OPEN-LOOP ROTORCRAFT SIMULATION MODEL FIDELITY USING ADS-33 CRITERIA

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Abstract

Flight tests have been conducted to measure the handling qualities characteristics of the ALYCAT Lynx (Aeromechanics Lynx Control and Agility Testbed), using the open-loop tests defined in Aeronautical Design Standard ADS-33. The research reported here aims to demonstrate the functional fidelity of the Helisim model of Lynx for handling qualities predictions, by comparing measured and predicted handling qualities as described in ADS-33. The model, which is one of the simplest versions used at the DRA for real time simulations, has a disc model of the main rotor with second order flapping dynamics and quasi-steady inflow determined by momentum theory. The data considered relate to the criteria for hover and low speed flight only. The predictions of on-axis responses have been shown to be satisfactory whereas the pitch-to-roll and roll-to-pitch coupling are incorrectly predicted, as is also the experience of other researchers. The method described in ADS-33 for analysing the height rate response had been used but was considered to be inappropriate for estimating the equivalent time delay. An alternative method, which includes the control input time signal in the identification process, was applied to the measured and predicted vertical velocity. A slight improvement was achieved, but the non-linearities of the measured and predicted responses prevented a reliable set of results being produced with the model structure that was being used.

Symbols

a_0	lift curve slope
k	steady state height rate
p, q, r	roll, pitch and yaw rates
Q	attitude quickness
s	rotor solidity
T	time constant
Z_w	heave damping
Z_{θ_0}	collective control power
ϕ, θ, ψ	roll, pitch and yaw angles
γ	Lock number
γ^*	equivalent Lock number
λ_0	non-dimensional uniform inflow
τ	time delay

Introduction

Flight tests have been conducted to measure the handling characteristics of the ALYCAT Lynx (Aeromechanics Lynx Control and Agility Testbed - see reference [1]), using the metrics defined in Aeronautical Design Standard ADS-33 [2]. This paper examines some of the data collected in order to validate the rotor disc version of the real time, flight mechanics model developed by the Rotorcraft Group at DRA Bedford, Helisim. The research aims to demonstrate the functional fidelity of the Helisim Lynx model for handling qualities by comparing the measured and predicted handling qualities parameters. The Helisim model, which is commonly used for piloted simulation studies at the DRA, is described briefly below. ADS-33 defines the required handling qualities for military aircraft in two principal chapters. Chapter 3 provides requirements in terms of open-loop handling quality parameters. Chapter 4 relies on pilot perception of handling qualities using the Cooper-Harper rating scale, for a number of mission related tasks, or Mission Task Elements (MTEs). This paper uses the open-loop requirements defined in Chapter 3. The criteria are defined for two speed ranges, namely, hover/low speed (0 to 45 knots) and forward flight (speeds greater than 45 knots). This paper relates only to the flight data collected at the hover. The assessments against ADS-33 criteria are not intended as an appraisal of Lynx performance, but rather as a demonstration of a tool for assessing the validity of the model for predictions of handling qualities. Whereas comparisons with ADS-33 criteria have been used to assess the overall functional fidelity, the time histories have been used to provide understanding of discrepancies where necessary. For all cases shown here the tests were flown with the autostabilisation disengaged. The handling qualities aspects addressed in this paper are response type, attitude quickness, height rate response and cross-couplings from pitch to roll and from roll to pitch.

Description of Helisim

The Helisim model is a generic helicopter model which comprises FORTRAN routines to calculate the forces and moments generated by each of the main components of the aircraft, based on the equations described by Padfield [3]. For this study the model was configured as a Lynx Mk 7 for comparison with the DRA test aircraft, the ALYCAT Lynx - see Figure 1. The main rotor is represented using a disc model with second order flapping dynamics and inflow derived from momentum theory. A first harmonic distributed inflow in the longitudinal direction is estimated using the calculated wake skew angle. Blades are assumed to be rigid with an equivalent centre-spring model to represent the stiffness. The tail rotor is modelled by a simple actuator disc with a coning degree of freedom, but no cyclic flapping. Thrust is factored to allow for fin blockage and losses due to tip flow and root cut-outs. Forces and moments from the fuselage are evaluated from 1-dimensional look-up tables based on measured wind tunnel data. Tables are referenced using either incidence (longitudinal, vertical forces and pitching moment) or sideslip angle (lateral force and rolling, yawing moments). The gearings between sticks and blade angles are mostly represented by linear expressions, although the interlink between collective lever / longitudinal cyclic angle is represented by a 2-dimensional look-up table. A linear model of the engine and rotorspeed governor is included.

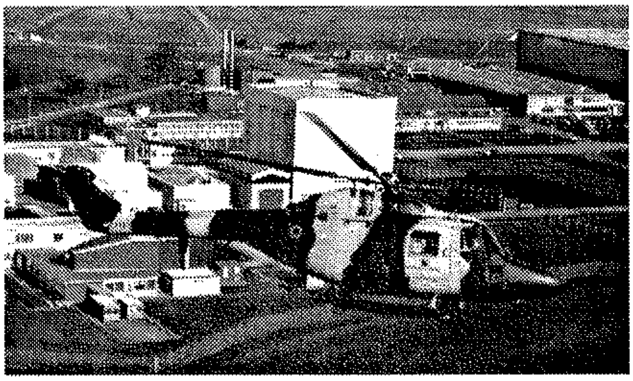

Figure 1; ALYCAT Lynx

Response Type

The response types required by ADS-33 are specified according to the mission type and the usable cue environment (UCE), defining the quality of outside world visual cues. A rate command type is suitable for most missions in a UCE of 1, with the exception of tasks involving divided attention such as sonar dunking and mine sweeping. In this paper, an assessment of the response type in each axis is made by examining the response of the unstabilised aircraft to step inputs on each control. For all cases it was clear that both the model and the real aircraft possessed a rate command response type. The comparisons of the on-axis angular rates obtained from flight and simulation, for each axis, are shown in Figure 2.

The time history comparisons for a step in longitudinal cyclic and lateral cyclic in Figure 2, show there are errors in the peak rates which evaluate to 30% in pitch and 21% in roll. Discrepancies between simulation and flight may be due to the absence of a model for the dynamic inflow, which is expected to give a reduction of about 20%, as discussed by Padfield [4]. The effect

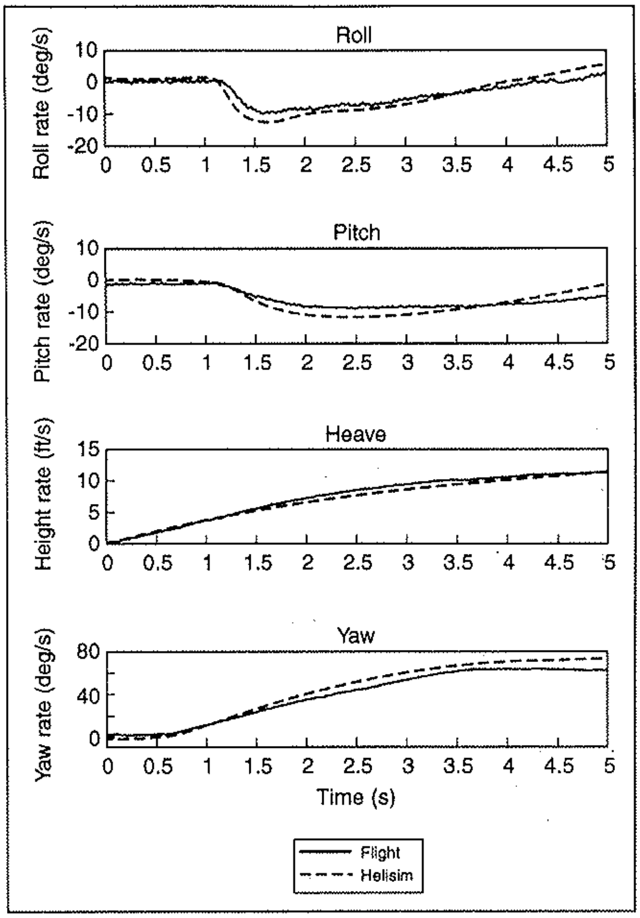

Figure 2; Response types

of dynamic inflow during a steady pitch and roll rate can be allowed for using an approximation which scales down the Lock number, as described by Curtiss and Shupe [5]. The effect is to change the amount of aerodynamic damping on the blade flapping motion, leading to an induced incidence distribution which is equivalent to that created by the reaction moment on the air mass in the vicinity of the rotor disc. An equal and opposite aerodynamic moment acts on the rotor hub. At the hover the equivalent Lock number is given by the expression :-

$$\gamma^* = \gamma \{ 1 / (1 + a_0 s / 16\lambda_0) \}$$

For the Lynx Helisim model at the hover the lift deficiency factor, as applied to Lock number, is equal to 0.665. With this modification incorporated the errors in peak rates reduce to 2% in pitch and 12% in roll (simulation now underestimates the roll rate).

The time history comparison of height rate response in Figure 2, shows that the model matches the flight data well, although there are some visible differences in the rate at which height rate builds such that the flight response builds quicker. The height rate response is discussed later in this report.

The comparison of yaw rate following a step input in pedals shows the model predictions are good. The yaw rate response is revisited below, in the discussion of yaw attitude quickness.

Attitude Quickness

The attitude quickness parameters for roll, pitch and yaw are given by the ratio of peak angular rate to the peak change in attitude angle. The test to measure quickness is performed by applying a pulse input (for a rate command response type) into either longitudinal cyclic, lateral cyclic or pedals to induce a change in either the pitch, roll or yaw angle respectively. The criteria are described in ADS-33 (paragraph 5.3.3) where it is stated that "the required attitude changes shall be made as rapidly as possible from one steady attitude to another". The quickness parameters are plotted against the lowest attitude on the first overshoot and compared with the level 1/2 and level 2/3 boundaries defined in ADS-33. Different criteria apply to "target and acquisition" tasks or more general MTE's.

The flight tests were conducted in clear air and consisted of a number of sharp pulse inputs on longitudinal cyclic, lateral cyclic and pedals in order to achieve a change in attitude. The amount of overshoot of the control relative to trim at the end of the pulse was kept as small as was practicable, to satisfy the requirements of the test. For all tests, the helicopter was trimmed at the hover and the pilot was required to make no off-axis corrections once the input was made. The measured control input signals were supplied to the simulation model.

Pitch Quickness

In both flight and simulation the attitude change is not sustained but washes off to zero and beyond quite quickly making the use of $\Delta\theta_{min}$ questionable, where $\Delta\theta_{min}$ is the minimum pitch angle on the first overshoot. Instead, the values of quickness are plotted against the peak change in attitude angle. The quickness values are given in Table 1 and are plotted on a quickness chart in Figure 3. The criteria boundaries shown are those for

	FLIGHT			SIMULATION		
case	q_{peak} (deg s^{-1})	$\Delta\theta_{peak}$ (deg)	Q (s^{-1})	q_{peak} (deg s^{-1})	$\Delta\theta_{peak}$ (deg)	Q (s^{-1})
1	4.99	1.67	2.99	8.70	5.62	1.55
2	6.41	3.52	1.82	8.73	5.39	1.62
3	8.91	5.15	1.73	11.61	7.72	1.50
4	-9.59	-7.14	1.34	-11.98	-9.21	1.30
5	-11.76	-8.67	1.36	-16.05	-13.10	1.23
6	-5.12	-5.50	0.93	-5.58	-3.52	1.59

Table 1; Pitch quickness

target acquisition and tracking tasks. Variations in the sharpness of the pulse account for the range of quickness values and changes in the peak attitude will depend on the duration and amplitude of the pulse. In general, quickness values from simulation are less than flight by up to 13% (although cases 1 and 6 are exceptions to this). From the quickness chart it is seen that all points from flight and simulation lie close to the level 1/2 boundary, with the exception of the flight data from cases 1 and 6. In particular, the quickness from flight case 1 appears to be an anomaly.

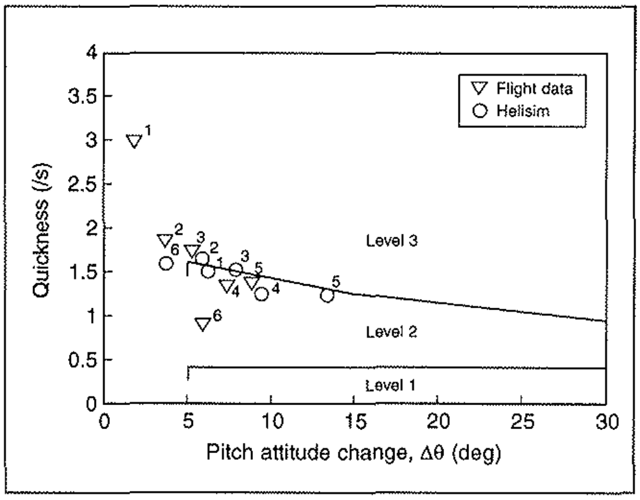

Figure 3; Pitch quickness chart

Figure 4 shows the time history comparisons of pitch rate and pitch angle for case 2. The solid line shows the flight data and the dashed line, the simulation. The error on pitch rate is 36%, which can be reduced to 18% by incorporating the equivalent Lock number, as illustrated by the dot-dashed line. The time history indicates that there is an error in the prediction of the initial pitch acceleration, which is most affected by control sensitivity.

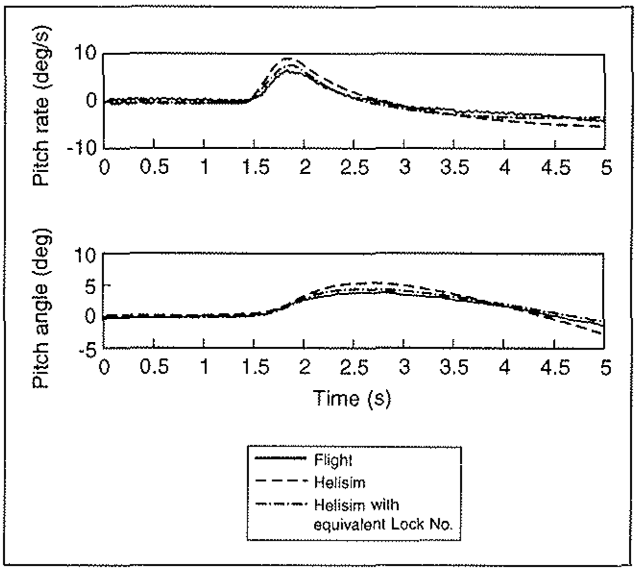

Figure 4; Pitch quickness time history comparisons

Roll Quickness

As with pitch quickness, in the majority of cases the roll attitude angle was not sustained but washed off to zero and beyond making the use of $\Delta\phi_{min}$ questionable. Instead, the peak roll angle was used for plotting values on the quickness chart. The quickness values from flight and simulation are given in Table 2 and plotted in Figure 5. Again, the criteria boundaries seen are those relating to the required quickness for target acquisition and tracking tasks. Pairwise comparison of the quickness values can be observed in Table 2 where it is seen the predictions of the roll quickness are good. With the exception of cases 1 and 2 where the attitude quickness is overestimated by up to 15%, in general the attitude quickness is underestimated by less than 10%. The performance from both flight and simulation appears to be level 1, although it is noted that in most tests the change in attitude was outside the quickness range.

	FLIGHT			SIMULATION		
case	p_{peak} (deg s^{-1})	$\Delta\phi_{peak}$ (deg)	Q (s^{-1})	p_{peak} (deg s^{-1})	$\Delta\phi_{peak}$ (deg)	Q (s^{-1})
1	-21.75	-7.11	3.06	-23.70	-6.73	3.52
2	-20.06	-5.64	3.56	-22.17	-5.50	4.03
3	-16.89	-4.31	3.92	-19.37	-5.15	3.76
4	28.73	7.85	3.66	31.89	9.53	3.35
5	33.96	10.67	3.18	37.09	11.87	3.12
6	31.34	9.42	3.33	35.20	10.84	3.25
7	24.75	6.47	3.83	28.18	7.79	3.62

Table 2; Roll quickness

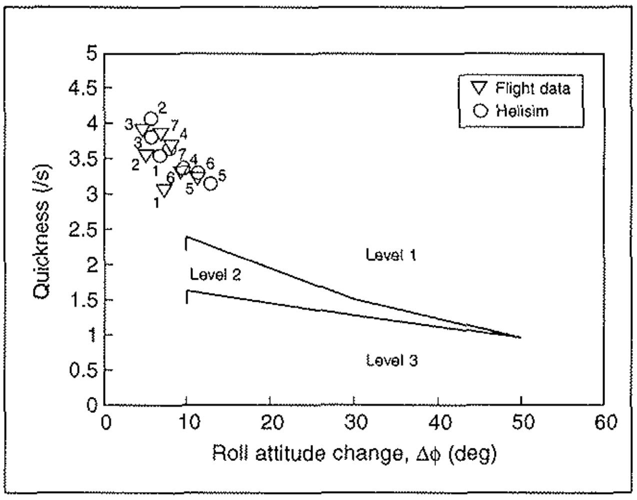

Figure 5; Roll quickness chart

Time history comparisons of roll rate and roll angle are given in Figure 6, where the flight data is plotted with a solid line and the simulation with a dashed line. The error on the peak roll rate is 11% (overestimated), which changes to 11% (underestimated) when the equivalent Lock number is used, as shown by the dot-dashed line.

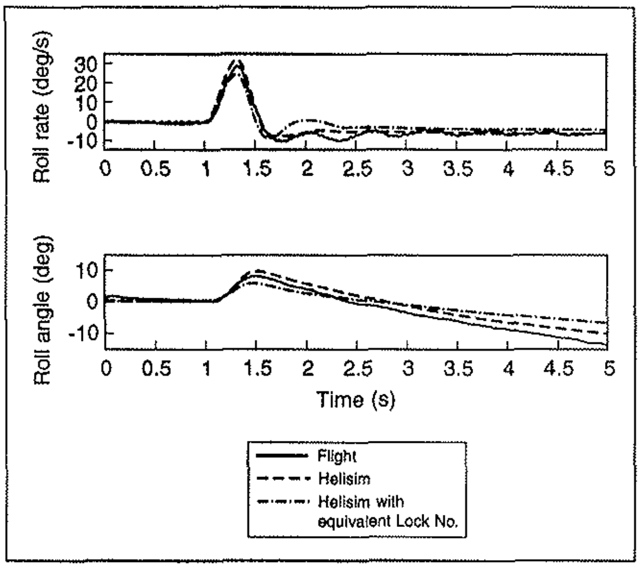

Figure 6; Roll quickness time history comparisons

There appears to be a difference between the accuracy of predictions in the pitch and roll axes which has been seen from the comparison of rates following both step and pulse inputs. After approximating the effects of dynamic inflow using the equivalent Lock number, the pitch rates were overestimated by up to 18% and the roll rates underestimated by up to 11%. For an isolated rotor at the hover it would be expected that, if present, the errors would be similar in pitch and roll. Interaction of the main rotor wake with the fuselage and tailplane may be the source of the difference between the pitch and roll axes. There is also some uncertainty in the values of body pitch and roll inertia, due to the practical difficulties of measuring these quantities.

Yaw Quickness

As seen with the pitch and roll axis, the attitude angle following the input was not sustained and so the peak attitude was used for plotting on the quickness chart instead of $\Delta\psi_{min}$. The measured quickness parameters from flight and simulation are given in Table 3 and plotted in Figure 7. The chart shows that, considering only the points with a yaw angle change of greater than 10 deg, the simulation and flight quickness parameters all lie in the level 2 region with some points from both measured and predicted data sets lying close to the level 3 boundary.

Table 3 shows that the peak yaw rate is overestimated with an error of up to 20% in most cases. The exceptions are cases 3, 5 and 8 where the inputs are smaller and the errors on peak yaw rate are 41%, 31% and 31% respectively. The errors on the predicted yaw angle change are far more variable, due to erratic behaviour of the flight data. The result is that the pairwise comparisons of attitude quickness, are poor. The variability of the flight data may indicate that the trim conditions were not well known. A relatively small drift in any direction would affect the damping forces generated by the tail rotor and, to a lesser extent, the fin.

	FLIGHT			SIMULATION		
case	r_{peak} (deg s^{-1})	$\Delta\psi_{peak}$ (deg)	Q (s^{-1})	r_{peak} (deg s^{-1})	$\Delta\psi_{peak}$ (deg)	Q (s^{-1})
1	13.47	15.56	0.87	16.21	-	-
2	14.65	8.23	1.78	16.29	24.62	0.66
3	9.30	2.98	3.12	13.15	-	-
4	17.32	19.29	0.90	18.83	19.86	0.95
5	9.86	3.97	2.48	12.94	16.43	0.79
6	7.04	5.11	1.38	6.98	3.49	2.00
7	10.14	12.77	0.79	10.74	7.98	1.35
8	-9.01	-3.19	2.82	-11.77	-12.36	0.95
9	-15.21	-19.15	0.79	-15.22	-21.73	0.70
0	-15.77	-30.64	0.51	-17.24	-19.14	0.90

Table 3; Yaw quickness

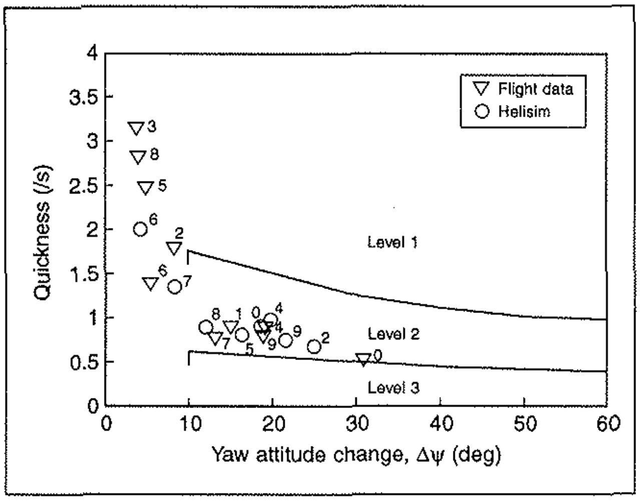

Figure 7; Yaw quickness chart

Height Rate Response

The requirements for the on-axis response to collective are described by the height rate response criterion given in paragraph 3.3.10.1 of ADS-33. The criterion states that "the vertical rate response shall have a qualitative first order appearance for at least 5 seconds following a step in collective input". The transfer function used to model the equivalent first order system is :-

$$(dh/dt) / \delta_c = (k e^{-\tau s}) / (Ts + 1)$$

where,

- τ = time delay
- T = time constant
- k = steady state height rate
- dh/dt = height rate
- δ_c = collective input
- s = Laplace variable

The tests were conducted by establishing an airspeed hover and making an abrupt step input to the collective stick. The response was allowed to develop for at least 5 seconds.

Reliable measurements of vertical velocity at hover and low speed were not available in the flight data so a reconstruction technique was used to obtain the height rate response. The method works by integrating the inertial measurements (linear accelerations, angular velocities and attitudes) according to the translational equations of motion to produce the inertial velocities in each of the body axes. For these flight tests, there were a few seconds of trimmed flight immediately prior to the input where the value of inertial velocity should be constant in each axis. If measurement biases were present on the accelerometers then these will be seen as slopes of constant gradient on the three component velocities obtained from reconstruction during the trim. The gradients were measured and provided an estimate for the bias on each of the acceleration measurements. Subtracting the biases from the accelerometer measurements and repeating the reconstruction, produced estimates of the velocities in body axis. By resolving these velocities into earth axes, the height rate was obtained. From the experience of using this method as part of a kinematic consistency checking program reported by Turner [6], the estimated height rate is considered to be reliable.

The ADS-33 parameters for an equivalent first-order response in height rate were identified using a constrained optimisation routine with a least squares cost function. A goodness of fit parameter, denoted by r^2, defined in ADS-33, is required to be in the range 0.97 to 1.03. This was implemented as a constraint on the optimisation solution. The value of steady height rate identified in the solution was constrained to be within ±40% of the maximum value of height rate on the response being fitted. This value was sufficient to prevent a problem seen in some cases where a solution was selected with a very high steady state and a very long time constant.

Both flight and simulation responses were used to identify an equivalent first-order model of the form shown above, the results of which are given in Table 4, and plotted against the ADS-33 criteria in Figure 8.

The chart in Figure 8 shows that all the test points from simulation gave level 1 characteristics whereas two of the points from the collective up tests from flight lie on the level 1/2 boundary. In both these cases the reason for the poorer handling quality is the extended time delay.

	FLIGHT			SIMULATION		
case	τ (s)	T (s)	k (ft/s)	τ (s)	T (s)	k (ft/s)
1	0.12	2.05	11.43	0.14	3.11	9.79
2	0.21	2.31	12.79	0.08	3.58	12.74
3	0.20	1.86	13.27	0.11	2.15	9.45
4	0.11	3.71	35.30	0.12	3.56	24.98
5	0.12	4.10	25.63	0.08	3.17	19.55
6	0.10	3.31	23.49	0.06	3.69	22.03

Table 4; Height rate response - identified parameters

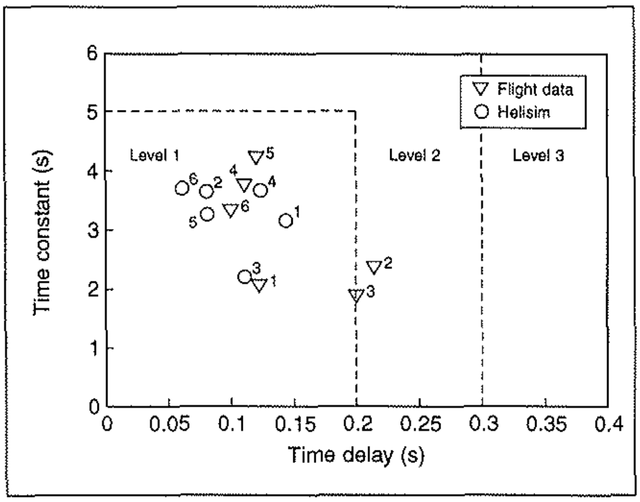

Figure 8; Height rate response - identified parameters

A good estimate of time delay is difficult with these data, as the pilot was unable to inject a perfectly sharp step (as assumed by ADS-33), instead, the inputs were ramps of a duration in the range 0.4 to 0.5 seconds. The point from which the input is assumed to occur was taken as the mid-point of the ramp, but this introduces an uncertainty which precludes a closer scrutiny of the time delay.

For the collective up steps the time constants estimated from simulation data were larger than flight by up to 55%. The height rate builds to its maximum value in a shorter time in flight than in simulation, which is consistent with the absence of a dynamic inflow model in the simulation, leading to an increased retardation of the height rate in the short term. For the collective down steps the results show more scatter, such that in case 5 the time constant is overestimated by 22%; in case 6 it is underestimated by 11% and for case 4 it is well predicted. One possible reason for the variability of the identified parameters is the unsuitability of the identification method. The difficulties of identifying the time delay which were mentioned above may have a disruptive effect on the identification of time constant, as the two parameters are not independent.

An alternative identification was conducted based on the following first-order model of the vertical velocity :-

$$dw(t)/dt = Z_w\, w(t) + Z_{\theta_0}\theta_0(t-\tau)$$

An output error method with a least squares objective function was used to identify values of τ, Z_w and Z_{θ_0}. As the input was included in the identification process, there was no need for an assumption of an instantaneous input, as is the case with the method used in ADS-33. This approach should provide a less ambiguous estimation of the time delay, τ.

Time history comparisons of vertical velocity from flight and simulation, for three collective up and three collective down cases, are shown in Figure 9.

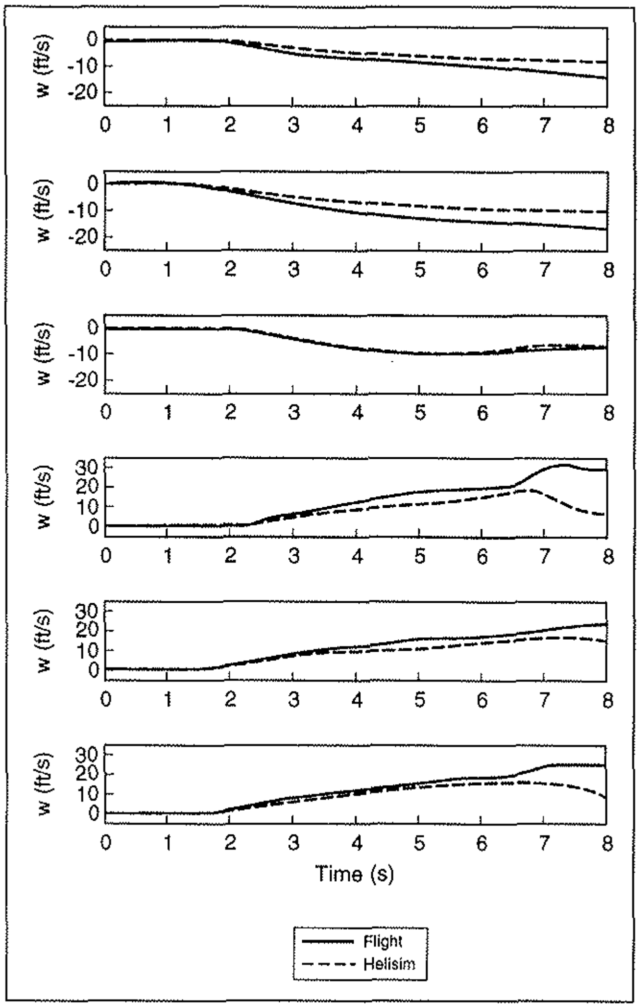

Figure 9; Time history comparisons of vertical velocity

The results of an identification using the flight and simulation responses shown in Figure 9, are given in Table 5. It is seen that, although a more rigorous method has been used to identify time delay, the scatter is still too large to suggest that accurate estimates have been obtained. The values of heave damping from simulation and flight are mostly between -0.38 and -0.41, which corresponds to a time constant of about 2.5 seconds. The exceptions are cases 5 and 6 from flight and case 5 from simulation, where the heave damping is approximately half that of the other cases. The values of control power are well grouped for the simulation but there is much more scatter in the flight results.

	FLIGHT			SIMULATION		
case	τ (s)	Z_w (s^{-1})	Z_{θ_0} (ft/s^{-2} deg)	τ (s)	Z_w (s^{-1})	Z_{θ_0} (ft/s^{-2} deg)
1	0.10	-0.37	-5.83	0.16	-0.38	-3.92
2	0.17	-0.38	-7.15	0.09	-0.31	-4.01
3	0.06	-0.39	-4.87	0.09	-0.32	-4.51
4	0.07	-0.39	-6.32	0.07	-0.41	-4.18
5	0.06	-0.16	-4.07	0.08	-0.41	-4.40
6	0.04	-0.18	-4.33	0.04	-0.21	-3.84

Table 5; Vertical velocity response - identified parameters

Figure 10 shows the results plotted against the ADS-33 criteria, where it is assumed that the time constant is equivalent to the negative reciprocal of the heave damping. As with the height rate response, it is seen that several points lie in level 2 region, although here it is the heave damping which fails to meet the level 1 requirement. In general it is considered that there is an improvement in the agreement between flight and simulation, as the majority of points group together.

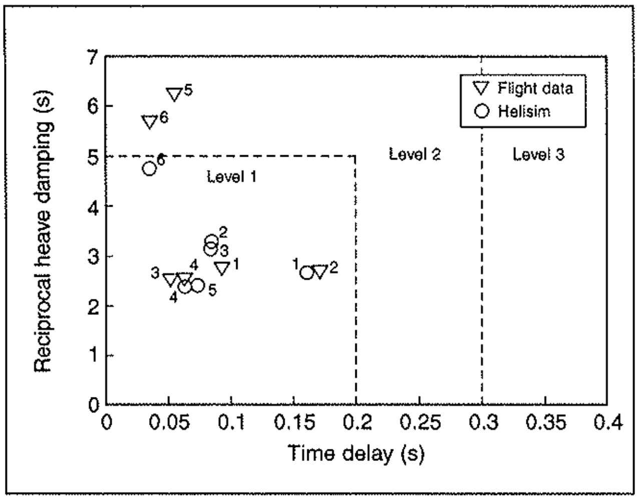

Figure 10; Vertical velocity response - identified parameters

The reasons for the scatter in the results are not clear. Figure 11 shows the variation of the solutions for heave damping and control power, from simulation, for case 1 (collective up) and case 4 (collective down), where the time slice used for the identification is plotted on the horizontal axis and the parameter value on the vertical axis. Figure 12 shows the corresponding plots from flight data. For a linear system with correct model structure, this type of plot would show that as the size of the time slice is increased the solutions converge to a particular solution. In these cases this is true up to the point where non-linearities in the response make the model structure no longer appropriate. The solutions listed in Table 5 are taken just before the solution diverges. Angular motions in pitch and roll axes are the main cause of the non-linearities in the vertical response. In the flight data there were small inputs made to the off-axis controls which limited the excursions in pitch and roll. The simulation model was supplied with the same inputs, which helped to reduce the off-axis motion, but not as effectively as in flight. This leads to increases and decreases in the predicted vertical responses in the last 2-3 seconds, which are not present in the flight data. Increases in vertical velocity seen in the flight data for the collective down cases, are a result of a large forward cyclic inputs which were used to initiate the recovery from the manoeuvre. At the times the inputs were made, the helicopter may have been approaching the vortex ring state.

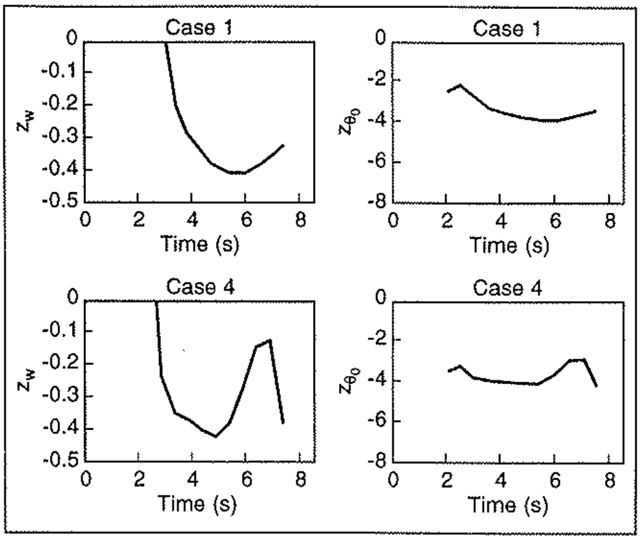

Figure 11; Solution time histories - Helisim

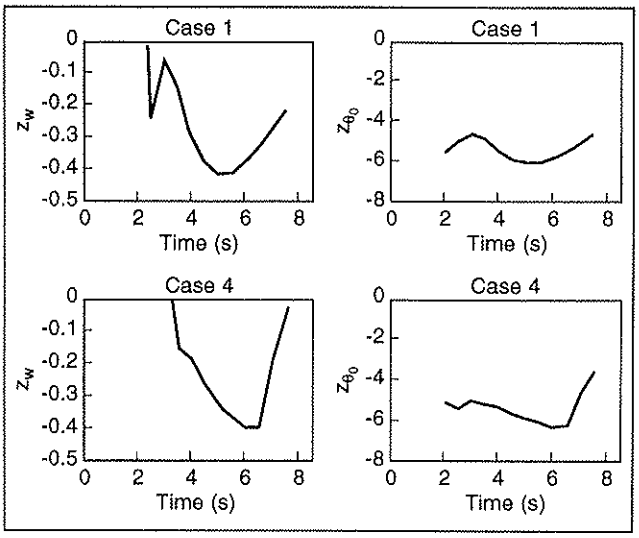

Figure 12; Solution time histories - Flight

A further identification was conducted using predicted responses to a clinical step inputs to collective of varying size. In order to limit the off axis excursions the simulation was run with the pitch/roll/yaw autostabilisation engaged, but with the heave channel disengaged. The results, given in Table 6, show that the heave damping is a function of input size. The low values of damping that have been identified for cases 5 and 6 (in Table 5) may be a result of the input amplitudes being in the region of 0.10.

Δcol	τ (s)	Z_w (s^{-1})	Z_{θ_0} (ft s^{-2}deg)
0.15	0.08	-0.32	-4.05
0.10	0.07	-0.30	-4.02
0.05	0.08	-0.28	-3.99
-0.05	0.07	-0.26	-4.07
-0.10	0.06	-0.25	-4.00
-0.15	0.06	-0.23	-3.97

Table 6; Vertical velocity response using clinical

From the above, it is concluded that the analysis method specified in ADS-33 for quantifying handling qualities in the heave axis, is unsuitable in two respects.

Firstly, the method is not appropriate for test data where the time taken for the collective control to reach its full deflection is significant with respect to the duration of the time delays being identified. The model structure used in the method assumes that the collective step is instantaneous. With the exception of cases where an automatic control input device is available, the time taken to apply a collective input is likely to make the identified value of time delay unreliable. An alternative method, discussed above, attempts to improve the estimates of time delay by including the control input signal in the identification. However, a clear improvement in the results has not been seen.

Secondly, there are significant non-linearities in the vertical axis which prevent the reliable estimation of time delay using a linear model structure. It has been shown that heave damping is a function of input size and that motions in pitch and roll axes will result in both increases and decreases in the vertical velocity which make the linear model structure inappropriate. Perturbations in pitch and roll also lead to translational motions which change the state of the aircraft from the hover, and hence the heave damping and control sensitivity.

Although not clearly demonstrated here, it is believed that including the input in the identification is an improvement on the method defined in ADS-33. The importance of limiting the off-axis motions in flight testing of the vertical response, is also noted

The time histories of vertical velocity from flight and simulation, in Figure 9, show that for both collective up and down steps, momentum theory underestimates the

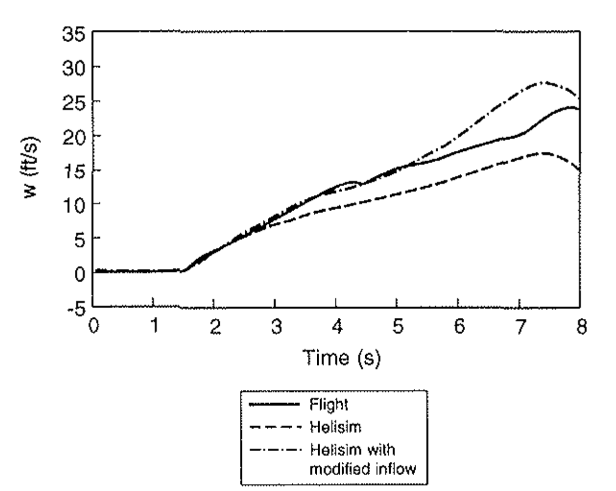

Figure 13; Vertical velocity comparison with modified inflow model

vertical response. In reference [7] Young developed an empirical model of the inflow, based on wind tunnel data, intended to approximate the inflow for all descending flight including the vortex ring. Figure 13 shows a comparison of the vertical velocity from flight (solid line) and simulation (dashed line), for case 5. The third (dot-dashed) line on the plot is the response of Helisim with the modified inflow model implemented. The response up to 5 seconds shows a good improvement of vertical velocity, whereas the last 2-3 seconds of response are influenced by off-axis motions. The modifications have no effect on the collective up response.

An investigation by Houston [8] into the heave dynamics of a coupled body/coning/inflow model in hover compared to a quasi-steady disc model, concluded that the coning and inflow dynamics were required in order to obtain a good estimate of the heave response. It was seen that a fourth order model was necessary to match the response over the first second. This would indicate that the use of equivalent time delay, as in ADS-33, is too simplistic. A dynamic inflow model coupled to a fuselage has been implemented at the DRA and may offer better predictions of the height rate response in the hover.

Pitch-to-roll and Roll-to-pitch cross-coupling

The criteria for pitch-to-roll and roll-to-pitch cross-coupling are defined in paragraph 3.3.9.2 of ADS-33. The criteria specify, for the different levels of handling qualities, the ratio of the peak off-axis attitude during the first 4 seconds of response to the on-axis attitude at 4 seconds, following an abrupt input in either longitudinal (pitch to roll) or lateral cyclic (roll to pitch). The test procedure permits heading changes to be controlled by inputs on pedals, but there were no such inputs made in the flight tests considered here.

The test inputs comprised longitudinal and lateral steps, initiated from a trimmed hover. The size of the inputs depended on what the pilot felt was aggressive enough for the test but not so aggressive as to require a recovery to be made before 4 seconds had elapsed.

Pitch-to-roll coupling

Results for the pitch-to-roll coupling are given in Table 7 and estimates of the size of the inputs are given in the first column. In several cases it was found that the direction of the on-axis motion reversed within 4 seconds, making the use of pitch angle at 4 seconds inappropriate for calculating the cross-coupling parameter. Instead, the cross coupling was taken as the ratio of peak pitch angle to peak roll angle. It can be seen from Table 7 that the peak on-axis (pitch) attitude is predicted by the model with varying levels of accuracy between 6% and 53%, with the exception of case 2 where the error is excessive. Cross referencing the achieved attitude with the size of input, it is seen that the response of the model is approximately in proportion to the size of the input whereas the attitudes recorded in flight are less consistent. There appears to be a randomness in the flight data that could not be expected in the model predictions. As the model has been provided with all the pilots inputs, the differences found in the flight data may have been caused by external disturbances during the flight test resulting from unsmooth air at the altitude at which the flight data was recorded or a hover condition with significant amounts of drift. The off-axis attitude response is, not surprisingly, even less consistent. The cross-coupling parameters are plotted in Figure 14 and show that, in general terms, the flight data for forward stick inputs are well grouped whereas those for aft input show a large scatter. For the forward stick results, the coupling to roll from flight is about 30% of the simulation prediction, but is in the opposite direction. Similar conclusions can not be drawn for the stick aft cases due the scatter on the flight data. It is intended to repeat these flight tests to obtain better data for the stick aft cases.

	FLIGHT			SIMULATION		
case (input size)	θ_{pk} deg	ϕ_{pk} deg	ϕ_{pk}/θ_{pk}	θ_{pk} deg	ϕ_{pk} deg	ϕ_{pk}/θ_{pk}
1 (0.09)	17.45	-7.80	-0.45	16.29	-12.11	-0.74
2 (0.11)	5.44	-13.93	-2.56	20.26	-22.45	-1.11
3 (0.08)	25.67	14.48	0.56	12.06	-10.79	-0.89
4 (-0.12)	-39.41	-6.38	0.16	-23.57	11.87	-0.50
5 (-0.09)	-26.31	-5.21	0.20	-16.93	10.86	-0.64
6 (-0.17)	-36.56	-1.77	0.05	-27.12	14.48	-0.53
7 (-0.11)	-22.44	-0.61	0.03	-23.77	12.72	-0.54

Table 7; Pitch-to-roll cross-coupling

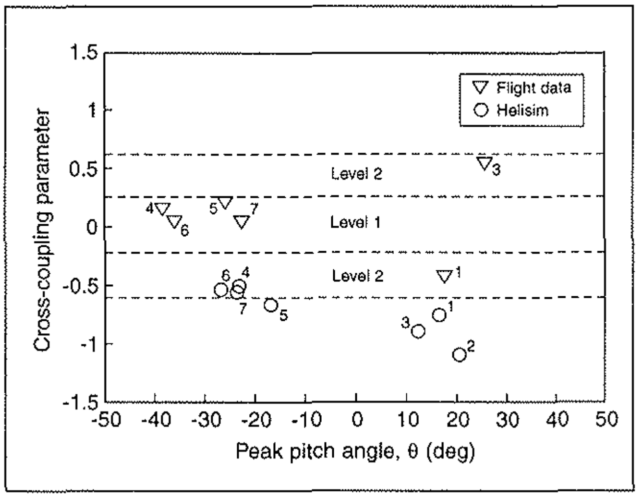

Figure 14; Pitch-to-roll cross-coupling

Roll-to-pitch coupling

Results for the roll-to-pitch coupling are given in Table 8 and share much of the same features seen above for the pitch inputs. Again, the on-axis (roll) attitude was used to calculate the cross coupling parameter, instead of the attitude angle at 4 seconds. The predicted on-axis attitudes seem to have a magnitude approximately in proportion with the size of the test input (given in the first column of Table 8), whereas those measured in flight are less consistent. A plot of the cross-coupling parameters is given in Figure 15 and shows that the flight data for stick inputs to the right are well grouped whereas those for inputs to the left have a very large scatter. The coupling to pitch for inputs to the right are smaller in flight than in the model, and are again in the opposite direction. As with aft inputs, the scatter on the flight data for inputs to the left precludes any conclusions being drawn for these cases. The tests will be repeated in an attempt to obtain data with less scatter.

The works reported by von Grunhagen [9] and Keller [10] demonstrate that it is a common problem that flight mechanics models of helicopters are unable to predict the cross-coupling accurately. It appears that most models are consistent in that the off-axis motion following an input to

	FLIGHT			SIMULATION		
case (input size)	ϕ_{pk} deg	θ_{pk} deg	θ_{pk}/ϕ_{pk}	ϕ_{pk} deg	θ_{pk} deg	θ_{pk}/ϕ_{pk}
1 (-0.11)	-11.71	3.10	-0.26	-19.31	-17.02	0.88
2 (-0.13)	-18.37	-8.80	0.48	-21.40	-18.66	0.87
3 (-0.13)	-26.39	-33.19	1.26	-22.13	-19.01	0.86
4 (0.10)	21.92	-10.47	-0.48	9.82	11.71	1.19
5 (0.16)	20.31	-10.01	-0.49	24.67	9.62	0.39
6 (0.12)	21.75	-6.28	-0.29	15.64	15.61	1.00

Table 8; Roll-to-pitch cross-coupling

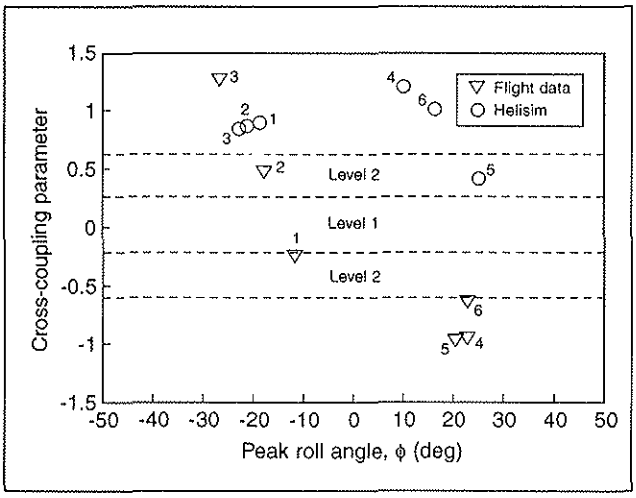

Figure 15; Roll-to-pitch cross-coupling

cyclic occurs in the opposite direction to that seen in flight, in both pitch and roll axes. The studies account for the poor prediction of cross-coupling by incorporating models of different phenomena. Von Grunhagen's study included the effect of swirl in the rotor wake and Keller modelled the harmonic components of inflow due to steady pitch and roll rates which distort the wake. The correct prediction of cross coupling remains a challenge and will be the subject of on going research to establish what are the dominant mechanisms which produce the coupling.

DRA High Fidelity Model

A new aeroelastic model has been developed at DRA Bedford for real time applications. The main features of the model are an individual blade representation of the main rotor including blade elastics. Blade deflections are represented using mode shapes as described by Simpson in reference [11]. The numbers of modes used to model flap, lag and torsion degrees of freedom are selected by the user. The inflow is modelled by a Peters-HaQuang dynamic inflow model as described in reference [12]. Blade section aerodynamic characteristics are determined using look-up tables indexed with incidence and Mach number. Fuselage, empennage and tail rotor are modelled in the same way as the Helisim model described earlier. A study of the open-loop ADS-33 characteristics is currently being undertaken. Also the closed loop characteristics of the model have been assessed in real time and will be reported in the near future.

Conclusions

A validation study of the DRA Helisim model has been conducted using flight data collected on the DRA's ALYCAT Lynx.

The unstabilised aircraft has been seen to have a rate response type in all axes, by both the flight and the simulation data.

In the majority of cases the pitch attitude quickness is predicted with an error no larger than 15%. Peak pitch rate is overestimated by up to 36% but this reduces to 11% if the dynamic inflow is accounted for by using the equivalent Lock number. Roll attitude quickness is also predicted within 15%. The peak roll rates are overestimated by up to 11%, but become underestimated by 11% when the equivalent Lock number is included. Yaw attitude quickness is poorly predicted despite the peak yaw rate being predicted within 20% in most cases. The reason for the errors is the apparent variable levels of damping in the flight data, which affected the amount by which the yaw attitude changed. For all axes the attitudes changes were not sustained for more than a few seconds, but instead, washed off to zero and beyond. This prevented the use of the attitude at the base of the first overshoot, as defined in ADS-33.

The height rate response has been analysed using the methodology proposed in ADS-33. It has been found that the method did not provide reliable estimates of time delay, due to the violated assumption that the collective input was a perfect step. An alternative identification method which alleviates this problem by including the input signal in the optimisation, has been used. Non-linearities, which are not accounted for in the model structure, prevented the new method from providing a reliable set of results. It has also been shown that heave damping, and hence time constant, is a function of input size.

The cross-coupling in pitch and roll predicted by Helisim is seen to be underestimated and in the wrong direction compared with flight. Similar findings have been documented by other researchers.

In general, the flight data collected at the hover has not always provided a consistent picture for comparisons with simulation results. This is almost certainly due in part to the difficulties of establishing a good trim condition, where the airspeed is zero.

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