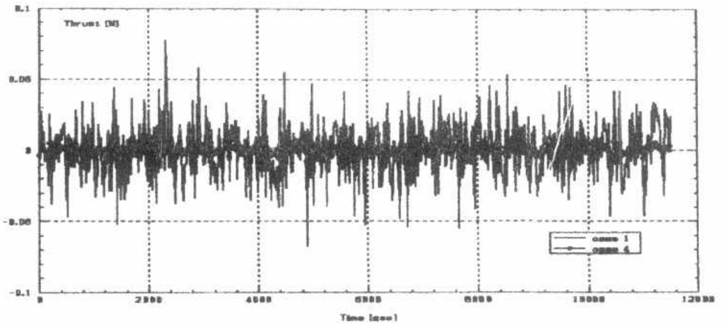

Figure 9 Thrust variation after lowering the ENB of the δZ

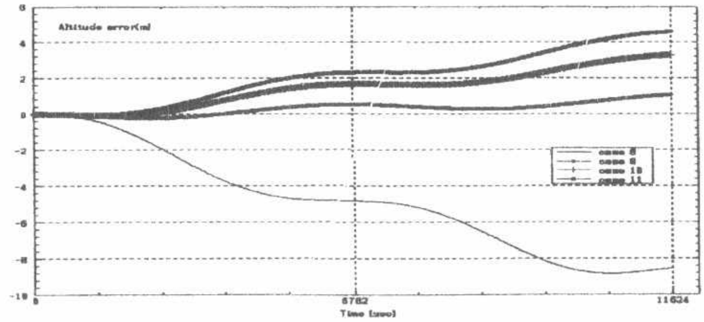

Figure 10 System Performance variation after lowering the ENB of the thrust; Negative thrust is forced to zero.

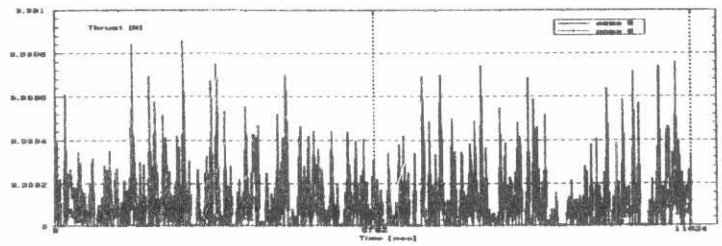

Figure 11 Thrust variation after lowering the ENB of the thrust; Negative thrust is forced to zero.