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On Aerospace Vehicle Simulation as a Designer's Tool

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ABSTRACT

The role of system simulation, encompassing the various features of the missile in system design, is brought out. Specific issues investigated through the simulation model are highlighted with supporting results from flight records, wherever possible. The requirement to consider slosh as 'inclined' rather than 'vertical' in situations where the missile is subjected to large lateral accelerations and orientation changes is brought out. Accentuation of slosh activity in free flight is demonstrated from both simulation results and flight records. The requirement to consider engine inertia carefully in roll loop analysis is highlighted. Results of structural response when flexural mode is excited are presented. The effect of quantisation noise in the control system in low dynamic pressure region has been investigated and compared with flight results. Occurrence of various important events predicted by the simulation programme in a long range mission and the actual vehicle performance are compared. Finally, the role played by the simulation programme in hardware-in-loop simulation is brought out.

NOMENCLATURE

λ	Slosh amplitude along the free surface
ζ_s, ω_s	Slosh damping coefficient and frequency
θ	Vehicle orientation in pitch
θ_m	Orientation of free surface
V_x, V_y, V_z	Inertial velocities in body frame
p, q, r	Inertial body rates in body frame
I_{xx}	Roll moment of inertia
T	Total thrust force
$\delta_p, \delta_{r1}, \delta_{r2}$	Engine deflection in pitch and deflections in yaw of engine 1 and engine 2, respectively
I_e	Engine moment of inertia about hinge point
S_e	First moment of inertia of engine about gimbal point
e_{gp}	Moment arm for roll
θ_e, ψ_e	Errors in pitch and yaw attitudes at start of closed-loop guidance
x_{cg}	Centre of mass measured from gimbal point

1. INTRODUCTION

System simulation model incorporating details of various subsystems like aerodynamics, flight dynamics, propulsion, control and guidance, and allowing for intra-subsystem studies, cause and

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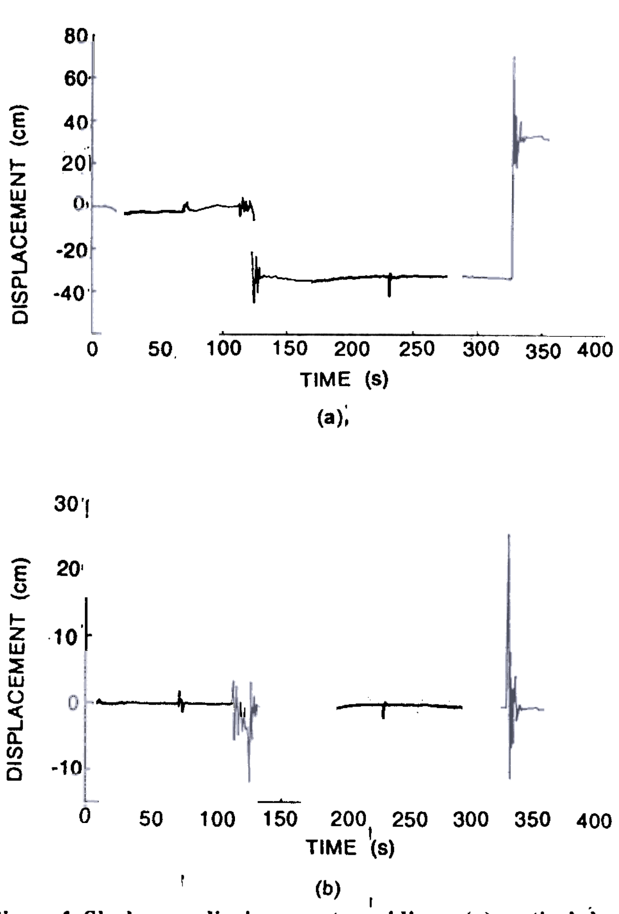


Figure 1. Slosh mass displacement—oxidiser : (a) vertical slosh, (b) inclined slosh.

effect studies on parameter variations and stochastic effects, constitutes one of the most important aspects of any aerospace vehicle system development programme. It has a central role to play in the overall system design, since any subsystem design proceeds interactively with the simulation model through various stages till the evaluation flight tests are completed.

Though the scope and sweep of simulation are quite large, attention is focused in this paper only on certain aspects typical of large systems, where considerations of bending and slosh are important. Results arising from incorporation of features like inclined slosh, vehicle flexibility, quantisation effects in navigation system and engine dynamics in roll channel are discussed in comparison with the results obtained from flight records, wherever possible. The signal role played by the six-degree-of-freedom (6-DOF) simulation model in providing information on wide ranging issues like guidance margin, control margin, flight event sequencing, provision of certain bench mark results for setting up the hybrid simulation test facility, evaluation of different guidance and control strategies and suitability of nominal paths for guidance of missiles are discussed briefly with results drawn from flight records.

2. ANALYSIS OF SLOSH

2.1 Slosh of Liquid Propellants

Slosh of liquid propellants along the free surface determined predominantly by vehicle forward acceleration and latax is considered. This reveals smaller slosh activity compared to the conventional approach applied in launch vehicles where slosh is considered normal to the fore-and-aft axis of the vehicle. The former approach is termed here as 'inclined slosh' and the latter as 'vertical slosh'. Vertical slosh generally results in much larger slosh amplitudes, causing undue concern regarding the validity of the model, since, for large amplitudes, the effects of wave breaking and fluid rotation in the tank would have to be considered. The development of inclined slosh formulation uses Langrangean energy approach[1]. Extensive experimental data have been generated for inclined slosh at different tank orientations, and with and without baffles, to arrive at non-dimensional frequency and damping ratios at these conditions.

Digital simulation for a surface-to-surface vehicle is carried out in the presence of slosh disturbances, with the option of considering either the vertical slosh or the inclined slosh model. In vertical slosh, it is explicitly assumed that slosh activity is always confined to a plane normal to fore-and-aft axis, whatever the orientation of the missile is. In the inclined slosh model, the liquid surface orientation is computed based on the average acceleration profile over an interval of time which is smaller than the characteristic time associated with slosh. The time interval used here

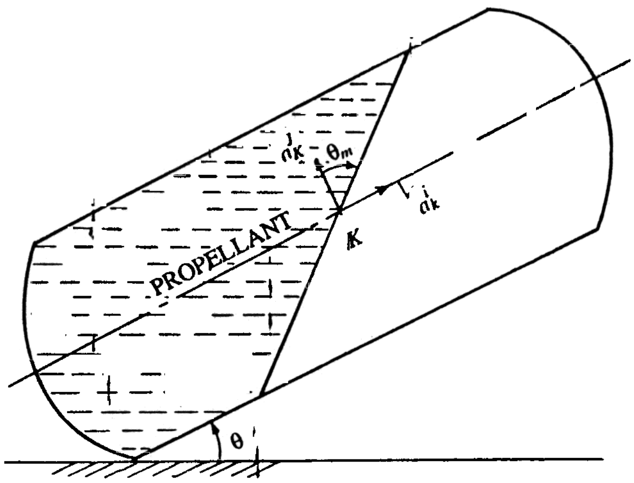


Figure 2. Orientation of the propellant under slosh

is 300 ms. It is assumed that the slosh activity is in that plane during this time period. It can be seen from Fig.1 that the slosh mass displacements are much larger in the vertical slosh model compared to those in the inclined slosh model. Except during the transients, slosh mass displacements are very small in the inclined slosh model.

The slosh activity is associated with the trajectory events. In the case of the particular vehicle for which this study was carried out, around the start of closed-loop guidance, slosh activity begins, followed by the slosh activity due to engine shut off and the consequent shifting of left over fuel and oxidiser. Gravity turn which starts around the mid phase of the flight also excites slosh. Again, in the terminal phase of the flight, the activity is seen when the vehicle starts upward and downward manoeuvres, respectively.

2.2 Slosh Phenomenon

Slosh phenomenon, modelled as a spring mass damper system, indicates that slosh activity increases during the non-thrusting phase of free flight. This can be explained as follows:

The equation for slosh is

$$\ddot{\lambda} + 2\zeta_s\omega_s\dot{\lambda} + \omega_s^2\lambda = \lambda\dot{\theta}^2 - g\cos(\theta - \theta_m) - \cos\theta_m(a_k^j) - \sin\theta_m(a_k^i) \qquad (1)$$

where a_k^i, a_k^j are the components of inertial acceleration of point K at which the slosh mass could be imagined to be attached to the vehicle (Fig. 2).

Writing the acceleration components as the sum of smooth and transient parts

$$a_k^j = a_k^j(s) + a_k^j(t) \qquad (2)$$

and using them in Eqn (1), and noting that for the free surface which is determined by the smooth components of acceleration,

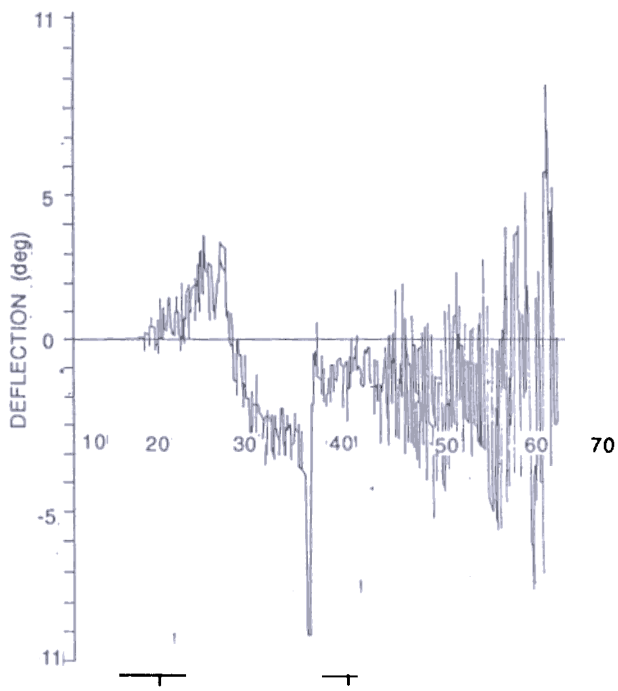
$$a_k^i(s)\sin\theta_m + a_k^j(s)\cos\theta_m + g\cos(\theta - \theta_m) = 0 \qquad (3)$$

The slosh equation, Eqn (1), reduces to

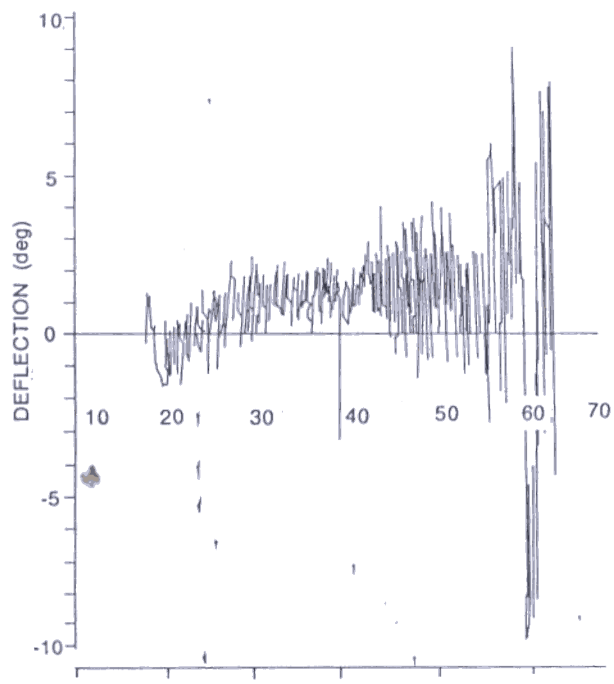
$$\ddot{\lambda} + 2\zeta_s\omega_s\dot{\lambda} + \omega_s^2\lambda = \lambda\dot{\theta}^2 - [\cos\theta_m(a_k^j(t)) + \sin\theta_m(a_k^i(t))] \qquad (4)$$

wherein the component of transient acceleration along the free surface becomes the driving term for slosh. The slosh frequency square, ω_s^2, depends directly on the component of vehicle- sensed acceleration normal to the free surface. In free flight under gravity, under conditions of very little aerodynamic drag, the sensed acceleration would be vanishingly small, and so would be ω_s. The slosh equation, Eqn (4), therefore, represents a spring mass damper, with a negligibly small damping force and spring restraint force, driven by tangential component of transient acceleration. This could be expected to result in large slosh amplitudes.

Simulation studies show such a type of behaviour for non-thrusting cases in free flight. Post-flight records for a particular flight indicate a delay of about 2.1 s in the firing of a set of ullage motors, thereby resulting in a non-thrusting gravity flight over this duration. Analysis of flight records of deflections (Fig. 3) during this period shows increased control activity. This is attributed to increased slosh activity at much reduced slosh



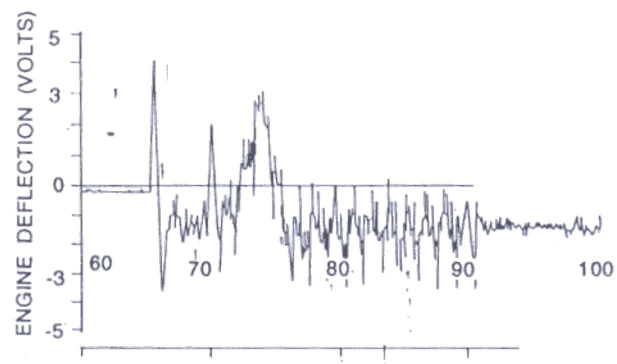
(a)



(b)

Figure 3. Fin tip control deflection: (a) pitch, (b) yaw

frequencies. The control deflections increase to 10° in magnitude during this zone. Subsequently, with the firing of the next set of ullage motors, slosh frequency increases, and the deflection requirement decreases.



(a)

Figure 4(a). Engine deflection (pitch)

2.3 Effect of Baffles on Slosh Activity

Baffles are used in propellant tanks to mitigate the slosh activity. The flight records (Figs 4(a&b)) indicate reduction in the magnitude of control demands at appropriate time. It is because, around this time the propellant level is at a position where the ring baffles become effective in terms of adding to the slosh damping. Therefore, the slosh amplitudes decrease and result in smaller slosh forces and moments. The preflight data for damping for the liquid propellant also show an increase around this time, which again is due to the baffles. Eventually, as the propellant gets consumed, the slosh mass also decreases and the excitation forces and moments due to slosh decrease.

3. FLEXURE

The simulation model has the flexural response of the vehicle also included in it. Equations for rigid body variables, flexure, slosh and engine dynamics are used to describe the motion. The variables get coupled in the equations of motion, which are derived using the Lagrangean formulation. The rigid body equations have as inputs loads which are caused by the rigid body motion, and the incremental loads caused by the local aerodynamic angles due to flexural motion as well. The modal functions are generated by loads which include rigid body loads and those due to flexural motion, thus causing coupling between

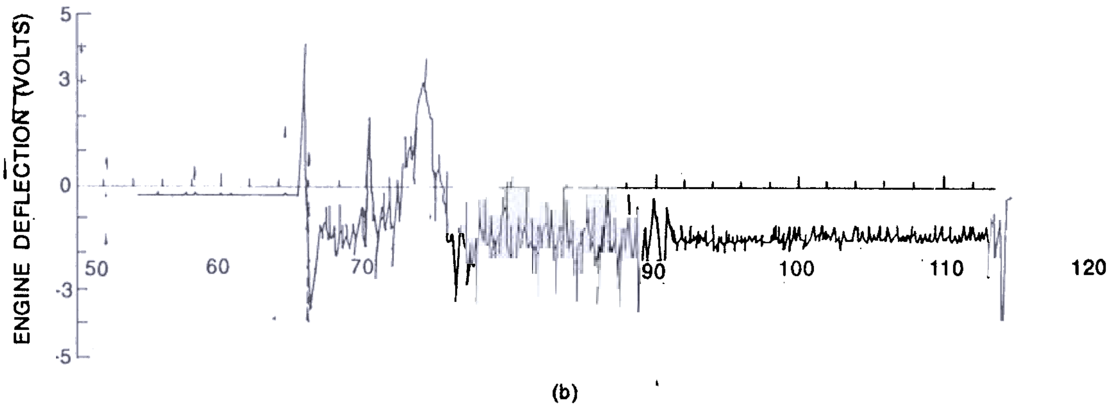
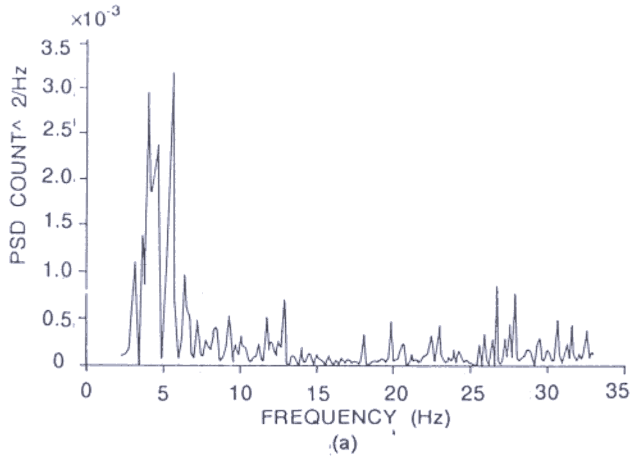


Figure 4(b). Engine deflection (yaw)

rigid body and flexure variables. Single modes are considered for bending and slosh.

The efficacy of the flexure and rigid body compensators designed and built into the control system is studied on the digital simulation programme by considering various critical flight phases. The phase of flight chosen here for illustration is that at the beginning of aerodynamic control and the parameter chosen is the incremental angle in pitch, $\Delta\theta$. A plot of the fast Fourier transformation (FFT) of $\Delta\theta$ taken from the simulation run is shown in Fig. 5(a). This indicates the flexural activity around 25Hz, which is the bending mode frequency of the vehicle. Also seen are the rigid body responses at low frequencies and the actuator activity around 12 Hz. The bending mode activity is caused by the aerodynamic loads, which try to drive the vehicle to the nominal trajectory.

A plot of the FFT of $\Delta\theta$ obtained from flight records is shown in Fig. 5(b). This also reveals the actuator and bending activity around frequencies 12 Hz and 25 Hz, as observed above. The rigid body response at lower frequencies (1-2 Hz) is not indicated in the figure. Figures 5(a) and 5(b) reveal that the frequencies of actuator and bending activity match only approximately, and the magnitudes experienced in flight, which depend on the actual loading conditions, are larger than those predicted.



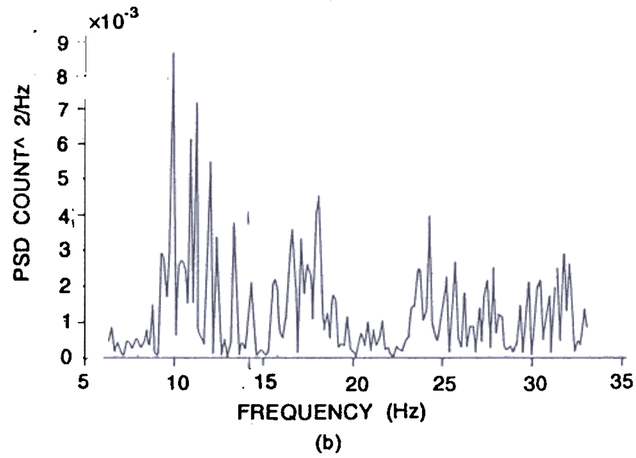


Figure 5. FFT on pitch incremental angle: (a) simulation, (b) flight.

4. ENGINE DYNAMICS

In the subject vehicle, pitch, yaw and the roll control are achieved through engine gimballing. Thrust vector control is employed till the vehicle gains adequate velocity i.e. dynamic pressure. Due to finite inertia of the engine, engine gimballing during control produces reaction on the main vehicle, which is known in the literature as a 'tail wags dog' (TWD) effect. This phenomenon is well-documented as far as yaw and pitch channel control are concerned and the simulation model has these features embedded in it.

The TWD effect produces a pair of zeros on the imaginary axis. At frequencies beyond the TWD frequency, signals tend to get amplified. Absence of proper roll-off can cause control problems if there exist any unmodelled dynamics beyond these zeros. Failure of the vehicle during one of the flight tests had prompted a more careful study on the roll channel. Even, the flights prior to the failure had exhibited a high frequency content in roll rate signal which had remained undetected due to crude sampling. Analysis revealed that the high frequency content in this channel was probably due to the TWD effect in the roll channel and some unmodelled dynamics. The simulation model was updated to incorporate the TWD effect in the roll channel. The kinetic energy and potential energy of the entire vehicle, with the twin-gimballed engines executing pitch, yaw and roll control, were used in the Lagarangean energy approach. The second order terms comprising product terms of yaw and pitch deflections and their derivatives were retained in the simulation model.

In order to get a feel of the involved nature of the problem, the general moment equation along X i.e roll channel in the presence of pitch and yaw motions is given below:

$$\begin{aligned}
M_x = {} & \dot{p}I_{xx} + 2(\dot{r}+pq)[S_e.X_{cg}+I_e]\delta_p \\
& - 2(\dot{q}-rp)[S_e.X_{cg}+I_e](\delta_{r1}+\delta_{r2})/2 \\
& - 2(q^2-r^2)[S_e.e_{gp}](\delta_{r1}-\delta_{r2})/2 + 4pI_e\delta_p\dot{\delta}_p \\
& + 4pI_e(\delta_{r1}\dot{\delta}_{r1} + \delta_{r2}\dot{\delta}_{r2})/2 - 2qr(e_{gp}^2m_e) \\
& + 2I_e\delta_p(\ddot{\delta}_{r1} + \ddot{\delta}_{r2})/2 - 2I_e\ddot{\delta}_p(\delta_{r1}+\delta_{r2})/2 \\
& - 2S_e e_{gp}(\ddot{\delta}_{r1}-\ddot{\delta}_{r2})/2 + 2rS_e e_{gp}(\delta_{r1}\dot{\delta}_{r1} \\
& - \delta_{r2}\dot{\delta}_{r2})/2 - 2S_e\delta_p(\dot{V}_y+r\dot{V}_x-qV_z) \\
& - 2S_e(\dot{V}_z+pV_y-qV_x)(\delta_{r1}+\delta_{r2})/2 \\
& + \text{Potential energy terms.}
\end{aligned}$$

This equation is used in the simulation model while analysing the performance of the system. By neglecting the pitch and yaw motion and considering only the roll channel dynamics, this complex equation reduces to an intitutively appealing equation

$$M_x = I_{xx}\dot{p} + 4pI_e\delta_r\dot{\delta}_r - 2S_e e_{gp}\ddot{\delta}_r = T\delta_r e_{gp}$$

where

$$\delta_{r1} = -\delta_{r2} = \delta_r$$

For control system analysis, after neglecting the product terms, this equation reduces to

$$\dot{p}I_{xx} = T\delta_r e_{gp} + 2S_e e_{gp}\ddot{\delta}_r$$

and in transfer function form to

$$\frac{p}{\delta_r} = (2e_{gp}.S_e/I_{xx})\frac{(s^2 + T/(2S_e))}{s}$$

It is clear from the above equation that the TWD phenomenon produces a pair of zeros on the imaginary axis, emphasising the need for proper attenuation beyond these frequencies. The TWD zero frequency in the case under consideration is around 35 rad/s. Analysis of the roll loop including the TWD zeros failed to reveal the behaviour seen in the flight. However, a study on the hybrid simulation platform, by incorpating the δ_r term inferred from an acceleration sensor mounted on the body of the engine, gave indications of a tendency towards oscillatory behaviour as seen in the flight. The roll loop problem was solved by introducing a compensating filter to cut off frequencies beyond 5 Hz, as was done in the pitch and yaw channels.

5. ACTUATOR ACTIVITY

For the vehicle under discussion in which inertial guidance is used, the sensors for navigation and control are common. As dictated by the better accuracy requirement of the navigation system, the sensed quantities available are, in general, incremental velocities and incremental angles which are obtained by integrating the body accelerations and rates as pulse counts. Since the pulse counts are reckoned as whole numbers, and integration of body rates and velocities over the chosen step length could lead to fractional counts, it implies that the incremental angles and velocities at the end of each step length occur with what amounts to a quantisation noise of less than one pulse count. This noise, compared to the signal, is larger at low dynamic pressure regions when body rates are small, implying that the signal-to-noise (S/N) ratio under those conditions is small. These incremental quantities are used by the navigation algorithm to compute position, velocity and orientation of the vehicle. However, the control system needs vehicle acceleration and body rate, which are obtained by extracting the derivatives from a least square curve fit to the $\Delta\theta$ and Δv samples. The process of differentiation tends to amplify the quantisation noise. When such a noisy signal is used for control, it tends to increase the actuator activity. It is more so when the S/N ratio is small as in the low dynamic pressure region, and the feedback and forward path gains are high.

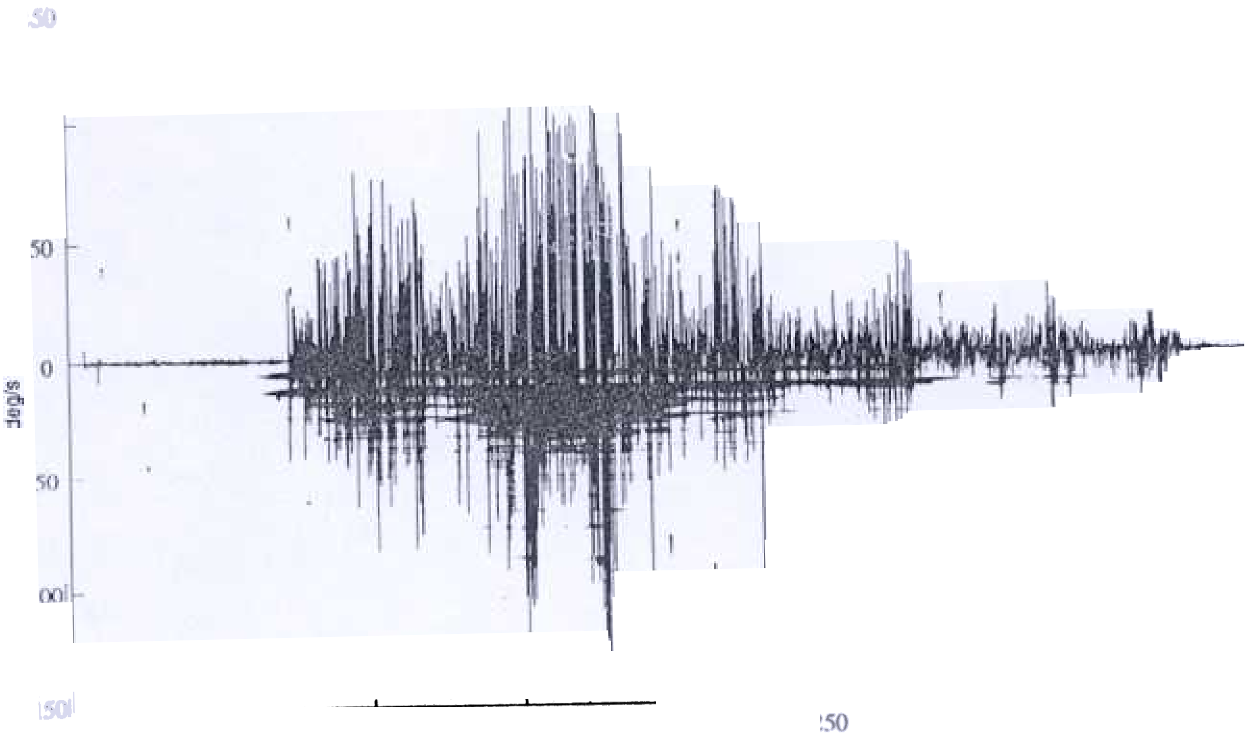
To show the dependence of actuator rate on the body rate level, the scale factor on $\Delta\theta$ (i.e. $\Delta\theta$ per unit count) and Δv is reduced.This results in a larger number of pulse counts, representing the situation of a large body rate signal. Actuator rate activity is shown in Fig. 6(a) for the nominal scale factor and in Fig. 6(b) for a scale factor reduced by a factor of 10. A decrease in the scale factor implies a reduction in quantisation noise, which results in reduced actuator rate.

A rate extraction digital filter was suitably designed, using the results of simulation studies, to reduce levels of noise and obtain acceptable levels of actuator rates. Pronounced levels of actuator activity due to the quantisation noise are also observed in the actual flight records in the low dynamic pressure region. A comparison of the deflection history (simulated and actual flight records) is made in Fig. 7. The pronounced actuator activity due to quantisation noise was first observed in the hardware-in-loop simulation (HILS) runs and was corroborated by the digital simulation runs. This resulted in the provision of an extra accumulator for the hydraulic actuator.

6. NAVIGATION SYSTEM

Strap-down inertial navigation system uses incremental angle and incremental velocity information provided by the Inertial Measurement Unit. It uses a rotation vector estimation method for attitude generation. It has different computation cycles for attitude update, velocity update and position update, determined based on accuracy, and computation overheads. To accommodate such features, the algorithms inherently make some approximations and it becomes necessary to evaluate the performance of such algorithms from the accuracy point of view. The digital simulation model, incorporating oblate gravitational field and geometric features arising due to ellipsoidal shape of the earth, was used to evaluate the performance of the navigation algorithm.

Shown in Fig. 8 is the error plot of the navigation system. The error is defined here as the difference of trajectories obtained from the actual navigation algorithm as employed on the vehicle and that from 6-DOF simulation programme. The navigation algorithm has in it various update cycles for incremental angles and velocities, quaternions, positions and direction cosine metrices. The body rates and accelerations were taken through sensor models. The simulation programme, intended to represent the true position of the missile, *vis-a-vis* the navigation system-computed position of the missile, was run with information update at every computation cycle. The vehicle was guided on to the nominal trajectory using the navigation algorithm information, and 6-DOF simulation programme trajectory was used for comparison with that obtained from navigation module. The error is attributed to the presence of sensor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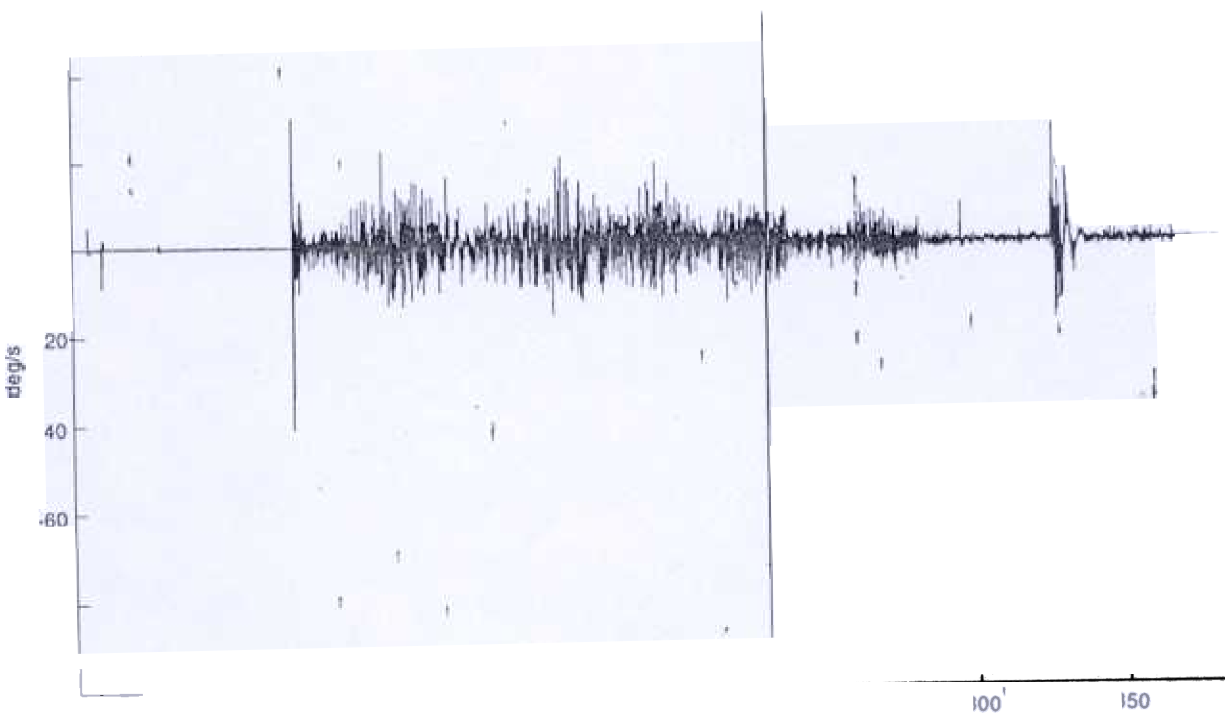
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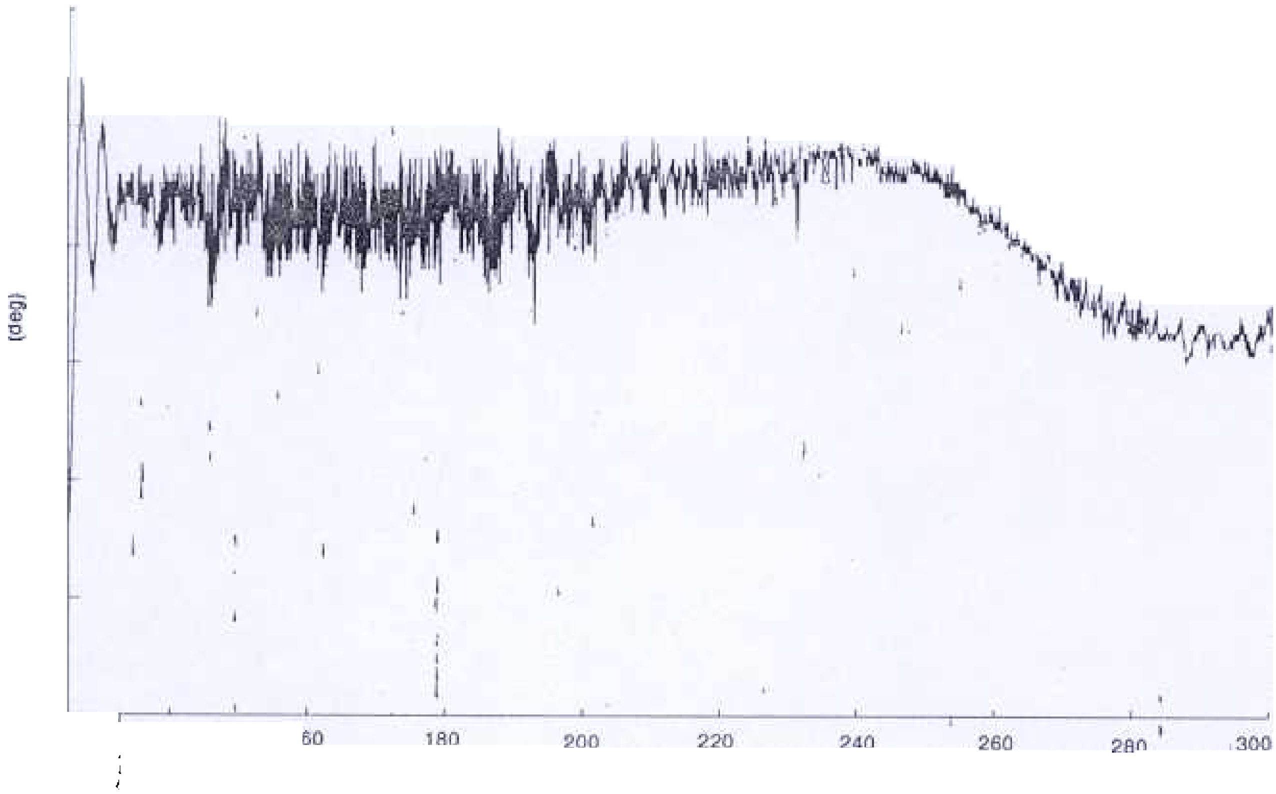

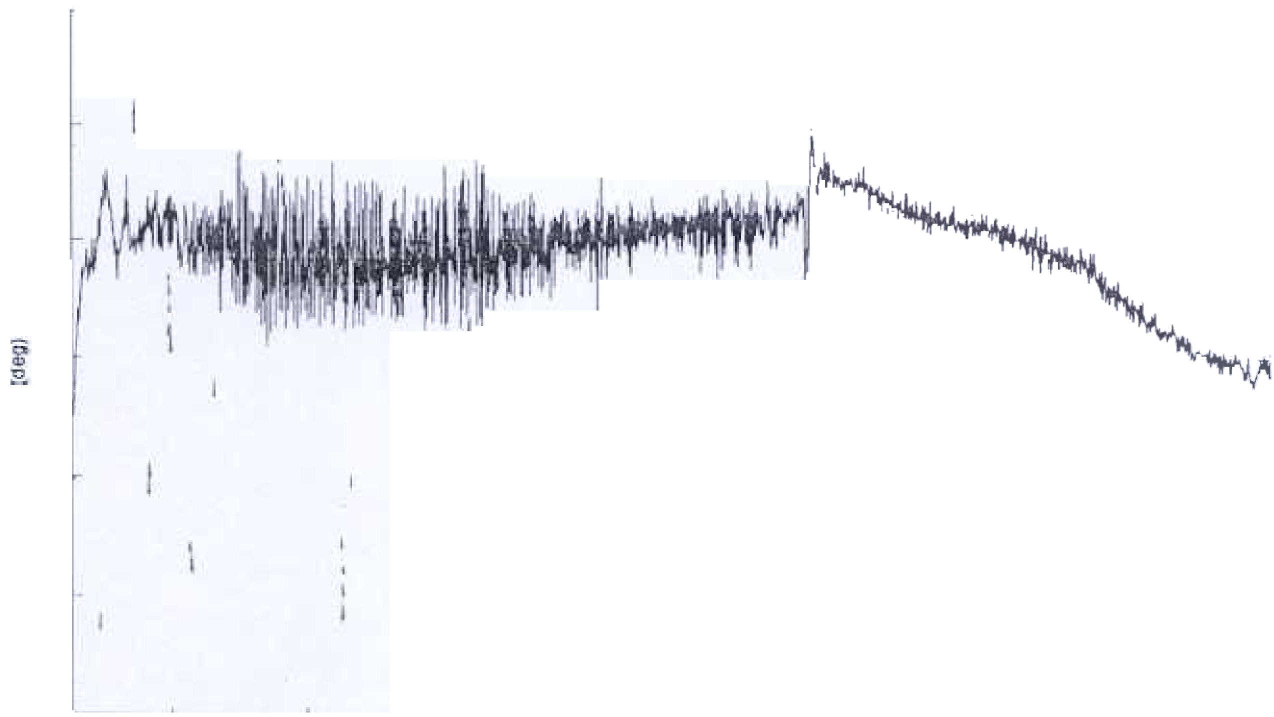

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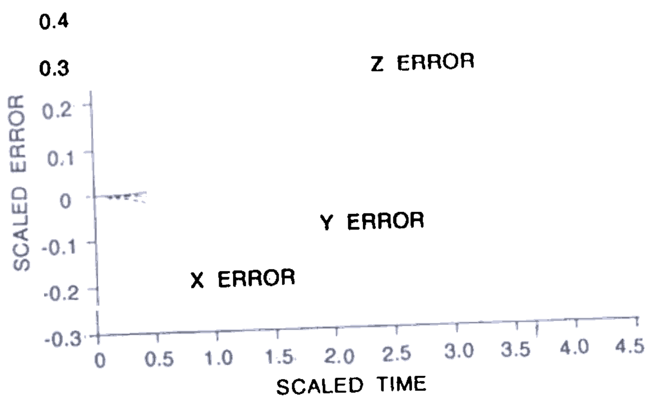

Figure 8. Navigation error

imperfections like drift rate and bias, which are included in the sensor model in the navigation algorithm.

7 GUIDANCE SCHEME

Two-candidate guidance schemes were evaluated using the simulation programme. Both the schemes were evaluated in terms of the terminal conditions achieved, integral of square of latax, and the time required to bring the vehicle within a tunnel of ΔRm around the nominal trajectory from the start of guidance. The aerodynamic and other parameters of the vehicle were perturbed within reasonable limits and the vehicle was subjected to different disturbances, such as wind shear and gust. One of the candidate schemes was chosen for implementation based on the performance as seen on the simulation model.

8. CONTROL SYSTEM DESIGN

One of the guided vehicles under discussion is designed to cater to targets in a certain range and altitude bracket. This results in substantial vehicle parameter variations from trajectory to trajectory. For instance, the mass variation over a trajectory may be as high as 50 per cent of the initial mass. This variation is dependent on the range of the nominal trajectory, since the unspent fuel is different for different ranges. Dynamic pressure varies from 3 KN/m^2 to 150 KN/m^2, and velocity from 1 Mach to 4 Mach. Similarly, there are wide variations in the centre of gravity (CG) position of the vehicle, as also of the other aerodynamic parameters. In addition, the vehicle is statically stable during certain phases of flight and unstable during certain others. The challenge posed to the control system designer to design an adaptive autopilot which works satisfactorily for different target conditions does not need further elaboration.

Different control schemes, as envisaged by the autopilot designer, were evaluated on 6-DOF simulation model to evaluate their performance for different ranges under different parameter perturbations and in the presence of disturbances. Based on the study, feedback was provided to the designer to enable him to arrive at a practical design philosophy which would ensure a stable controller under all conditions. Specifically, gain schedule laws to determine the control gains as a function of dynamic pressure and a technique for pole placement for the known plant parameters were evolved interactively through the use of the simulation model.

The use of gain schedule laws did not give satisfactory results even for a fixed range trajectory. Hence, it was realised that to cover the targets at different ranges and heights, a control algorithm which uses the knowledge of the plant and places the closed-loop poles adaptively would be the most appropriate way of stabilising the plant under all permissible perturbations. This algorithm, namely, the choice of closed loop poles based on parameter estimates, was evolved and fine-tuned through extensive simulation.

9. NOMINAL TRAJECTORY

One of the missile systems under discussion uses a nominal trajectory following guidance scheme. It follows a nominal trajectory which is prestored before vehicle launch. Since the target could be located anywhere in a certain range and height bracket, a software package was developed to generate an appropriate nominal trajectory after launcher and target information is furnished. While generating the nominal trajectory and the corresponding guidance data set, which are loaded

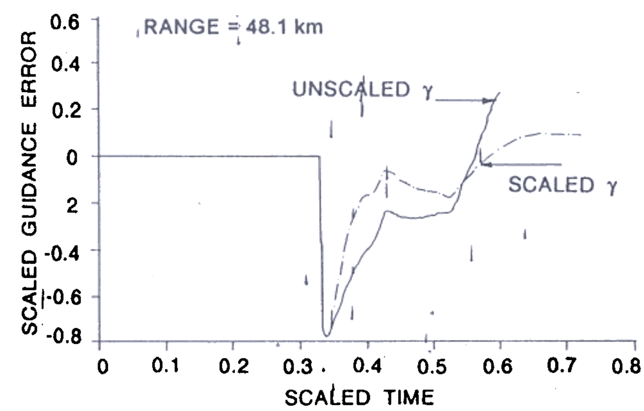

Figure 9. Comparison of guidance error with gamma and scaled gamma

on the vehicle prior to launch, the scheme takes into account the terminal conditions necessary for proper functioning of the warhead. The terminal conditions depend on the type of warhead and also on the vehicle capabilities. It is necessary to evaluate the nominal trajectory generated by this software for different ranges and heights of the target before accepting the software for field use.

The nominal trajectories were thoroughly evaluated through the simulation model to ensure that the vehicle is in a position to follow the generated nominal trajectory under all parameter perturbations and that it meets the terminal and inflight constraints. For this evaluation, 200-300 nominal trajectories were generated and each trajectory was separately evaluated by perturbing the vehicle parameters. The process of developing the software for nominal trajectory generation was carried out interactively in conjunction with the simulation model.

The guidance scheme is based on the stored nominal trajectory which is generated prior to launch, based on launcher and target position information. During nominal trajectory generation (by interpolating from a set of parameters), it is unlikely that the designer would be able to generate a trajectory which exactly spans the launcher-target distance. To overcome this problem, the interpolated trajectory is scaled in range appropriately before loading on the vehicle. However, the γ history is used from the unscaled trajectory. It was observed that there is a tendency for the missile to wander away from the nominal trajectory even after gathering, when the scaling factor is large, since only the down range is scaled. This was seen to be basically due to the use of γ history from the unscaled trajectory. By using the proper γ history from the scaled trajectory, the departure from the nominal trajectory could be kept small (Fig. 9). This indicates the need to keep the scaling factor small. Conversely, the study revealed that the spatial grid points, where trajectory parameter sets were stored, were required to be kept sufficiently close for certain target conditions to keep the scaling factor close to unity.

10. PITCH PROGRAMME SELECTION

For long range missions, the initial pitch programme plays a very significant role in the overall mission performance. Vehicles with high L/D ratios (typically larger than 20) experience large loads, accentuated by severe wind conditions with gust and shear, leading to severity of structural loading. Wind velocities of about 70 m/s exist at altitudes of 14-16 km, giving rise to high loads on the vehicle. Extensive simulation studies have been carried out to arrive at an optimum pitch programme for the problem in hand keeping in view the overall vehicle performance like guidance/control margin, stage separation dynamics, etc. For the pitch programme chosen, comparison of preflight predicted values of various dynamic parameters with the postflight records shows a close match, clearly indicating thereby the fidelity of the simulation model (Table 1). The parameters chosen for comparison pertain to control switchover, stage separation and thrust termination.

11. STAGE SEPARATION

Stage separation, another important event in long range missions, poses severe problems in terms of control availability, arising due to high disturbance torques, if initiated in regions of high

Table 1. Comparison of observed and predicted parameters at different events

Event	Flight recorded (telemetered)	Predicted (based on simulation results)
	Inertial velocity (m/s)	
FTC switchover	V_1	$V_1 + 6.5$
0.2 g sensing prior to stage separation	V_2	$V_2 - 6.5$
First stage separation	V_3	$V_3 - 7.0$
Thrust termination	V_4	$V_4 - 2.3$
	Body rates (deg/s)	
Stage separation		
	3.5	8.0
q	2.0	2.0
	0.8	2.0
	Attitude errors (deg)	
Start of closed-loop guidance		
pitch	5	2
yaw	2	2

dynamic pressure. The simulation programme was used as an effective tool to study the control availability for safe stage separation, and a proper separation logic was arrived at. The dynamic pressure observed in flight was close to the predicted value of 0.03 atmosphere. The body rate comparison at separation is also included in Table 1.

12. CONTROL MARGINS

Different control systems, such as secondary injection thrust vector control (SITVC), fintip control (FTC), thrust vector control (TVC), reaction control (RCS) are provided to ensure adequate control from liftoff till reentry into the atmosphere. The simulation programme has been employed to study the control adequacy (margins) for long range missions under various parameter variations. It has been the basic tool in the design process to provide valuable inputs to the control designer to fine-tune the controller to ensure that no excessive control activity exists anywhere in the flight regime.

Table 2. Comparison of predicted time and actual time for various events

Event	Predicted time (s)	Back-up time (s)	Actual flight time (s)
SITVC switch over to FTC	T_1	T_1+2.02	T_1+0.81
Tank pressurisation command	T_2	T_2+6.88	T_2+2.38
Stage separation command	T_3		T_3+1.85
Closure-of-guidance loop	T_4		T_4+0.78
Thrust termination command	T_5		T_5+1.22

13. EVENT SEQUENCING

The onboard computer (OBC) is the prime decision-making authority in a flight vehicle. A number of decisions based on time or events dictate the trajectory of the vehicle. Under off-nominal conditions (i.e. thrust variations, aerodynamic parameter variations, etc.), critical events like control switchover from one mode to another, onboard tank pressurisation, stage separation, closure of guidance loop, thrust termination, among several others, could occur at instants of time widely different from the nominal values of time. Proper mission sequencing becomes mandatory to ensure that the OBC issues the commands appropriately. The simulation programme has been used extensively under different parameter variations to generate the in-flight event sequence, in addition to the back-up logic in the event of non-occurrence of some of the events. A comparison of the time of occurrence of various events as recorded during the flight, with the predicted values (Table 2), indicates a close match.

14. GUIDANCE MARGIN

For long range missions, the exo-atmospheric trajectory constitutes more than half the total flight duration. The desired range is achieved by injecting the payload with a suitable combination of velocity and flight path angle. These quantities vary for different ranges. The thrust cutoff is achieved by the cross product steering-based explicit guidance scheme. This scheme ensures that at cutoff, the residual velocity to be gained is acceptably small. One of the points of concern here is the adequacy of guidance margin or propulsion reserve at cutoff, which are influenced primarily by thrust variation, drag variation and mass variation. The designer aims at a certain margin based on the simulation runs. The vehicle was designed to have a margin of 3 s under nominal condition. The margin was about 5 per cent of this value under extreme perturbation conditions. One of the important points which emerges from the simulation runs is that, based on the guidance margin information, range enhancement could be attempted or propellant loading could be readjusted to achieve a given range.

15. HARDWARE-IN-LOOP SIMULATION

HILS is a very important activity for the success of a mission. Expensive hardware (electric, pneumatic, hydraulic actuators) used for different control systems get tested with the OBC and sensors in loop, with the actual flight event sequencing. This activity ensures that the performance of the vehicle meets all the mission requirements.

Prior to this activity, it is essential to validate the real-time simulation programme used in HILS with an all-digital benchmark trajectory programme that incorporates the various computation routines and data formats as would be used in the hybrid simulation component of the HILS setup. The digital simulation programme serves here as a truth model to validate the bench mark hybrid programme and in turn the HILS programme in terms of complete missile performance. The digital programme also serves to provide information on whether some of the hardware would be called upon to perform at their extreme performance limits (like gimbal limits, etc.) during the course of the trajectory. Ignoring this support role provided by the digital simulation programme could lead to loss of fidelity of the HILS programme or could cause extensive damage to the hardware when used in the HILS runs.

16. CONCLUSION

The brief discussion presented in the paper covering various aspects of the flight vehicle demonstrates the requirement for a good simulation model for use in design and to make performance predictions. Furthermore, the numerical results presented through figures and tables reveal a reasonably good match which is a requirement for the model to be used as a designer's tool.

The subsystem design, encompassing features like slosh, flexure, etc. is carried out by the designer and tested at some critical operating conditions taken from flight envelope. To test these under representative dynamic conditions, simulation on a model is a must. A reasonably good closeness of the model predicted and flight results serves to enhance the confidence of the designer in the fidelity of the model.

The model was used extensively to evaluate the performance of the navigation, guidance and control systems. It assisted in arriving at suitable locations for the slosh suppression baffles, and an extra accumulator for the hydraulic system (which were borne out by the flight) and in arriving at a suitable guidance scheme, nominal trajectory generation scheme, pitch programme and guidance margin. It is hoped that the foregoing discussion and the results presented demonstrate the role of the simulation model as a useful tool in the hands of a designer.

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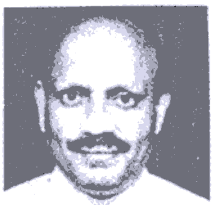

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