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BOEING

Mr. M.H. Carr
United Kingdom Ministry of Defence
Equipment Support (Air)
Chinook IPT
UK Resident Project Team (CHPE)
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Dear Mr. Carr:

Subject: Chinook ZD576 Accident on the Mull of Kintyre – Evidence before the House of Lords Select Committee: Request for Boeing Review and Re-simulation.

Reference: Chinook HC Mk2/2A/3 – Contract CON1B/1281 – PDS Task 07A001ES; (AIR)(PHL)/140/7/PDS/01, Dated 5 March 2002.

The Subject request was conveyed by the Referenced letter from the Chinook IPT asking Boeing to provide the following:

1. Review for accuracy and comment on MOD selected testimony before the House of Lords Select Committee on Chinook ZD576.

2. Perform the following:
    a. Review original B-29 modeling efforts performed in support of the Board of Inquiry.
    b. Using Boeing's BH Sim math model, re-run the original simulation with a view toward validating the original B-29 simulation effort.

c. Using Boeing's BH Sim math model, run new simulations as required that reflect the latest information available from the AAIB report on the Subject accident.
d. Provide descriptions of both the B-29 and BH Sim math models.

3. Provide an analysis of the overall flight with regard to time, speed and distance with an eye toward establishing a minimum speed at the waypoint change.

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We are providing the following in response to the Subject request:

- Enclosure 1 responds to the first task element by providing the requested commentary.
- Enclosure 2 responds to items a, b, and c of the second task element.
- Enclosure 3 responds to item d of the second task element.
- Enclosure 4 provides a response to the third task element.

The provision of this data constitutes the complete response to PDS Task 07A001 and Boeing considers this task to be closed.

Sincerely,

Ronald L. Gionta

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Enclosure 1 – MOD Requested Boeing Commentary
Enclosure 2 – 8-7430-1-3724 Response of HC Mk2 Helicopter to Longitudinal & Collective Control Inputs
Enclosure 3 – 8-7430-1-3719, 5 June 2002, BH Sim and B-29 Model Descriptions
Enclosure 4 – Mull of Kintyre – Analysis of Available Data

The MOD requested that Boeing provide comments on selected sections of the HOL Select Committee Report on Chinook ZD 576 (Original testimony is provided followed by Boeing comment in italicized, blue text).

Summary of observations/comments:

- *The Boeing simulation work was exactly that – a simulation, not a factual reconstruction and should hardly be considered "defective". The B-29 transient analysis program used in the simulation did not itself model FADEC, but the simulation was adjusted to reflect engine responses comparable to FADEC. The simulation was an analysis tool requested by the BoI/AAIB.*
- *The Boeing simulation was a small element of a large body of information on the accident and was not essential to the conclusions of the board.*
- *The original modeling used a nominal LCT schedule for the forward and aft rotors rather than a more realistic range schedule. The effect of was to skew our results to the high side on airspeed. Our latest simulation tools reflect the broader LCT schedules.*
- *Data unavailable to Boeing at the time (RNS252) indicates GS≅160 knots, IAS≅135 knots just prior to impact.*
- *The HC Mk2 can readily achieve 150kt/1000fpm climb rate.*
- *Post-impact power-down indications on the cockpit rotor-tach are not reflective of pre-impact rotor speed. Especially when considering a desyncronization occurred following initial impact.*
- *Excessive rotor coning does not occur until very low rotor speeds are reached (<<91%).*
- *Sufficient factual evidence existed for the BoI to determine the probable accident cause without the Boeing simulation.*
- *Data downloaded from the #2 engine DECU showing no evidence of torque or temperature exceedance and the matched power conditions of the engines post-impact indicate there was no sustained (~1 sec or more) emergency power demand. No other evidence indicated any FADEC or engine faults.*
- *For there to have been a multi-axis lower control jam that rendered the aircraft incapable of turning or climbing at its maximum emergency capability after the waypoint change and then subsequently clears to allow a last second pitch-up maneuver is considered to be extremely improbable.*
- *Speculation on a number of possible alternative, but relatively improbable, explanations for what might have happened to ZD-576 pales when compared to the classic CFIT (controlled flight into terrain) causal chain established by the BoI.*

Detailed comments follow:

It is essential that these comments be interpreted in the proper context and to that end the following clarification is provided:

> *Boeing support of aircraft accident investigations is handled by each operating unit's Air Safety Investigation office. Air Safety Investigation specializes in bringing together Boeing resources and expertise to provide assistance to appropriate investigating authorities. This support is focused on the determination of cause to ensure fleet safety. This was the case with the support that was provided to the RAF Board of Inquiry (BoI), through the AAIB, following the 2 June 1994 accident involving HC Mk2 ZD576 on the Mull of Kintyre.*
>
> *Since our primary interest is flight safety, our comments are focused on balanced information and probability rather than absolute fact, since, in the engineering/safety realm at least, things are usually more approximate than absolute. In many instances with aircraft accidents, what may be considered as fact, is not really fact at all and may even be contradictory of other" facts". For this reason, every "fact" must be weighed according to its relative probability of truth. It is rarely known with absolute certainty exactly what all the reasons for an aircraft accident are. One can only determine a most probable cause and as many accidents in the past have amply demonstrated, even with cockpit voice*

and flight data recorders, there is always room for doubt. Consistent with our understanding of the RAF BoI's charter, as the investigating authority in this case, a determination of probable cause ("the selection of an inappropriate rate of climb to overfly the Mull of Kintyre safely") was made based on a preponderance of the evidence.

The focus of the HoL Select Committee has clearly been the establishment of doubt concerning the reviewing officers meeting the standard of proof for a finding of negligence against the pilots of ZD576. We have no interest in blame but are concerned with Chinook fleet safety. We are interested in the whole picture of what may have occurred to cause this tragic accident and what may be learned to prevent such accidents in the future.

While it seems improbable that such an experienced aircrew could have ended up flying an airworthy aircraft into terrain, the alternative explanations are even more improbable. A lower controls jam affecting at least three independent axes which subsequently relieves itself to allow for a pitch-up maneuver prior to impact is certainly much more improbable.

From the Summary (paras 3,5 & 6):

3. The House of Lords appointed us, in July 2001, to consider whether this finding of negligence was justified. In preparing this report we have not only considered the evidence which was before the investigating board and hence the reviewing Air Marshals, but also additional evidence both oral and written. This additional evidence related among other things to the weather in the area at the time of the crash, to various mechanical problems which had affected Chinook Mk 2 helicopters since their introduction into RAF service, and to what reliance could be placed upon the results of a mathematical simulation carried out by the aircraft manufacturers, Boeing, to determine the movements of the aircraft during its last few seconds of flight.

5. From this information, and from certain other information as to the position of the aircraft and some of its components at impact, Boeing concluded that the aircraft had made a final "flare" upwards some 4 seconds before impact, and that for an unspecified time before this the aircraft was climbing at a rate of 1000 ft per minute at an airspeed of 150 knots, which with a tail wind of 25 knots gave a groundspeed of 175 knots or thereabouts.

   Boeing produced results from its simulation concluding "that initial airspeeds approaching 150 kt with climb rates approaching 1000 ft/min are most likely to result in longitudinal/collective pull-up maneuvers which meet all the specified criteria."

6. For reasons set out in our report, we consider that Boeing's conclusions cannot be relied upon as accurate. Since these conclusions are the basis for the conclusions of the investigating board and the Air Marshals that the aircraft was under control at the time of the final flare, it follows that there is insufficient evidence to the required standard of proof that this was the case.

   We fail to see conclusive evidence that the results of the Boeing simulation were the basis for the BoI's and the Air Marshal's conclusion that the aircraft was under control "at the time of the final flare". By our reading, it seems that the decision that the aircraft was under control just prior to impact was based on a large body of information, evidence and analysis of which the Boeing simulation was a small part. One significant element of that conclusion may also have been that there was no evidence that the aircraft was not under control just prior to impact.

From the main report: Evidence bearing on the Boeing simulation

120. Since the investigating board and the Air Marshals placed considerable reliance on the Boeing simulation it may be convenient to refer to it again in more detail at this stage. Before doing so

however it is necessary to examine the functions of the two major controls in the aircraft. The collective increases or decreases the pitch of all the blades of the rotors as it is raised or lowered thereby causing the aircraft to climb or descend. At the same time movement of this control by connection to the FADEC system increases or decreases engine power to maintain rotor speed at approximately 100 per cent. The cyclic stick alters the pitch of the rotor head which is then tilted in the direction in which the aircraft is intended to go, namely forward, sideways or backwards.

Actually, the cyclic stick works quite a bit differently for a tandem rotor helicopter. Pitch control for forward and backward motion is provided by differential collective control of the forward and aft rotors. Sideward control is provided by cyclic tilt of both forward and aft rotors. Yaw control by the pedals produces differential cyclic pitch of the forward and aft rotors.

121. Detailed examination by the AAIB of the flight control system disclosed that the DASH extensions found did not correspond to a high speed level flight condition whereas the LCTA extensions did, and it appeared possible that the settings could reflect a dynamic aircraft manoeuvre at the point of impact. Boeing were therefore asked to undertake a study to assess the consistency of the settings and to define the possible manoeuvre. The simulation was a mathematical exercise which, as Mr Cable stated, was "looking really for fairly gross manoeuvres over a pretty short period of time" (Q 957). It was not intended to produce an accurate reconstruction of events but rather to demonstrate what could have happened within certain parameters (Q 982).

This assessment of the Boeing simulation efforts is essentially correct. The simulations were intended to find convergence with a set of end data provided by the BoI and the AAIB from known or deduced conditions at impact. It was clearly known to AAIB and the BoI that this was a simulation and just another set of information to be considered in context with the rest of the information on the accident.

The simulation/modeling efforts were conducted at the request of the BoI, with technical oversight from and coordination with the AAIB, who were advised fully on the reliability and limitations of this type of analysis.

122. Mr Cable provided Boeing with his findings from the wreckage of the aircraft as to pitch attitude, flight path angle, actuator extensions and ground speed together with certain other information provided by the board (Q 950). Information from the SuperTANS disclosed that:

(i) When the way point change was made the aircraft was 0.81 nautical miles from way point A (0.95 nm from where it crashed), on a bearing of 018 degrees T to way point A (022 degrees T to where it crashed). No information as to height, speed or course was available; but, on the assumption of a straight course at a steady 150 knots, Racal (manufacturer of SuperTANS) suggested that this change was made "about 20 seconds before the accident".
(ii) Some 15-18 seconds before impact the aircraft was at a height of 458 plus or minus 50 feet at an unknown position, on an unknown course and at an unknown speed.

123. The Boeing simulation considered a wide range of possible starting conditions, i.e. conditions pertaining immediately prior to a final manoeuvre. Having rejected possible conditions at an airspeed of 135 knots, they concluded that an airspeed of 150 knots (groundspeed 174 knots) with a ROC of 1000 feet per minute provided "a ready match" with the criteria and was therefore the most likely (AAIB statement para 8). From this simulation, using among other things the state of the actuator extensions and attitude of the aircraft as found by Mr Cable, Boeing deduced that 2.9 seconds after the final manoeuvre had been initiated the airspeed was about 135 knots, the rotor speed 204 rpm or 91 per cent design speed and the groundspeed 158 knots.

As previously mentioned, Boeing considered a set of conditions provided by the BoI/AAIB which consisted of the following:

Fixed Initial Conditions
Aircraft Gross Weight (All up Mass) – 37,700 lbs. (17,100kg)
Center of Gravity – Midpoint of Allowable Range (STA 325)
Density Altitude – 420 Ft.

Variable Initial Conditions
Indicated Airspeed – 135 & 150 knots (for practical purposes TAS was used)
Rate of Climb – 0, 500 & 1000 ft/min.

Conditions at Initial Impact (Power Supply to LCTA & DASH Assumed at This Point)
Aircraft Pitch Attitude - 30º Nose Up (Estimated/derived)
Flight Path Angle - 20º Climb (Estimated/derived)
Cyclic Trim – 3.8º or more (Both LCTAs at or close to full extension)
Dash Actuator Extension – 23%

Given the fixed initial conditions Boeing was requested to look at combinations of variable initial conditions combined with various combinations longitudinal and collective stick inputs that would simultaneously meet the conditions at initial impact.

Notice that the 500 ft/min originally specified and airspeeds between 135 and 150 knots were never explored. This was due to the fact that this simulation effort was rather labor intensive and not enough time was available to fully explore all combinations within the prevailing time constraints. At the time, it probably was determined that the work done was sufficient to look at possible pre-impact maneuvers and see if they produced the conditions found at initial impact. Since this was a simulation, and thus an approximation, it was considered sufficient for its intended purpose.

Another point to consider is that at the time the simulation work was being done at Boeing, we were not privy to most of the information that the BoI and the AAIB had (quite probably there was information, now available, that they did not yet have). Boeing was not requested to consider whether its simulation results correlated well with other accident data. Indeed, the resolution of the simulation may have been considered adequate by the Board in terms of its original intent – a simulation/approximation. Specifically, there was no requirement imposed on rotor RPM. Rotor droop simply implies that the aircraft is demanding more power than the engines can produce. We now believe that prolonged rotor droop did not occur based on the data extracted from the #2 DECU.

124. It will be noted that, apart from the evidence of Mr Holbrook, there was no other evidence of the speed of the aircraft prior to the moment of impact. In the absence of a time at which the waypoint change took place and a position at which the height of the aircraft was disclosed, there are no facts from which the speed of the aircraft prior to the initiation of any final manoeuvre could be calculated. It follows that Boeing's 150 knots airspeed is a postulated figure rather than one calculated from known facts. This postulated figure then becomes the basis for the further postulated figure for ROC. Furthermore, the simulation gives no indication of the length of time prior to the assumed final manoeuvre during which the aircraft had been proceeding at the postulated airspeed.

It should be noted that the RACAL Report on the RNS252 SuperTANS Navigation System provided steering data, calculated once per second, that prior to power loss recorded time and range to waypoint B of 32.8 minutes/86.63nm. Aircraft groundspeed just prior to impact can be calculated from this and yields 158.5 knots (or 162.6 knots from the display at power down). We *suggest that this constitutes evidence of the aircraft's speed prior to the moment of electrical power loss to the TANS.*

The final position of the longitudinal cyclic trim (LCT) actuators gives a very good indication of the airspeed at the instant of power loss to those actuators. On a nominal basis, this airspeed is 144 knots based on forward LCT actuator extension, and 147 knots based on aft LCT extension, with a +/-10 knots tolerance on each. Since the high climb angle and climb rate indicated at the first impact site are unsustainable by engine power alone, the aircraft must have decelerated immediately prior to impact, exchanging kinetic energy (airspeed) for potential energy (altitude). Furthermore, additional airspeed would have been lost at, and subsequent to, first impact. Therefore the airspeed three or four seconds before first impact must have been at least somewhat higher than the specific airspeeds indicated by the final LCT actuator extensions above when the actuators lost power. Hence the 150 knots postulated airspeed was entirely within the realm of possible airspeeds immediately prior to impact. Our more recent simulation efforts using a more robust simulation model and additional data about the mishap have shown a more likely entry airspeed to be in the range of 135kts (note that in many cases we are using TAS and IAS interchangeably since they close enough for practical purposes under these conditions).

At the time the original simulation work was done several postulated combinations of initial airspeed and climb rate were considered. The 150kt/1000fpm was the only combination which simultaneously met all four criteria of flight path angle, pitch attitude, DASH actuator position and LCT actuator extensions at the moment of impact, and it did this when not just one, but three different longitudinal and collective control input combinations were applied at the initial point to generate the pitch-up maneuver. All postulated maneuvers initiated at the lower airspeed (135 kt) failed to simultaneously meet one or more of the four criteria but in many cases were close. Our more recent simulation work has disclosed that many cases at 135 knots do meet all the specified criteria.

125. The rotor speed of 91 per cent derived from the simulation is significantly different from that of 100.5 per cent found by the AAIB on the instrument panel (statement para 7.2.2). Maintenance of rotor speed at or about 100 per cent design speed is of critical importance for safe helicopter flying. If rotor speed falls much below 90 per cent there is a danger of the blades of each rotor coning up and meeting at the top due to the reduction in centrifugal force which at higher speeds keeps them apart (Q 918). FADEC is designed to keep rotor speed at normal design speed and, if rotor speed had fallen to 91 per cent, maximum if not emergency power from the engines would have been expected. The position of the DASH actuators was not consistent with the use of such power. Mr Cable doubted whether the 91 per cent figure was accurate (Q 971) but he also explained how difficult it was to know the time to which the 100.5 per cent reading on the instrument panel related given the fact that there were at least three different impacts before the aircraft came to rest (Q 967).

The computed 91% RPM at first impact is for an undamaged helicopter in a steep transient climb, when rotor power requirements may exceed engine power availability by a significant margin. The rotors will then slow down in spite of any FADEC demand for more power, because the engines cannot supply it. If at this point the first impact causes severance of power to the forward rotor, as is suggested by the AAIB (5.10 Initial Impact to Final Impact), (sync shaft components were found near the first impact site), the power of both engines will immediately go to the aft rotor only, and this rotor would rapidly accelerate back to 100% RPM or more in response to the sudden doubling of input power from the engines. The rotor RPM sensor, installed in the combining transmission near the aft rotor, would relay this increased RPM signal to the cockpit instrument before subsequent impacts break the helicopter completely apart. Hence the computed rotor RPM of 91% at first impact is not incompatible with the cockpit display of 100.5% at the final impact site. Furthermore, if the FADEC controls continued to function for a moment after first impact, they would back off on engine power commands as the *aft rotor returned to 100% RPM, which would explain why the engines were found to be at an intermediate power setting (Q183), and not at full power, at the final impact site.*

Calculations show that excessive rotor coning does not become critical until very low rotor speed. At 50% RPM, the coning angle would be about 15 deg. At 90% RPM it is about 4.5 deg, which is only 1 deg greater than normal for the aircraft at the weight involved here. Hence the computed rotor RPM of 91% at first impact is nowhere near low enough to result in "the blades ..... coning up and meeting at the top due to the reduction in centrifugal force". A rotor RPM of 91% should not be, and was not, considered unusual in an emergency pullup scenario.

Regarding the statement on DASH actuator position not being consistent with the use of such power, please note that position of the DASH actuator is not a function of engine power levels. Note also that there is only one DASH actuator although it does have separate actuator motors on each end.

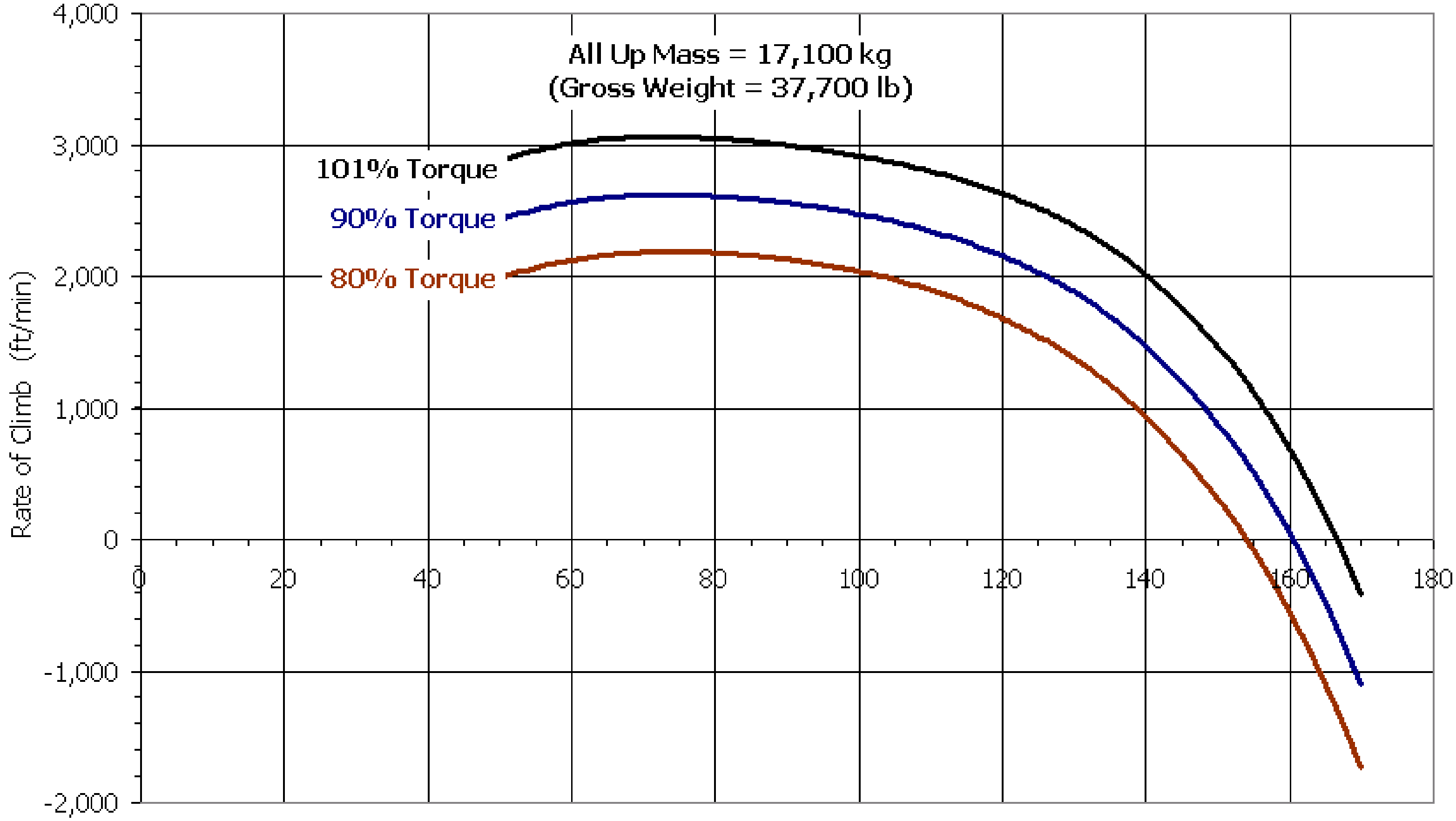
We now believe that prolonged or significant rotor droop did not occur based on the data extracted from the #2 DECU which was not available to us at the time the original simulation work was performed.

126. The groundspeed of 158 knots at impact derived from the Boeing simulation exceeded by 11 knots that of 147 found in the cockpit ground speed indicator (AAIB statement para 6). Moreover, the postulated ROC of 1000 feet per minute at 150 knots airspeed is unattainable. Squadron Leader Burke doubted whether it was achievable with ZD 576's load (Q 920). Witness A explained that while flying he had tried to see whether Boeing's chosen ROC was obtainable at 150 knots and had found that it was impossible in similar conditions (QQ 813-23). He had achieved no more than 400 feet per minute at 150 knots.

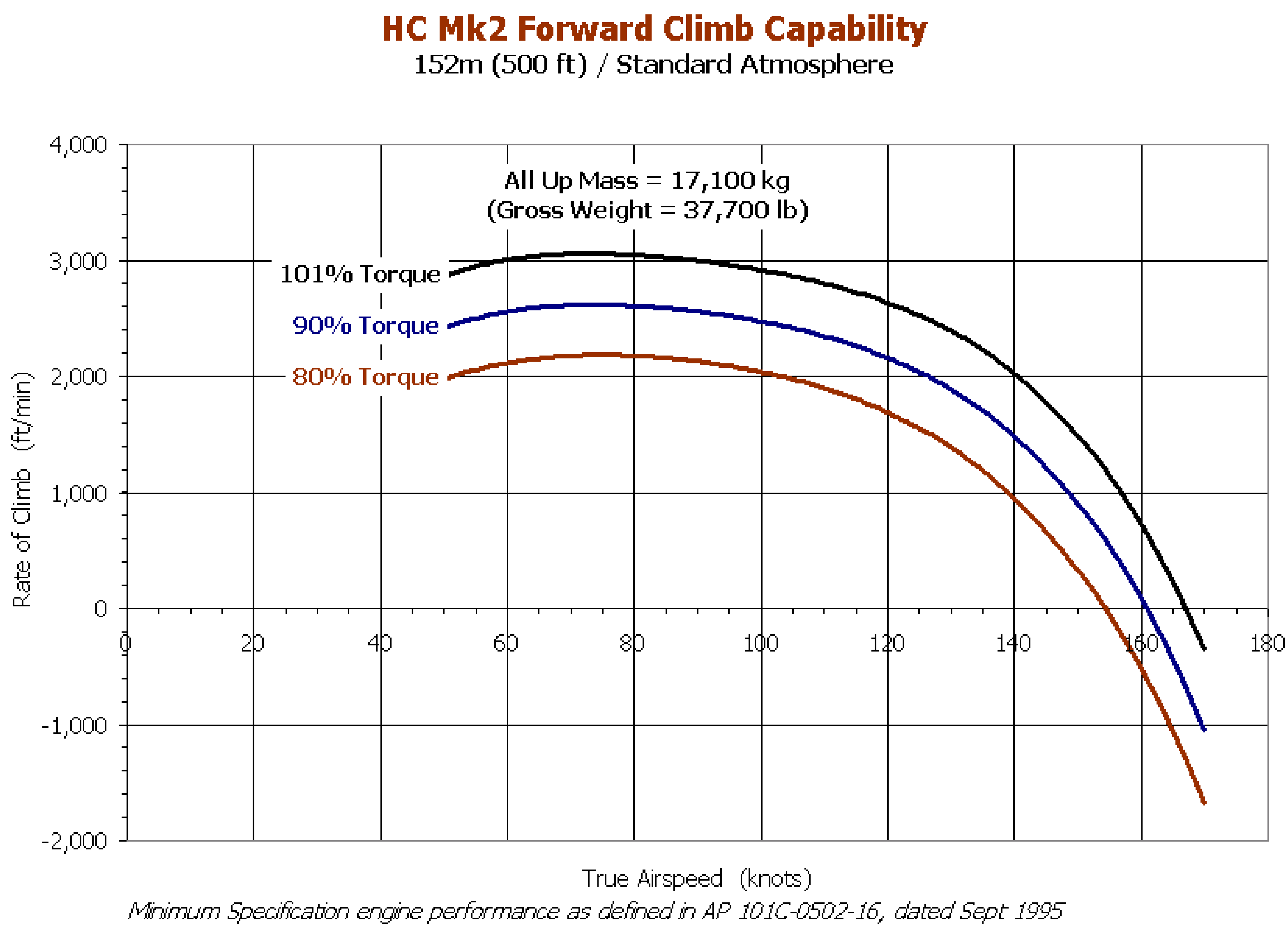
Some time may have elapsed between initial impact and the shutdown of the Groundspeed and Drift Indicator (GSDI). It may have continued to operate for a second or two after initial impact, recording slower airspeeds as the helicopter subsequently decelerated. Hence the 147 knot groundspeed found on the cockpit indicator at final impact is not incompatible with a 158 knot groundspeed at first impact.

The various references to rate of climb (ROC) vs. airspeed that the HC Mk2 supposedly cannot achieve are puzzling. Boeing Philadelphia's Aerodynamics Department, responsible for calculation and compilation of the performance data in the HC Mk2's Operators Manual have produced charts for ROC vs. Airspeed that show the following: For an HC Mk2 in the conditions that ZD576 was operating in at the time of the accident, ROC capability at 150 kts (TAS) should be approximately:
450fpm@Maximum Continuous Power (83% Torque);
1000fpm@Maximum (30 min) Power (92% Torque);
1500fpm@Drive System Limit (101% Torque);
This data is based on an aircraft with minimum performance engines. The performance charts (Pages 7 & 8) from which this data was taken have been reviewed by several Boeing test pilots who found them to be accurate. While we don't have flight test data to present here to confirm this, we are fairly confident that this performance is achievable. Due to the sensitivity of the ROC/TAS relationship as the aircraft approaches limiting conditions, variation in pilot technique, difficulty in selecting and maintaining limiting torque and gauge errors, collection of this data can be difficult in less that optimally controlled conditions. While testimony may seem credible, there is no substitute for properly instrumented aircraft in a controlled flight testing environment.

HC Mk2 Forward Climb Capability
Sea Level / Standard Atmosphere



Minimum Specification engine performance as defined in AP 101C-0502-16, dated Sept 1995


HC Mk2 Forward Climb Capability
152m (500 ft) / Standard Atmosphere
All Up Mass = 17,100 kg
(Gross Weight = 37,700 lb)
101% Torque
90% Torque
80% Torque
4,000
3,000
2,000
1,000
0
-1,000
-2,000
Rate of Climb (ft/min)
0
20
40
60
80
100
120
140
160
180
True Airspeed (knots)
Minimum Specification engine performance as defined in AP 101C-0502-16, dated Sept 1995

127. Sir John Day had arranged for someone to fly the Chinook simulator at 150 knots; they achieved a ROC of 650 feet per minute (Q 1075). He accepted that a ROC of 1000 feet per minute and a speed of 150 knots were not compatible (Q 1075). However, he put it to us that he had always said that the ROC was about 1000 feet per minute, not precisely that. This comment was made in the context of his own calculations, based on the range of heights shown by the TANS some 15-18 seconds before impact, which showed that between those times and the start of the final flare the range of possible ROC was between 650 and 1350 feet per minute. The simulator which produced a ROC of 650 feet per minute at 150 knots also produced one of 1150 feet per minute at 135 knots, which was a speed for which Boeing found it "very difficult or impossible" to match the predicted conditions with the initial impact data (AAIB statement para 8, QQ 1074-80).

See comment on ROC performance in 126 above.

128. The Boeing simulation postulations of a ROC of 1000 feet per minute and a speed of 150 knots were essential to the conclusion that a final flare was initiated some 4 seconds before impact. Now that those postulations have been shown to be unattainable, the circumstances and indeed existence of any such flare must be very doubtful. That there was such a flare was crucial to the Air Marshals' conclusion that the crew must have been in control of the aircraft for the last 4 seconds before impact (e.g. QQ 280, 1088). Sir John's calculations (above) give no support to such a conclusion, since they are independent of and in no sense a substitute for Boeing's postulations.

Since the Boeing simulation was a simulation it involves quite a few cases of postulation based on factual data, some estimates or approximations and various sets of initial conditions provided by the BoI/AAIB. It certainly seems postulation that any of this was essential to a conclusion that there was a "final flare" and that the "flare" was crucial to the conclusion that the aircraft was under control prior to impact. Here are a few things that were known about the aircraft just prior to impact:

- *The aircraft was climbing (post impact trajectory).*
- *The aircraft was carrying an appreciable airspeed (~135kts).*
- *The aircraft impacted in an upright attitude.*
- *The attitude of the aircraft at impact (pitched up) indicated a high probability that a pitch-up maneuver just prior to impact had been initiated.*

The simulation was not really necessary for the Board to reach reasoned conclusions about the final flight path of the aircraft. The simulation was intended merely to show that the aircraft could be commanded into its final impact attitude and that evidence from the wreckage was consistent. We believe that that was shown within limits. The simulation was not precise nor was it intended to be. Hindsight now may indicate another iteration and that has now been done. At the time, it was probably not deemed necessary by the Board.

The "postulations" of ROC=1000 fpm and Airspeed=150kts have not been shown to be unattainable. Testimony regarding aircraft performance can hardly be considered proof. The conclusion that a " final flare" was initiated prior to impact is based upon LCTA and DASH actuator positions combined with aircraft pitch and climb angle derived from impact evidence. The simulation merely shows that the aircraft could have been commanded into the attitude that it was in at impact.

129. Furthermore Mr. Cable explained that the Boeing simulation did not model FADEC. "It had to be a representation of a simple engine governor for each engine, which would have really quite different characteristics, I think, in small areas anyway, from the FADEC" (Q 957). The simulation presupposed that the aircraft was at all times under control and flying a straight course although there was no evidence that this was necessarily the case.

The Boeing computer program used in this analysis did not include a representation of the FADEC engine control system. Instead, the engine response was modeled by a simple first order gain and lag. However, the most important effects of FADEC were included by comparing the engine torque output computed by the simple first order system to an independently computed ideal FADEC response to a similar collective control input, and then adding in this difference "manually" during a second running of the same computed flight case. So, in effect, FADEC was modeled and is reflected in the results presented to the BoI in 1994.

The simulation was intended to explore possible maneuvers that the aircraft may have been put through prior to impact. Boeing was not asked to simulate how the aircraft might have behaved with any sort of degraded flight control capability. This could possibly have been because there was no evidence the aircraft was not under the control of the aircrew.

130. Mr Perks, who had worked on such simulations with a MoD team in the late 1970s and early 1980s, explained that for a given transient manoeuvre all the key modeled parameters had to be matched within reason to actual historical records. Two of the most important parameters were rotor speed and torque from the engines, in relation to which no historical records were available. He remarked on the disparity between the rotor speed required for the simulation and that indicated on the rotor speed gauge; and also on the fact that, whereas the simulation manoeuvre required engine power to be at absolute maximum, the indications found by the AAIB were that the engines were at an intermediate power setting (Q 183). Furthermore, none of the witnesses on the Mull who had heard the aircraft had noticed any noise suggesting a violent manoeuvre, and there were no data to suggest that the engines had exceeded normal values.

It is suggested that Boeing Philadelphia's Flying Qualities group is quite experienced with flight simulations.

The B-29 modeling program used by Boeing is based on historical data representing the performance capabilities of the Chinook including transient analysis of maneuvers. While rotor speed and torque from the engines may be considered to be key parameters, another important element associated with each of those parameters is time. It is helpful to know when each of those parameters is sampled and under what conditions. As the AAIB attempted to explain, in a multiple impact accident, it cannot always be established with certainty when a particular event occurred and not at all uncommon to find different data from different pieces of equipment depending on when during the sequence it lost power.

In a transient maneuver it is expected that parameters will change and subsequently return to normal. In this particular case it would be expected to get some droop in rotor speed and that the engines would respond to the increased demand for power with increased torque output. We do not know that the reading of 100.5% Nr corresponds to the value at initial impact. We now have a pretty good idea from the #2 DECU that sustained (more than a second) emergency power/torque was not demanded or it would have been recorded as an exceedance. When the simulation was originally run the 91% Nr and the resulting demand for power did not seem unreasonable.

Regarding the lack of noise suggesting a violent maneuver, Boeing would like to suggest that a Chinook impacting terrain at high speed is a relatively violent maneuver and that quite possibly the witnesses on the Mull may not have been able to distinguish discreet auditory elements of the pre-impact through post-impact sequence.

131. Mr Perks proceeded, "On the Chinook Mk 2 aircraft the engine control systems have aircraft rotor speed as a primary input, with collective pitch as a supplementary input. If the rotor speed is too high, the engine is driven to idle power. If too low, the engine is driven to maximum output

power. If collective pitch is changed, the engines will also be affected. Any form of extreme manoeuvring would have forced the engine control systems to respond immediately. The engine controls should, therefore, have been anywhere other than at normal settings. Normal settings implies the engine controls were not seeing major changes in their inputs, and that is not consistent with the violent manoeuvring postulated by Boeing. Whatever the pilots were doing, collective pitch was not being affected, and neither was rotor speed, given the evidence in the wreckage." Thereafter he expressed the belief that the simulation should be "discounted" as evidence.

This statement makes the assumption that all the instruments and engines "froze" simultaneously during the transient condition. Refer to the comments on (125) above regarding the compatibility between rotor RPM at initial impact vs. indicator readings found at the final impact site, and also for statements regarding the condition of engine controls at first impact vs. their condition at the final impact site.

What could have been noted regarding emergency power was the data downloaded from #2 engine DECU which shows no evidence of torque or temperature exceedance that would indicate a sustained emergency power demand from that engine. Since all evidence indicated that the engines were power matched at impact, following impact and during rundown, there is a high probability that the #1 DECU would have provided similar data. In addition, all the engine and DECU data taken together suggests that there had been no FADEC anomaly or technical fault distracting the crew prior to impact.

132. Where does this leave the simulation? We conclude that it would be quite inappropriate to treat the results of the simulation as proven fact.

Boeing agrees with this and contends that the simulation was never represented as being anything other than a simulation. It was never purported to be a reconstruction of the accident. The simulation was simply another tool at the investigator's disposal. The simulation was a "sanity check" on the other evidence collected from the accident scene. That the simulation should be "discounted" is inappropriate, since it was never represented as being more than what it was – another source of data for the Board.

135. During the course of his evidence Sir John on more than one occasion emphasised that his conclusions were based on fact and not on hypotheses. It is therefore appropriate to look at some of the matters which he treated as fact. (Page references are to HL Paper 25(i).)

(a) "We know that about 20 seconds before impact with the ground the crew made a way point change" (Q 280, p 118 col 1). This figure which derives from the Racal report on the SuperTANS is based on a power down speed of 150 knots and a straight course from the WP change to impact at that speed. It is therefore at best an estimate and not a fact since the only factual evidence of speed at or after the change is the indication from the ground speed and drift indicator of 147 knots at initial impact (AAIB report, paragraph 7).

The RACAL Report on the RNS252 SuperTANS Navigation System provided steering data calculated once per second that prior to power loss that recorded time and range to waypoint B of 32.8 minutes/86.63nm. Aircraft groundspeed just prior to impact can be calculated from this and yields 158.5 knots (or 162.6 knots from the display at power down). We would conclude that the RNS252 data should be considered to be another, possibly more, accurate indication of speed just prior to impact.

(b) "We know for a fact … that some four seconds before impact the crew started to flare the aircraft" (Q 280, p 117 col 1; Q 1088). Not so. The Boeing simulation, using assumptions now shown to be incompatible, produced this result. On no view could it be described as fact and there is no evidence either way as to what caused the aircraft to impact the ground in the position described in the AAIB report.

The pitched-up attitude at impact resulted from either a set of control inputs by the crew or some unexplained phenomenon. The simulation demonstrated that the aircraft could have been commanded to the impact point through controlled flight and that no actuator or control positions were incompatible with this postulation.

(c) "They had chosen to fly straight over the Mull of Kintyre, and we know that because they had set up this 1000 feet a minute ROC" (Q 301). There is no evidence that they had chosen to overfly the Mull, and indeed the making of the waypoint change suggests the contrary. Furthermore the 1000 feet a minute ROC derives entirely from the Boeing simulation with all its deficiencies referred to above.

Equally, there is no evidence that the aircrew chose to turn away from the Mull and not overfly it.

(d) "What is for sure is that they were in a 1000 a minute cruise climb in that last 20 seconds before the final four seconds of flare" (Q 304). This is far from being sure given the deficiencies in the simulation already referred to.

Boeing agrees that an exact rate and duration of climb cannot be inferred from the simulation work that it performed; it disagrees with the characterization of the simulation as "deficient". The simulation results provided to the BoI and AAIB were never misrepresented as being a "reconstruction". What the results of our simulation showed was that, for several of the cases examined, at certain rates of climb and airspeeds, a normally functioning HC Mk2 could be maneuvered to a state where impact with terrain would simultaneously result in the criteria provided by the BoI and the AAIB.

Regarding deficiencies in the Boeing simulation; the controversy over rate of climb versus airspeed prompted a thorough review of the B-29 model and how it represents LCTA position versus airspeed. We discovered that the model used a nominal value LCT schedule for the forward and aft rotors rather than a more realistic range schedule. The effect of this on our modeling was to skew our results to the high side on airspeed and to exclude lower (135kt) possible airspeeds. Our latest simulation tools reflect the broader LCT schedules.

(e) "We know they did not pull emergency power" (Q 311). Sir John later agreed that the impact could have destroyed any evidence of emergency power being pulled (Q 1097).

As the AAIB has described, the demand for emergency power would be indicated by evidence from the Emergency Power Panel. The telltale indicators on that panel are magnetically latched after five seconds of sustained demand for power at or exceeding the limits for emergency power and would not be expected to remain latched in a relatively high speed multiple impact accident. Another indication would have been from the counters which would only advance to the next minute after accumulating at least a minute following indicator latching.

Another reliable piece of evidence that emergency power was not demanded from the engines was the #2 engine DECU. Data downloaded from this unit shows no evidence of torque or temperature exceedance that would indicate a sustained (~1 sec or more) emergency power demand from that engine.

Part 6: Conclusions

148. We consider that Sir John's conclusions on this matter must be weakened by his reliance on matters which he treated as facts but which have been demonstrated to our satisfaction to be not facts but merely hypotheses or assumptions. It must be a matter of speculation what would have been the Air Marshals' conclusion if the Boeing simulation had not been available, or if its deficiencies had been identified.

Boeing fails to see that the results of Boeing's simulation are essential to the overall conclusions of the Board. The BoI already knew from other data that:

- *The aircraft was climbing (avg. R.O.C. of approximately 700 fpm from RNS252 data).*
- *The aircraft was carrying an appreciable airspeed (~135 kts. from RNS252 data; G.S.-Wind)*
- *The aircraft impacted in an upright attitude.*
- *The attitude of the aircraft at impact (pitched up) indicated a pitch-up maneuver consistent with last second terrain avoidance.*

With this information already available, the Boeing simulation only provides additional information—some of it possibly contradictory since we did not have the SuperTANS data given above and other information available to us at the time. The Boeing simulation was not necessary to the conclusion reached by the BoI.

150. Sir William appeared to accept that if, having visibility of 1000 metres, the crew had altered course at the way point change and flown maintaining visual contact with the coast, they would have been "perfectly entitled" to do this (QQ 355, 394, 1039). However on his second appearance he rather departed from this view and explained that, if the aircrew had 1000m visibility, they would have seen that they were displaced from their planned track some 8 or 9 seconds before they made the way point change, and should therefore have altered course earlier (Q 1059). Both Air Marshals attached importance to the results of the simulation, and in particular to the high speed at which the aircraft was assumed to be traveling at or before the waypoint change, an importance which must now be considered doubtful given the deficiencies already referred to in the simulation. In any event, even if the aircraft was traveling at the assumed high speed at or before the waypoint change, no reason has been suggested as to why speed could not have been reduced in making any subsequent turn, thereby reducing its radius.

162. The Boeing simulation was prayed in aid to fill in some of the foregoing gaps but as already described it can only determine what could have happened rather than what did happen and was itself deficient in the following respects, namely:

(i) it did not take account of FADEC,

As previously described, while the model did not incorporate a FADEC module, the overall simulation did take FADEC accurately into account.

(ii) it postulated a combined speed and ROC which have been found by Witness A and Sir John Day to be unattainable,

Boeing believes the modeled speed and ROC combination to be attainable.

(iii) it also produced a rotor speed of 91 per cent which was a fairly extreme position differing considerably from that found on the instrument panel and of whose accuracy Mr Cable had doubt,

The indications from the instruments can be highly suspect for reasons already given here. If Boeing had been aware at the time the original simulation work was done that there was *evidence that the engines had not been commanded to emergency power we would have, in coordination with the BoI/AAIB, constrained Nr accordingly.*

(iv) it produced a groundspeed during the final manoeuvre of 158 knots which exceeded by 11 knots the speed of 147 found in the ground speed indicator, and

As previously mentioned, there is nothing surprising about this in a multiple impact accident scenario. Additionally, since Boeing produced a simulation that attempted to converge on a solution rather than a reconstruction that exactly matched a set of historical data, this disparity is relative in nature. Both are relatively high groundspeeds and give a good idea of the range of speed of the aircraft around the time of initial impact.

In addition, data extracted from the RACAL RNS252 SuperTANS indicates that groundspeed just prior to impact was around 158-163 knots.

(v) it hypothesised a final manoeuvre initiated by the crew some 4 seconds before impact, and that prior thereto the aircraft had been under control on a steady course and speed.

It did not hypothesize a final maneuver; it describes possible maneuvers that resulted in the aircraft being in the attitude and configuration that it actually was in at impact or immediately thereafter. While the control inputs, speeds and rates may be hypothetical, the final condition of the aircraft derived from the impact area and wreckage are not. Once again, the simulation was a tool to help understand what was possible for an HC Mk2 to perform under given conditions.

163. Both Air Marshals accepted as a matter of fact that the aircraft was under control when the waypoint change was made and at the moment 4 seconds before impact when the simulation assumed that the final flare was initiated. So far as the way point change is concerned we accept that it is highly unlikely that the crew would have made a way point change if they had thought that they were not in control, but it is possible that if some loose article had jammed the controls during steady flight this would not manifest itself until the controls were moved in order to alter course. Squadron Leader Burke referred to his experience of test flying with control and engine malfunctions when after a period of steady flight dormant faults can appear when a manoeuvre is initiated or engine speed is reduced or increased (Q 705). There is no evidence that such was the case here but equally no evidence that it was not. Alternatively, the movement of the controls to alter course could have precipitated a jam.

The probability of a control jam was something that appeared to have been thoroughly explored by the BoI. The fact that the flight control pallet insert debonding issue was very much in mind by the entire Chinook community at the time ensures that it would have received adequate attention as a causal element. In considering a flight control jam, the following logic may have been applied:

For an upper control jam to have occurred, there would have been no dormancy or latency possible, it would have immediately become noticeable to the crew since the AFCS is constantly providing stabilization input to the upper controls. This would typically manifest itself as a dual-axis "fall-off" or loss of control.

For a lower controls jam, the possibility for latency is greater as is the probability that only one axis of control would be affected since the lower control runs for all four axes are physically separate. It has been postulated that a jam of this sort could go unnoticed until control inputs are made to initiate a course change. If this were the case, and the crew knew where they were, would they not have turned away from the high ground, if the jam was in *the thrust channel, or pulled emergency power (not done) to attempt to clear the high ground if the jam was in the roll, pitch or yaw channel?*

For there to have been a multi-axis lower control jam that renders the aircraft incapable of turning or climbing at its maximum emergency capability after the waypoint change and then subsequently clears to allow a last second pitch-up maneuver is considered to be extremely improbable.

164. So far as the aircraft being under control at the moment four seconds before impact is concerned, we do not consider that there is evidence to justify such a conclusion to the required standard of proof. Indeed, apart from the simulation, such evidence as there is - to which reference will shortly be made - suggests the contrary.

Boeing has never been made aware of any evidence that ZD576 was not under control just prior to impact. There are, however, a couple of pieces of information that would suggest that the aircraft was under control:

- *Of all of the possible attitudes in roll, pitch and yaw that the aircraft could have been in if it were out of control, the attitude of the aircraft at initial impact is upright.*
- *The attitude of the aircraft at impact was consistent with the controlled reaction by a pilot to the sudden appearance of rising terrain.*

The fact that, over the life cycle of a fleet of aircraft (~900,000 flight hours for the Chinook), a small number of flight control anomalies have been reported hardly constitutes evidence that ZD-576 was uncontrollable for a short period of time just prior to impact.

167. The AAIB were not able to exclude the possibility of a control jam given the level of system damage. Nor could they exclude the possibility of pre-impact detachment of the thrust balance spring attachment bracket and other inserts. It will be remembered that this bracket had some three weeks previously detached from the aircraft's thrust/yaw control pallet (see above, para 56). The AAIB were unable to assess the functionality of number 1 DECU owing to gross fire damage. Metallic contamination of the hydraulic system of the integrated lower control actuators found by the AAIB was thought to have been present pre-impact but not to have contributed to the accident; however, the subsequent experience of the US Army and their recommendations (see para 104 above) suggest that such contamination could cause disturbance in the normal operation of those components at the time. DASH runaways have caused temporary loss of control problems as Squadron Leader Burke explained, and UFCMs and false engine failure captions have also afflicted Chinook Mk 2s. Mr Cable accepted that it was possible that there had been an intermittent engine fault which had subsequently reverted to normal before the impact. The problems arising from the newly installed FADEC system had not all been resolved by June 1994; and the Boeing simulation has been shown to have relied to some extent on postulations which are impossible in performance and parameters some of which do not fit with what was found by the AAIB. Can it in these circumstances be said that there is absolutely no doubt whatsoever that it was the voluntary action of the aircrew - including not only both pilots but also MALM Forbes who in our view was probably assisting with the navigation - which caused the aircraft to fly into the hill?

One of the purposes for doing the simulation work was to see whether the aircraft could be commanded into its impact attitude and whether there was correlation with other evidence from the wreckage and impact site. If that had not been possible it might have suggested some anomaly or other avenue for investigation.

While there was no positive evidence to exclude pre-impact detachment of pallet mounted components in the flight control closet, it was well known to UK Chinook community and to the crew and maintainers of ZD576. The fact that one of the pallets had just been replaced on that aircraft suggests that that area of the aircraft would probably have been receiving adequate inspection attention, if not additional scrutiny.

Nothing in the extensive investigation of the engines and FADEC systems suggested any sort of anomaly and since the engines were at matched power settings at rundown and the #2 DECU showed no significant faults, in all probability the engines were functioning normally.

Metallic slivers found in one screen, of one of the hydraulic systems, of one of the integrated lower control actuators, were found and determined to have been present pre-impact. The AAIB determined that there was no evidence that this had contributed to the accident. Indeed, since it is impossible to totally prevent a certain amount of contamination in any hydraulic system they are equipped with filters and screens.

The reference to subsequent U.S. Army experience regarding uncommanded flight control movements and hydraulic system contamination as being related to ZD576 is not applicable. The aircraft in the referenced report (USASC 97-305) described a CH-47D that experienced an uncommanded departure from controlled flight. The report disclosed significant hydraulic system contamination but "did not discover a single contributing factor" to this incident. The supplement to this report, released six months later (USASC 97-305 Supplement), focused on the significant levels of Barium, Chlorine and Zinc in the aircraft's hydraulic systems (both flight control hydraulic systems were found to contain 20% water contamination). This caused the U.S. Army to evaluate commercial practices and adopt filter changes and the use of hydraulic system purification equipment. The aircraft in the referenced report had 4606 flight hours on it; ZD576 had 57 flight hours since delivery, hardly a fair comparison.

Speculation on a number of possible alternative, but relatively improbable, explanations for what might have happened to ZD-576 pales when compared to the classic CFIT (controlled flight into terrain) causal chain established by the BoI.

175. We consider it appropriate to identify those matters to which we have had regard which were not before the Air Marshals when they considered the investigating board's report:

(a) the more detailed evidence of Mr Holbrook as to the weather conditions at sea, and the probability that the crew would have seen the land mass from some distance offshore;

(b) the evidence of Mr Perks, Witness A and Squadron Leader Burke;

(c) the deficiencies in the Boeing simulation with particular reference to the facts that

(i) it did not take account of FADEC and

In fact, the Boeing simulation did, the model didn't.

(ii) it used a postulated speed and ROC which have been shown to be incompatible; and

Please see previous Boeing comments regarding speed versus ROC. Boeing was provided speed and ROC data as input.

(d) the possible effect of contamination in the hydraulic fluid in the integrated lower control actuators, as referred to in the US Army report of June 1997.

As previously stated, the gross level of contamination discovered in the hydraulic systems of a 13 year old helicopter with over 4,600 flight hours on it can hardly be considered representative of one with a new hydraulic system and 57 flight hours.

THURSDAY 27 SEPTEMBER 2001
MR KEN SMART, MR TONY CABLE AND MR REX PARKINSON

Chairman

231. You refer at pages 59 to 61 (8.) to the Boeing simulation and indeed I think a good deal of reliance has been placed on Boeing's conclusion firstly as to the speed of the aircraft at the time of impact. Have you any views on how accurate the Boeing simulation was likely to be?

(Mr Cable) Just as regards the speed, if I may say so, my Lord Chairman, there are other indications, not of an accurate speed but certainly from the crash site the distance the aircraft has travelled uphill, just immediately looking at the site, suggests a fairly high speed.

232. You mean that is after break-up, is it?

(Mr Cable) After the initial impact.

233. Yes, when the thing began to break up.

(Mr Cable) Yes. It has travelled about 200 metres over the ground and about 90 feet vertical climb, I think, before the final impact, so that of itself suggests—it is difficult to put a number on these things—clearly a fairly high ground speed at that point. My initial assessment I believe was well over 100 knots.

234. That was before you got Boeing's results?

(Mr Cable) Oh, certainly, yes. There is also the hover meter, I think it is, ground speed indication which was a fairly reliable indication of I believe 147 knots ground speed, so as regards speed I think there are some other pointers that as it happens fit fairly well with what the Boeing simulation was saying.

235. The hover meter was intact, was it, when it was found?

(Mr Cable) Yes. I am using the wrong term. It is the drift and ground speed indicator I think. It was not intact, no, but there were some good witness marks on it, what I felt were reliable witness marks.

236. Indicating speed?

(Mr Cable) A hundred and forty seven, yes. As regards the Boeing simulation, this was really a joint effort between the Board of Inquiry and myself. I believe the Board tasked or asked Boeing to do this and I did not get greatly involved apart from supplying what I had assessed as the impact parameters and the dash and longitudinal cyclic trim actuator extensions, so I provided what I felt was the pitch attitude, the flight path angle, the dash and the LCTA extensions, and a couple of other things that came from the site. The Board asked Boeing Helicopters to do this assessment. I did get involved partway through because I was in Philadelphia and I went and discussed it with the guys who were doing this. My overall feeling was yes, it was probably quite a reliable conclusion that they had come to. I do not think it was 100 per cent but then very few things in this business are. My judgement was that it was probably fairly reliable.

Lord Tombs

237. But would you agree that a simulation is not an account of what happened? It is saying that it could have happened and the parameters would have allowed it to happen. It is not proof that something did happen. It is saying that something fits within the parameters.

(Mr Cable) Correctly it is a maths model. That is correct. They did try a large number of initial conditions and, as my statement says, could not get near the final initial impact conditions other than with this fairly tight set of conditions before the manoeuvre and in this sort of manoeuvre.

238. It shows that it was possible.

(Mr Cable) Yes. As I say, they could not achieve the final result—

239. Easily?

(Mr Cable) No. They could not achieve it without this sort of manoeuvre and the initial conditions that I have put down here. Yes, I had fairly high confidence that that was indicating what had gone on.

240. I do not think that is the case really if I may say so. I think that any simulation tells you that within the parameters you set, something is possible or not. There is no historic data fed into it. It is a "what if" situation.

(Mr Cable) Yes. It was a maths model of the aircraft and the only way that they could achieve the final conditions was…

241. The final conditions and the input conditions, the beginning and the end ones, are assumed.

(Mr Cable) Yes.

WEDNESDAY 7 NOVEMBER 2001
MR TONY CABLE AND MR KEN SMART

Chairman

950. Did you give the instructions to Boeing as to what was needed, or did that come from the RAF, or was it a combination of both?

(Mr Cable) The two Boeing accident investigators remained for quite a considerable period of time, and I clearly discussed it with them. On, I think it was, 15 August, I sent Boeing Philadelphia a fax, the intention of which was to define for them the initial parameters that I had established from the accident site and wreckage examination, which were the aircraft pitch attitude, flight path angle, DASH actuator extension, LCTA extension and ground speed. I also, in that fax, included some other basic parameters, which I obtained from the Board of Inquiry, such as aircraft weight and sea level pressure, temperature, and so forth, which were needed for the study, but I was acting as a messenger on those issues. And it was agreed all round that it was better to have one document which defined the whole set of parameters that were needed for Boeing to do the mathematical model study, rather than having a number of different documents.

Lord Tombs

951. May I just follow this up a little, this Boeing mathematical model, Mr Cable. When we saw you previously, you said, "The Board asked Boeing Helicopters to do this assessment. I did get involved part way through because I was in Philadelphia and I went and discussed it with the guys who were doing it." Now we did ask the Ministry of Defence for the specification for the study, and they replied "The Board did not give Boeing Helicopters any written specification but rather commissioned the work through Tony Cable, AAIB". And they subsequently say, "I subsequently contacted the AAIB, who confirmed that Mr Cable recalls that much of the early discussion on the simulation runs was conducted verbally between himself and the Board and Boeing." Now how deeply involved were you in the simulation specification requirements and objectives, because those two things do not quite match; but, of course, it was seven years ago, so perhaps they would not?

(Mr Cable) I do not know how to answer that question really. There were three-way discussions between myself, the Boeing representatives who were present and the Board of Inquiry; the details of those I cannot remember. The outcome was, it was quite clear to everybody, all those three parties, that, as I say, it was a reasonably complicated issue to establish how the actuator extensions could have got to the point they did, and that the best way of approaching this would be the math model that Boeing had already set up for the aircraft flight controls and characteristics.

952. We will get on to that later. So you were involved throughout the study?

(Mr Cable) I had some involvement in that process. As I say, I passed on the parameters I had established and forwarded on those that the Board gave me, and in the latter half of August 1994 I was in Philadelphia conducting the strip examination of the flight control actuators, and I took the chance then to discuss with the appropriate group the progress of the math modelling, and that was the Flight Qualities Group. The discussions were really for me to try to understand something about the model; the intention certainly was not for me to try to do a full validation or verification of the model. In a lot of these instances, one needs to trust a manufacturer, to some extent; although clearly to be wary that they may have an agenda. I believe this is usually apparent, if this is the case. I did not feel there was any of this, in this case, that the people I was working with, the people on the ground, were taking an objective viewpoint and trying to find out answers.

953. I was trying to find out whether there was a specification for the work; and, insofar as your letter gave three rates of climb, for two speeds, and you apologised for the fact that provided six studies, in that sense, you were the specifier?

(Mr Cable) As I say, I was passing on information. Those cases were agreed, I am sure, in discussion between myself and the Boeing reps and the Board of Inquiry, as I recall. I cannot remember the detailed discussions.

954. It does not seem as though anybody was in control, does it; it just happened?

(Mr Cable) I do not think it needed a great deal of control.

Lord Tombs: If one wanted a simulation done, one would normally say what the starting-point was and what the objective was, and you do not expect mathematicians to do a study without being told what it is they are to study. So somebody was in control; it seems to have been you. I do not want to pursue this further, Chairman.

The simulation was performed, not by mathematicians, but by engineers who are specialized in the performance and handling qualities of the Chinook. The simulation was performed using a "math model" – essentially the same sort of math model that would be used to provide an aircraft's flight characteristics in a flight simulator. Boeing was asked to perform the simulation by the BoI and our *liaison to the Board was the AAIB. Boeing works through the AAIB in (RAF) accident investigations because they are typically the technical focal for the investigating authority.*

The purpose of the simulation, by our recollection, was to try to determine whether a normally functioning helicopter could be commanded by a set of control inputs into an attitude at impact consistent with a set of end data provided by the BoI and the AAIB from known or deduced conditions at impact, all of which were provided in the our report.

Chairman

955. You heard a moment ago, no doubt, Mr Cable, Squadron Leader Burke pointing out the disparity in the rotor speeds, (a) as found by you, from the rotor speed indicator, and (b) that which was assumed, or used, by Boeing in their simulation; and we see that if we look at your report. You see in paragraph 8 is the Boeing assumption, 204 revs, 91 per cent, and your figure is in paragraph 7.2.2, and you refer to the Triple Tachometer, 100.5 per cent, and you give it a weighting 2, which is `evidence appears positive'.

(Mr Cable) Yes, my Lord.

956. Would you accept that they cannot both be accurate? It must be one or other of these, or, possibly, I suppose, even both, according to what was damaged, but one or other is likely to be incorrect?

(Mr Cable) Possibly. I think it needs to be remembered, throughout the information that I present in the statement, that this was a complicated accident, and, in particular, from my point of view, it was a multiple impact accident, so there were certainly three major impacts, depending on how you count them, three or four, over a period of some seconds. I do mention in the section on Flight Deck Indications that it is generally impossible, from the rest of the evidence, to say where signal supplies were lost, so when there is a transducer producing a signal on one part of the aircraft the signal passes by wire to the particular gauge, so at some point in the sequence that wire is broken. There is another wire coming from a power supply area to supply power to the indicator; at some point the power goes off, or the power wire gets broken, and at some stage the indicator may receive an impact which leaves marks on it. It was not possible, from the evidence that I could find at the accident site, to say what that sequence was. I thought that the electrical power probably went off fairly shortly after initial impact. I could not say, with an impact on the lower rear part of the aircraft, as the initial impact, that the power went off immediately, so there was time for indications to change before the impact that then caused a mark on a particular gauge, and this, of course, could be different for the different indicators. It was an unsatisfactory situation, but that was the evidence that was available. And throughout this investigation the evidence was remarkably thin, from my point of view, I must say. We spent a great deal of time trying to find evidence.

The 91% rotor RPM is not an assumption or starting point, but is a calculated result arising from the rapid increase in rotor torque required during a particular pull-up maneuver at this speed, which the engines cannot supply, resulting in a loss of rotor speed. Both the 91% and 100.5% may be valid for rotor RPM at slightly different points in time, as explained in the comments on (125) above.

As previously stated: We now believe that prolonged or significant rotor droop did not occur based on the data extracted from the #2 DECU which was not available to us at the time the original simulation work was performed.

957. Are you suggesting then that the rotor speed may have altered between the initial impact and the time when, to put the matter crudely, the gauge packed up, on the second or third impact?

(Mr Cable) That is one possibility, my Lord. The other aspect that I recall was that the Boeing model did not accurately model the FADEC, so it had to be a representation of a simple engine governor for each engine, which would have really quite different characteristics, I think, in small areas anyway, from the FADEC, which may also be a factor in this. It is something which is going to exist in all models and simulators, they are never going to be fully representative, and they will be representative to a greater or lesser extent. I was fairly satisfied with the involvement I had, talking to the Flight Qualities people at Boeing, that, for the purposes that the model was being used, which was looking really for fairly gross manoeuvres over a pretty short period of time, the limitations I heard about were probably not going to make a tremendous difference.

For all practical purposes the influence of FADEC was adequately modeled in the Boeing simulation.

Lord Tombs

958. Do you mean that the difference between 91 per cent and 100 per cent is not significant?

(Mr Cable) I am not sure, to be honest. I do not know enough about it.

959. It is a 10 per cent difference, is it not?

(Mr Cable) Yes, but I do not know at what stage the 91 per cent has occurred.

960. It is true, the simulation, you said, was a good one?

(Mr Cable) It is the number, 204 rpm, I think it was, that Boeing had at the end of their 2.9 seconds.

961. Yes, that is right.

(Mr Cable) Now what the time history is of the rotor rpm over that period I do not know; whether the rotor rpm has dropped off a lot at the end, I cannot say.

962. I am not clear whether you are saying that the 91 per cent is right or wrong. You say that the Boeing simulation, or modelling, did not, you think, or you are not sure it is including the FADEC performance?

(Mr Cable) It did not include FADEC performance.

963. That is a fairly serious thing, is it not, because that is the whole point of the Chinook II, that it has an automatic engine power control?

(Mr Cable) It is important in some aspects. I think, for the modelling of the flight characteristics of the aircraft, no, it probably is not, in this sort of manoeuvre.

964. I find that very unconvincing, Mr Cable, frankly.

(Mr Cable) I am not an expert on the model. I can only tell you what I know about it, my Lord.

965. Are you really saying that we should disregard the aircraft evidence that you found and thought was fairly reliable and say that the Boeing figure was more likely?

(Mr Cable) I say, in the statement, that I considered that, I forget the words but, reasonably reliable, which is a judgement, based—

966. Evidence found appears positive? (Mr Cable) Yes. It is a judgement based on best evidence.

967. But you are now suggesting that you have abandoned that in view of the Boeing figure; is that right? (Mr Cable) No, I am not abandoning it. I make a statement, before listing those indications there, that this is a multiple impact situation, and it is not possible, from the evidence I have, to say when these events occurred.

968. Well, can I repeat the question. Which figure do you think is the right one, if either?

(Mr Cable) I do not know. I think you have to accept, with all the evidence I present, that this is really not an exact science, it is a judgement based on best evidence, which in this case was really quite thin. And there is a difficult balance to achieve, particularly on the flight deck indications, between saying, no, that evidence does not look 100 per cent, therefore I will not mention it, and saying, that is what the indication was, therefore it must be right. Each of these things is quite a complex matter.

969. Yes, I have a great deal of sympathy with you.

(Mr Cable) And the best one can do is to make a best judgement based on professional experience.

970. I have a great deal of sympathy with you; this was a dreadful crash and a lot of damage and very difficult to judge what, in fact, happened. But what I am afraid is that people will read too much into the comments in your report. I can understand that you will make the best judgment you can and cannot guarantee it.

(Mr Cable) This was a report made to the Board of Inquiry for the purposes of the Board of Inquiry, with fairly careful wording, as far as I could, to try to give a balanced view of what I found.

971. So, can I just summarise, you do not back either the 100.5 or the 91, you do not know?

(Mr Cable) I doubt if the 91 per cent is accurate, because, as I say, the FADEC was not modeled on the model. I had a fair degree of confidence in Boeing, who understood fully what the purpose of the math modeling was, that if this had been a major factor which grossly affected the results they would have known about it; they knew about the model, they had been using it for many years.

The Boeing computer program used in this analysis did not include a representation of the FADEC engine control system. Instead, the engine response was modeled by a simple first order gain and lag. However, the most important effects of FADEC were included by comparing the engine torque output computed by the simple first order system to an independently computed ideal FADEC response to a similar collective control input, and then adding in this difference "manually" during a second running of the same computed flight case. So, in effect, FADEC was modeled and is reflected in the results presented to the BoI in 1994.

972. Okay, I think it is pretty unsatisfactory, but that is my point of view.

(Mr Smart) My Lord Chairman, I wonder if I could perhaps intervene here and perhaps help the Committee.

Chairman

973. Yes, please do.

(Mr Smart) Listening to the evidence given by a number of witnesses now, and seeing some of the answers to the questions that you have posed, it is clear to me that some parties here have chosen to make more of the evidence presented in Mr Cable's report than he would himself. He has carefully qualified the degree of accuracy of the parameters that he has presented here. And it is clear, and it is the same on many accidents, without the flight recorder data that we would perhaps normally find on more modern aircraft, and even would expect to find on an aircraft of this generation, that you cannot be sure; particularly in the impact like this, which took place over a period of time. This is not one single event which has stopped everything and the systems damage that we see would have been representative of what was happening in the aircraft at that particular time of impact, this is an impact, or a series of impacts which have taken place over a period of time, which presents evidence in a scrambled form. So Mr Cable has made that clear in his report, I hope, and I think I would advise caution to anybody to take one particular figure out of that and say is this accurate or not; the chances are that you are going to draw the wrong conclusion, if you try that sort of approach.

Lord Tombs

974. I sympathise with that view, Mr Smart, but the question I was asking you was whether this indication, which is categorised as 'appears possible', was to be accepted, or the simulation which was described as `very convincing'? We have two figures, with a reasonable provenance, which are incompatible, and I was seeking for which one we should—

(Mr Smart) But we do not know which period of time, and how, that occurred, to be the case; that was the issue here. It is a complicated set of circumstances.

Lord Brennan

975. May I ask you, should we exercise a similar degree of caution about those who draw from this report the conclusion that there is no evidence of malfunction? (Mr Smart) I would suggest, in that way, my Lord, that, yes, you should. Again, Mr Cable made that point in his report. Lord Brennan: Yes, he did, twice.

Lord Tombs

976. In the report? (Mr Smart) In the report, yes.

Chairman

977. Are you able to comment, Mr Cable, on the Boeing conclusion that the aircraft was climbing at a rate of 1,000 feet per minute, at an air speed of 150 knots; can you comment on whether that is, in fact, a possibility, or not?

(Mr Cable) I do not think I can, my Lord. That is an aircraft performance question that . . .

978. It is really outwith your field?

(Mr Cable) That I have not been into and I am not qualified to talk about. It was something which, as I understand it, the Boeing math model said the aircraft could do.

Lord Tombs

979. Would you expect Boeing to know if it could or could not?

(Mr Cable) It is a very simple performance question, yes; it is just that it is not my area.

980. So, presumably, you assumed that, since they quoted it, it was possible? (Mr Cable) Yes; and, obviously, the Board was aware of this and were familiar with the Chinook.

Chairman

981. Can you remember whether the simulation indicated when the climb at that rate started?

(Mr Cable) I think I should just describe the math modelling a little bit more, I am afraid. It basically assumed a set of conditions based on our cases, which we suggested in the fax I mentioned, of air speed and climb rate, and, effectively, as far as the model was concerned, that was an infinite period at those conditions. So the model then allowed that to run for one second, just for convenience, and at the one second point the postulated manoeuvres were commenced, so the rear longitudinal stick and the up collective, or whatever was, postulated; so that was started at the one second point. The purpose of the model then was to try to assess the behaviour of the aircraft in the following seconds, which normally was a question of 4 or 5 seconds before things went totally haywire.

982. In this case, it was, what, 2.9, was it not?

(Mr Cable) Yes; in that period, the few seconds after the one second start of the manoeuvre point. The idea then was to look at the various parameters that we had established from the accident site and the wreckage, in other words, the pitch attitude, the flight path angle, the actuator extensions and the ground speed, and to see, for each of these cases that were tried, whether there was a point in those following seconds at which all those parameters, in general terms, approximated to the ones we had got. So that was the reason for doing a number of different cases. The model could not work backwards, you could only start from a point, work through and see if it took you to the conditions that you wanted to establish.

983. And, of course, nobody knew the height above sea level at which the point was taken, did they? (Mr Cable) No.

984. So the rate of climb, the Boeing conclusion says, climb rate 4,670 feet per minute, vertical distance travelled 128 feet in 2.9 seconds; well, my limited knowledge of mathematics rather suggested that that was a climb rate of about 2,650 feet per minute?

(Mr Cable) Yes, I would agree with that, Sir, as an average climb rate over that 2.9 seconds. The 4,670 feet per minute was the instantaneous climb rate at the end of the 2.9 seconds. So you start at 1,000 feet a minute climb rate at one second, you finish after 2.9 seconds, 3.9 seconds, whatever, at the 4,000 feet a minute; average rate is 2,600, whatever, but the instantaneous rate at the end is 4,000—

985. So they are not climbing at a steady rate, they are climbing at an increasing rate?

(Mr Cable) Yes, it is a very dynamic manoeuvre. If I could just go back and answer your original question, which I do not think I have done yet. As regards when the steady state conditions might have started, it is a complex matter, I think, which I cannot really answer, the reasons being that you would have to define what conditions you are looking at before the steady state condition commenced. Basically, in fairly simple terms, the DASH actuator is a relatively fast-acting actuator, with a stroke time, full travel, of something like 4.5 seconds, and it is in a part of the control circuit which is not loaded very highly. So the loads will change depending on aircraft manoeuvres, and so forth, but I suspect that, under any conditions, the DASH would travel from one end to the other in between 4 and 5 seconds, so relatively fast-acting. The LCTAs, on the other hand, act directly on the rotor heads and see the forces imposed by the blades, and are much beefier units which operate slower; so when they are unloaded I believe the full travel time is in the order of 8 seconds, it is different for front and rear because they have different travels, but in that order. When they are under higher loading, it can be up to around 40 seconds, or more, for full travel. Now both of those actuators, I would assume, are designed so that they do not have excessive lag for all normal manoeuvres of the aircraft, and they both operate according to airspeed, although, as I say, there are other inputs that go into the DASH. So the fact that the LCTAs were fully extended, according to their schedule, they only reach this point at 150 knots; if the aircraft, prior to the steady state conditions, were faster than 150 knots, they would still be fully extended when the steady conditions started, because they cannot extend any more.

The original modeling work did not include the full range of possible airspeeds at which the LCTAs could be in the extended position (approximately 130 to 148 knots). This had the effect of eliminating the lower airspeed and skewing the results to the high side. This has been eliminated in the latest simulation work done with BH Sim..

986. You cannot tell.

(Mr Cable) If the aircraft were initially slower than that, they would be partially retraced; when the steady state conditions that we have suggested here started, then at that point, because it is 150 knots that we are postulating, at that point, both LCTAs would start to extend to their full extend point. So it depends on what your air speed was before that how long you need and what the loads on the head are; fairly complex. The short answer would be, I suspect, that in all sensible conditions that I can envisage, including some fairly wild manoeuvres, it would just be a few seconds to actually set up the actuator extensions at those steady state conditions. That is my feeling. I cannot prove it.

987. And it is upon all these matters you have told us that the Boeing simulation concluded that the final flare was initiated 4 seconds before impact, is that right?

(Mr Cable) From the process I described of the modelling, the manoeuvre started at the one second point from an arbitrary zero. And in this case, which is where the, what they refer to as the critical conditions, where all these parameters more or less met the ones that we had established, in this one case, the critical conditions occurred at 3.9 seconds; in other words, 2.9 seconds from the start of the manoeuvre.

988. Was there any evidence that you found of emergency torque having been used before the impact?

(Mr Cable) There is not any system on the aircraft that will detect emergency torque. At high power conditions there are two possible limitations, and depending on the circumstances one or the other limit will be encountered first. So one will be the maximum torque, in other words, the rotational force, if you like, that the transmission is permitted to take, and the other one will be the maximum power turbine inlet temperature. So I believe this may be referring to the emergency power panel, which does register excessive power turbine inlet temperature, and if the PTIT on an engine is exceeded there is a clock starts in this panel, and after 5 seconds it will cause an indicator, a magnetic dolls-eye indicator, to operate and to some extent latch, and will also start a clock running. The clock on this standard of Chinook registered in whole minutes, so it is possible, after your 5-second grace period, when you actually start clocking up excessive PTIT time, that there would not be a change in indication of the clock; if you happen to start at a whole minute, you have got 59 seconds, or 60 seconds, before the thing will go to the next minute. The later ones record seconds; these ones just record minutes. I concluded that, well, I passed the clock indications to the Board in the statement. I cannot recall, offhand, whether there had been a change from the previous recorded values, which should be recorded by the ground crew, I think, before each flight; so the Board would have checked that. The dolls-eyes indicators are sort of like a simple stepper motor, so they are operated by magnetic fields and kept in a particular position by a permanent magnetic field. I concluded that under G loading, under acceleration, linear acceleration, rotational acceleration, there must be a possibility at certain levels of acceleration of the indicators moving. It is impossible to say what level acceleration this panel received on the accident site, so I concluded, I judged, that the dolls-eyes indicators, their `as found' position was not verifiable as being accurate.

Lord Tombs

989. And the clocks?

(Mr Cable) The clocks, as I say, I recorded the `as found' setting, I did not get into a great deal of detail about whether the clocks may have moved under impact forces, but accepted that you may well have quite a number of seconds of excessive PTIT without the clock changing, because it was recording whole minutes.

990. You said, in a letter to us, Mr Cable, that it appeared quite possible the magnetic indicator and meter indications could be altered by shock loading caused by the crash impacts. That is the case, is it?

(Mr Cable) Yes. I did not get far into looking at the clocks.

991. That was in the letter?

(Mr Cable) Yes; sure.

992. So we do not know whether emergency power had been used?

(Mr Cable) That is correct, yes.

Boeing agrees almost entirely with the AAIB's testimony and especially with its attempts to describe how difficult the accident investigation process can be. It is quite common to have disagreement among pieces of detailed evidence. As evidence is iterated over and reexamined the confidence level in the veracity of individual pieces of data can and often does change. This is something that does not seem to be very well appreciated by those outside of the accident investigation field.

Burke

724. What I find interesting is in the Air Accident Investigation Board's Report dealing with the Boeing simulation, they say various things were tried to find consistency, to use a general word, and "a ready match was found where initial conditions combined an air speed of 150 knots and a climb rate of 1,000 ft/min and below", but the finding of the AAIB on the instruments was 147 ground speed?

Answer: Yes.

725. We know there was a wind of about 20-25 knots so there seems to be a disparity between—

Answer: There are a number of disparities between—

726. —Boeing's assumed speed and what was found on the instruments.

Answer: There were a number of disparities between the simulation and what was actually found. There is one major one which I would wish to draw your attention to later on rotor speed. My view is that—and I was going to talk about this later—I would imagine the aircraft was planning to fly, and you have to pick a speed to fly a route for planning purposes when you are flight planning and the standard speed would be 135 knots indicated air speed on the Chinook, and I have seen nothing in any of the evidence over the years to make me think that they flew at anything other than that speed. That is a Mark I cruising speed, because it was a speed where it was below a high vibration speed. There is a particularly high vibration area on the Chinook, about 140 knots, it is to do with the airflow between the two rotors.

Boeing agrees with this statement.

727. You, presumably, would try and keep under the high vibration?

Answer: I think they planned 135 knots. Mark IIs when they came from the factory were smoother than the Mark 1s and it did not take very long for them to get just as rough at the Mark 1s. When the aircraft came out of the factory the rotors were checked very carefully through different conditions, it takes a very small number of hours flying to make them just the same as the Mark 1s. I think 135 knots is standard cruising speed.

135 knots is considered to be standard cruising speed.

728. In your observations on this you query whether a Chinook would be capable of accelerating from 130-135 knots airspeed to 150.

Answer: In the climb that it had on that is a point I wish to raise. You really have to make an effort to climb and accelerate, although it is operating below 18,000 because of the restrictions on it. You cannot have used very much fuel and you have to make really a large power input to do that.

729. You refer to what I think you mentioned a moment ago, to the disparity in the rotor speed used by Boeing and that found by the AAIB?

Answer: Yes, my Lord, it is a terribly important point.

730. We find it on page 20 of the AAIB report, the rotor was found at 100.5 per cent.

Answer: They gave that a category two certainty, which means it is a pretty sure one. The rotor speed indicator on a helicopter if you were flying on a nice clear day of 1,000 feet, no clouds, the one instrument that just must work on your helicopter is the rotor speed indicator, you rely on it to keep the rotors going round, if it goes too slowly the rotors are dragging on hinges, only held out by centrifugal force, except at very low speed on the ground, if they go too fast they might fly off. The limits are pretty small. Normal operation on the Chinook the FADEC would keep the rotor speed within about 0.5 per cent. The limits—I have some old flip cards here—even with the power off, without any power driving to the engines the absolute minimum, I think, was 91 per cent that the rotor speed could go down to. I do not think I have it here, I do, that is only a transient speed, you were not allowed to go down lower than that. If they go lower than that the blades actually cone up like this, (indicating) and if they get too far you cannot get them back and the helicopter will fall out of the sky. If they go too fast they will ultimately fall off.

731. We better get that demonstration into the notes, you put your hands up.

Answer: When the helicopter rotor blades go around too slowly, the blades, which are on hinges, fly up like that, (indicating). It is like a horizontal propeller on its side, there are three wings going round and they are each producing lift. If you put too much lift demand on them you get lots of drag, things slow down and then they are not going fast enough to produce a nice big rotor disc because they cone up. The difference in speed is really remarkable to what was found and what Boeing simulated it to be.

As previously stated, calculations show that excessive rotor coning does not become critical until very low rotor speed. At 50% RPM, the coning angle would be about 15 deg. At 90% RPM it is about 4.5 deg, which is only 1 deg greater than normal for the aircraft at the weight involved here.

732. Speed on the rotors.

Answer: Not on the steady limits but on the transient limits. It is a major discrepancy, I cannot stress how important that discrepancy is.

733. Perhaps we will take those a little bit further, the thrust lever, which I take is the collective—

Answer: The thrust or the collective, the American term is thrust lever.

In a tandem rotor helicopter it is referred to as a thrust lever.

734. —that was found in the position you would normally reach 100, is that right?

Answer: It was found well up and you would expect a very high demand of torque, a very high engine power.

It might be expected that during the final fractions of a second before impact the pilot would have had in extreme thrust and stick inputs (full thrust, full aft stick).

735. Does that fit in with the Boeing simulation or not?

Answer: I do not think it does. Again there are experts better than me to talk about this, the rotor speed is a glaring discrepancy.

736. You would expect the rotor speed to be higher?

Answer: No, the rotor speed as it went in, if the lever is right up, should be lower, like the simulation. The simulation in many ways was a good simulation, I do not know if it is accurate or not, that is pure speculation, I do not think that it is that far out, but there is a big difference in rotor speed, between what actually happened to the aircraft, what was found and what Boeing said.

737. To get this into the notes to make sure I understood it, as the collective was increased to 100 per cent or showing 100 per cent, you would have expected the rotor speed to have produced—

Answer: Was it actually 100 per cent, I do not have the notes in front of me, or fully up? If so, there are many other things you would expect. You would expect the engines to be at maximum power at that stage. In fact the engine would have had to be at maximum power to droop the rotor speeds of 91 per cent, with the aircraft going up.

As has been explained previously, this was a very dynamic maneuver in which the engines may have reached maximum power and the rotors drooped just prior to impact but not of such duration as to register in the DECU. Shortly after impact, this would have been negated by the effects of the desyncronization that occurred.

TUESDAY 16 OCTOBER 2001
WITNESS A

Chairman

807. What do you deduce, if anything, from the position in which the aircraft was flying with the left rudder 77 per cent after the accident?

Answer: Again that is a highly anomalous position for the rudder pedals to be, the yaw pedals to be in. There is absolutely no reason for applying that amount of yaw pedal during forward flight and the only reason I can think of for applying that much yaw pedal would be if the aircraft was becoming extremely difficult to control. The other theory I have to say put forward by the Board of Inquiry is that the displacement of the yaw pedals was caused by the force of the impact itself which was predominantly on the starboard side of the cockpit causing the right pedal to the forced back and the left pedal to be forced forward, so I do not think that can be ignored either.

Considering that initial impact was on the right front of the aircraft on terrain sloping right to left it would not be surprising to find the co-pilots right pedal forced aft and consequently left pedal forced forward.

808. You do not think—?

Answer: That could be ignored. It is certainly a possibility.

809. That this happened on impact?

Answer: Yes, my Lord. Could I just say, my Lord, while we are on the subject of theories, there are another couple of things that I have thought about. The main reasons I have pursued these theories is that the evidence used by the Board of Inquriy to postulate what happed to ZD576 is at best questionable. We have heard the RNS252 described as a black box and again it is a black box but only is as much as it is a box which is coloured black.

810. As opposed to orange or whatever the real black box is?

Answer: Exactly. It is certainly not an accident data recorder. In the Board of Inquiry the manufacturers are quoted as saying that "the equipment is not designed to provide `historic' data but attention has been paid to analysis of data items which may indicate the situation at a time earlier in the flight". At the point when the waypoint change occurred when one of the pilots went heads in to change the 252 over, it gave the geographical position of the aircraft over the sea but no height information. 15 to 18 seconds before impact the 252 gave a height for the aircraft but no corresponding geographical position. These two two-dimensional fixes were married together by the Board of Inquiry to provide a three-dimensional picture of events. It is however not certain that this three-dimensional track was the one taken by ZD576. Again as you have heard from Squadron Leader Burke these two points could have been joined by any manner of oscillations on the way.

Just because avionics equipment is not designed to provide 'historic' data does not mean that the data extracted from them cannot be used for such purposes. It is standard procedure in an aircraft accident to try to recover any non-volatile memory from avionics components that may shed light on what happened to the aircraft. Just because an aircraft does not have a flight data recorder does not mean there is no data available to recover from other avionics.

811. Just remind me, did the first reading which gave the position you referred to not give the height?

Answer: That is correct, my Lord, it did not give the height at that time.

812. Did it give the time?

Answer: I would need to go to the Board of Inquiry, I cannot remember exactly, I am afraid.

813. Because if it did not give the time it would mean that the 20 seconds which we have seen on various plans between the way point change and the crash must have been an assumed figure.

Answer: That is correct but I honestly cannot remember whether there was a time allocated to that. The thing that is more difficult for me to reconcile is a suggestion that the crew selected an inappropriate rate of climb to get over the Mull of Kintyre. You have heard during the course of these proceedings the crew elected to conduct a cruise climb in deteriorating weather. The Boeing simulation addressed a number of possible sets of parameters. Annex X, page 2 of the Board of Inquiry states—it is the Boeing simulation. "Of all cases examined only a few initiated at the 150 knot 1,000 feet per minute climb condition simultaneously met all the above criteria." The above criteria were pitch attitude, flight path angle, DASH actuator position and cyclic trim. "A few others initiated at the 135 knots, 1,000 feet per minute climb condition met the attitude, climb angle and DASH actuator criteria but failed to meet the cyclic trim criterion." So we are presented with a crew cruise climbing at 150 knots, 1,000 feet per minute rate of climb towards the Mull of Kintyre, which is interesting, because it is not physically possible, not within the normal operating envelope of the aircraft. A 17 tonne Chinook at a density altitude of around 400 feet and an airspeed of 150 knots will only climb 400 feet per minute before the aircraft limits are reached and specifically in that case the torque limitation of the gear box system of 101 per cent, and I know that because I have tried it. If the crew were climbing at 150 knots and at a 1,000 feet per minute rate climb it was not a cruise climb, they were already using emergency torque and would have been very aware of the fact.

As stated previously, for an HC Mk2 in the conditions that ZD576 was operating in at the time of the accident, ROC capability at 150 kts (TAS) should be approximately:
450fpm@Maximum Continuous Power (83% Torque);
1000fpm@Maximum (30 min) Power (92% Torque);
1500fpm@Drive System Limit (101% Torque);

814. Are you saying that either they were not doing 150 knots or they were not climbing at the rate which is stated there, one or the other?

Answer: One or the other, or both, my Lord.

815. Or neither.

Answer: The Boeing simulation, which was only a mathematical model, not an actual simulation, existed purely to explain the parameters of the aircraft at the time of impact. A lot seems to have been placed on this so-called simulation by the reviewing officer of the Board. I just wanted to explain that, perhaps, while a mathematical model meets the end criteria that mathematical model is probably flawed in view of real life.

The Boeing simulation was never considered to be a completely factual reconstruction of the accident sequence. It was a simulation, the heart of which is a math model as with all flight simulators.

816. Of course it presupposes, on any view, that the aircraft was under proper control at the time, does it not?

Answer: That is correct, my Lord

817. Anyway, you estimated that 150 knots at 1,000 feet per minute is just not possible.

Answer: That is correct.

818. To climb 1,000 feet per minute in a Chinook Mark II what speed would you have to be doing?

Answer: Perhaps I should take a step back and explain a little more about cruise climbing the aircraft. A cruise climb is a climb which occurs to enable the forward progress of the aircraft to be unimpeded. However, cruise climb, and crews are always aware of this, does not create great rates of climb. If Air Traffic Control asked an aircraft to alter its altitude by 5,000 feet the captain may elect to use cruise climb. If the height was required to be altered by a large amount then probably, almost certainly, the crew would reduce speed to a more efficient climbing speed, which depending on the rate of climb required can be anything from the minimum power speed, which is around about 80 knots, or, perhaps, 100 or 120 knots to get a greater rate.

To climb at 1000 fpm in an HC Mk2 in the conditions that ZD576 was operating in at the time of the accident, the True Airspeeds would be approximately:
142 kts @ Maximum Continuous Power (83% Torque);
150 kts @ Maximum (30 min) Power (92% Torque);
156 kts @ Drive System Limit (101% Torque);
etc.

819. What I was wondering is to achieve this figure of 1,000 feet per minute climb what sort of speed would the aircraft have to be travelling forward?

Answer: I do not know, that would have to be reexplored in the air, I would estimate about 130 knots to about 135 knots. Chairman: Certainly not 150 knots.

Lord Tombs

820. If it were 150 knots it would be 400 feet.

Answer: That is what we found in conditions approximately as close as we could to on the day of the accident, my Lord.

Chairman

821. You tried this, do you say, you tried this to see whether the Boeing simulation would be appropriate and could be relied upon?

Answer: No, to see whether the rate of climb is achievable at that speed. We did not attempt to replicate the Boeing simulation in any way, shape or form.

822. When did you do that?

Answer: Quite recently, my Lord.

Lord Tombs

823. What prompted you to do it?

Answer: I have spent many years looking at the evidence and it was something that leapt out at me after we had been forced into flying aircraft at 150 knots for a variety of reasons. The aircraft is not normally flown at 150 knots, it quite an unusual speed to fly at anyway. We did end up flying at that speed one day and I realised how much power we were using to keep the aircraft level and considered that it would be very difficult to climb the aircraft. As it turned out when we did try it we achieved 400 feet per minute. That may be varied depending on individual aircraft; some engines may be slightly more efficient than others. I would not imagine it would make a large material difference to the rate of climb.

Once again, for an HC Mk2 in the conditions that ZD576 was operating in at the time of the accident, ROC capability at 150 kts (TAS) should be approximately:
450fpm@Maximum Continuous Power (83% Torque);
1000fpm@Maximum (30 min) Power (92% Torque);
1500fpm@Drive System Limit (101% Torque);
This data is based on an aircraft with minimum performance engines.

824. We do not know whether emergency power was used or not, because in the last letter from the AAIB they said that the latch indicator which showed that it had been used could easily have been altered by impact so they attach very little reliability to those indicators.

Answer: There is another aspect to that, the limitation which the aircraft reached when we tried to do this was not a temperature limitation of the engines which would cause these latches to fall, it was torque limitation, which would have no record kept on the aircraft following the application of that emergency torque. So the normal torque limitation is 101 per cent, if for some reason the crew had used 120 per cent torque to climb no record would remain on the aircraft of that application of torque.

The data downloaded from #2 engine DECU which shows no evidence of torque or temperature exceedance that would indicate a sustained (more than a second or so) emergency power demand from that engine. Since all evidence indicated that the engines were power matched at impact, following impact and during rundown, there is a high probability that the #1 DECU would have provided similar data. This strongly suggests that sustained emergency power was not demanded.

825. Could they do that without using emergency power?

Answer: They could certainly get greater than 100 per cent without using emergency power. There would come a point where the engines were working so hard that they would go into emergency power too. Lord Tombs: I follow. That is very interesting.

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18 June 2002
8-7431-1-3724

To: Siglin, Shaw P10-44

cc: Keller, James P32-31
Landis, Kenneth P31-31

Subject: Mull of Kintyre - Response of HC Mk2 Helicopter to Longitudinal and Collective Inputs

References: (1) IOM 8-7431-1-3183, "Response of HC Mk2 Helicopter to Large Control Inputs", Boeing Helicopters, 11 July 1994.
(2) IOM 8-7431-1-3225, "Response of HC Mk2 Helicopter to Large Collective and Longitudinal Control Inputs", Boeing Helicopters, 30 September 1994.

Enclosed is the Flying Qualities contribution to 8-7D20-DSS-0306, which constitutes Enclosure 2 to that document.

John F Kannon

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X-13624

Mull of Kintyre
Response of HC Mk2 Helicopter to Longitudinal and Collective Control Inputs

Summary

The response of the HC Mk2 helicopter to a large array of collective and longitudinal control inputs has been investigated using the Boeing Helicopters Chinook Flight Simulator in the unpiloted non-realtime mode. The aircraft gross weight and altitude for the study were 37700 lb and 420 ft density altitude. The airspeeds immediately prior to the control inputs were 135 and 150 knots, and the initial rates of climb varied from 0 to 2000 fpm.

The objective was to determine whether one or more resultant maneuvers could be defined which, starting from the above initial steady flight conditions, would match certain physical evidence at the site of first impact at the accident scene referred to in Reference 1. The physical evidence consisted of strong indications that the pitch attitude was 30 deg nose up at impact, with a 20 deg climb angle. The DASH actuator was found to be at 23% extension, and the LCT actuators corresponded to rotor cyclic trim settings of 3.65 deg on the forward rotor and 3.75 deg on the aft rotor.

Based on the results of the simulator investigation, the following conclusions are drawn.

(1) - Beginning with initial flight conditions of 135 kt TAS and 1000 fpm climb rate, and with a 25 kt tailwind component, it is within the maneuvering capability of the HC Mk2 aircraft to simultaneously attain the pitch attitude and climb angle specified above at the impact site.

(2) - The DASH and LCT actuator extensions attained in the simulated maneuver were consistent with the actuator extensions found at the accident scene.

(3) - The resulting pullup maneuver probably involved a moderate up collective control input and a larger aft longitudinal stick input.

(4) - The flight profile associated with this simulated maneuver avoids contact with local terrain prior to the actual initial impact point.

Approximately 230 control response records were generated in an organized array of control input combinations, the results of which are presented in Appendix C. In addition, about 40 records of an exploratory nature were generated in the preliminary phase of the analysis to define the approach and circumscribe the issues involved. All of these simulation results will be retained in Boeing Helicopters records.

Introduction

The response of the HC Mk2 helicopter to rapid up collective control inputs combined with rapid aft longitudinal stick inputs has been investigated for a wide range of potential input combinations, using the Boeing Helicopters Chinook Flight Simulator (BH Sim) in the unpiloted non-realtime mode. Specifically the range of control inputs considered is:

Collective	0 to 3.00 inches @ 0.25 in. increments	(as limited by full up control stop)
Longitudinal	0 to 2.50 inches @ 0.25 in. increments	

Each control input was considered to be a ramp input lasting 0.5 sec which, for the larger inputs, represents a much higher input rate than normally used in most flight situations. In most cases, the collective and longitudinal inputs were applied simultaneously, but in some cases the inputs were phased so that one preceded the other by up to one second.

The helicopter conditions at the time of the maneuvers considered herein are:

Gross Weight	37700 lb
Center of Gravity	Sta 325 in.
Density Altitude	420 ft
Tailwind Component	25 knots

The initial steady flight conditions prevailing immediately prior to the control inputs were varied as follows:

True Airspeed	Steady Climb Rate
135 knots	0 fpm
135 knots	1000 fpm
135 knots	2000 fpm
150 knots	1000 fpm

The objective of the investigation was to attempt to define combinations of collective and longitudinal control inputs which would produce maneuvers during which the following conditions would simultaneously occur.

Pitch Attitude	30 deg nose up
Flight Path Angle	20 deg climb
DASH Actuator Position	23% extension
LCT Actuator Positions	3.65 deg (fwd)
	3.75 deg (aft)

Additional constraints were imposed by the allowable altitude gain, engine output torque and rotor speed variation during the maneuver:

Maximum Altitude Gain	195+/-50 ft in 18 sec prior to impact
Maximum Engine Torque	125% for 0.5 sec
Minimum Rotor Speed	97% of 225 RPM

The overall approach to the investigation was to apply the full array of collective and longitudinal control inputs defined above to the HC Mk2 simulation model at each of the four initial steady state flight conditions listed. The resultant output records were then examined to see which control combinations simultaneously met the pitch attitude and climb angle criteria of 30 deg and 20 deg respectively, within a +/-5 deg tolerance. Those control combinations that produced records which met the pitch/climb criteria were then further investigated to determine how many of them met the Additional Criteria relating to DASH and LCT actuator positions, maximum permissible altitude gain, maximum engine torque and minimum allowable rotor speed.

A typical simulation case was then selected to demonstrate a representative flight profile of the HC Mk2, including level flight at 615 ft above MSL, transition to a steady climb rate, and ending with the selected pitchup maneuver to an altitude of 810 ft above MSL.

Finally, there is a discussion of the potential impact of varying several important parameters of the investigation, including variation of initial flight conditions, differences in control inputs, and minor changes in pitch attitude and climb angle criteria.

Improvements to Chinook Simulation Capability

Before proceeding with the investigation, two important improvements were made to the capability of BH Sim to represent the control response of the Chinook in general, and the HC Mk2 in particular.

Rotor Stall Representation

The stall characteristics of the blade elements incorporated in the rotor model can significantly affect the simulated control response of the helicopter when maneuvering near its power and/or airspeed envelope limits. Since the pullup maneuvers investigated in this study involve penetration of the rotor stall boundary, the rotor stall representation to be used in the simulation was given some scrutiny.

In the Boeing simulation program, three choices of rotor stall representation are available:

(a) Blade element rotor off.
(b) Blade element rotor on - Extended lift slope off.
(c) Blade element rotor on - Extended lift slope on.

With option (a), there is essentially no representation of rotor stall effects. The airfoil lift slope is assumed to be linear without limit, and the drag is assumed to be a parabolic function of angle of attack, as in classical rotor theory.

With option (b), the lift and drag of each blade element of the rotor are obtained from tabulated wind tunnel test data for the appropriate angle of attack and Mach number of that element. These wind tunnel data incorporate stall effects as represented by the static lift and drag coefficients of the airfoil at angles of attack beyond stall.

With option (c), the oscillatory nature of the blade element angle of attack is recognized as the blade rotates about the rotor hub. Under these oscillatory conditions, the stall at each blade element is delayed to a higher angle of attack than under static conditions. This is represented in the simulation by extending the airfoil lift slope to a higher stall angle than the static wind tunnel data in option (b) would indicate. Appropriate adjustments are also made to the airfoil drag coefficient.

For simulated maneuvers at flight conditions well removed from the helicopter envelope limits, each of the three stall representation options will produce substantially similar results. However, for maneuvers initiated at the high power high airspeed conditions involved in this study, option (a) will give unrealistic results, and is therefore discarded. The simulation was then run with rotor stall options (b) and (c), using the final maneuver control inputs presented in Reference 2. The maximum rotor thrust and maximum blade loading coefficient obtained from BH Sim is tabulated below for each stall option.

Initial Condition		Option (b)	Option (c)
135 kt / 1000 fpm	Maximum Thrust Max. Ct/(sigma)	40000 lb 0.140	52000 lb 0.182
150 kt / 1000 fpm	Maximum Thrust Max. Ct/(sigma)	44000 lb 0.154	58000 lb 0.203

Since the maximum blade loading coefficient Ct/(sigma) is generally considered to be about 0.160, Option (b) is conservatively selected as the more realistic blade stall representation. This agrees well with the maximum Ct/(sigma) = 0.144 attained in the original analysis reported in Reference 2. It is also more compatible with the the maximum steady high G banked turn capability of Ct/(sigma) = 0.125 demonstrated in flight test on the CH-47B at 135 kt TAS.

FADEC Representation

The Boeing simulation of the 55-L-712 engines and fuel control is based on documentation provided by the engine manufacturer to Boeing. This engine simulation is a map-based model that has been correlated to a comprehensive thermodynamic model of the engine for operational conditions. It assumes both engines operate in a load-matched fashion. Dynamics are limited to the gas generator speed and some elements of the fuel flow stepper motor. Ambient condition variations are accounted for in all of the performance map data.

The digital fuel control or FADEC is modeled directly in accordance with the detailed requirements specification for the unit by Chandler Evans. It models all aspects of FADEC performance except start up/shut-down, load sharing (because the simulated engines are matched and do not require balancing), reversionary control, and failure logic. In addition, since the overall model is a rigid airframe representation, notch filters in the power turbine governor control laws, intended to avoid structural modes, have not been implemented. Otherwise, it fully models all modules in the FADEC explicitly, as specified by Chandler Evans.

The performance of the integrated engine/fuel control/ rotor/airframe simulation model at Boeing has been validated through a variety of flight test data points with U.S. Army and foreign CH-47 models for this engine configuration and others that are similar.

Simulated Helicopter Responses to an Array of Control Input Combinations

A typical maneuver record is presented in Figure 1(a), (b) & (c) for the response resulting from the simultaneous application of 0.5 in. collective and 2.00 in. longitudinal stick at 1.0 sec, designated as Case A-24. In Figure 1(a), the climb angle data have been artificially increased by 10 deg; e.g. the initial steady climb angle of 3.53 deg for 1000 fpm climb at 160 kt ground speed (135 kt TAS + 25 kt tailwind) is shown as 13.53 deg at T = 0 sec. Consequently an intersection of the pitch attitude and climb angle traces at 30 deg would represent the simultaneous attainment of 30 deg pitch and 20 deg climb at that point. In the example shown in Figure 1, the intersection at 26 deg represents the simultaneous attainment of 26 deg pitch and 16 deg climb at 3.7 sec. By focusing on such intersection points, a large amount of output data can be rapidly scanned to determine which combinations of control inputs come closest to simultaneously meeting the 30 deg pitch attitude and 20 deg climb angle criteria specified above as representative of the conditions at impact.

Simulation output records for the full array of control input combinations investigated herein are presented in Appendix C. The results of applying the scanning procedure to the data in Appendix C are shown in Tables 1 through 4, in which intersection angles and times are presented for the simultaneous collective and longitudinal stick inputs indicated in the tables. The initial steady flight conditions for each table are noted therein. Those cases which failed to result in an intersection of the pitch and climb traces are designated by a "X".

In most cases an intersection at the desired 30 deg value did not occur. However, a significant number of cases came within +/-2 deg of this criterion, and many others came within +/-5 deg. These cases are designated by heavy shaded boxes and light boxes respectively around the appropriate entries in the tables. Tallying the number of boxes in each table produces the following score for each initial condition considered. Note that in Tables 3 and 4 there are no entries for collective inputs greater than 1.75 in. and 2.00 in. respectively, because the up stop prevents larger collective inputs when initially trimmed at these high power conditions.

			Criteria Met Within	
Table	True Airspeed	Steady Climb Rate	+/-5 deg	+/-2 deg
1	135 knots	0 fpm	6 cases	2 cases
2	135 knots	1000 fpm	13 cases	4 cases
3	135 knots	2000 fpm	26 cases	7 cases
4	150 knots	1000 fpm	4 cases	1 case

From these data it is seen that the greatest number of control combination cases which meet or come close to the 30 deg attitude / 20 deg climb criteria occur for initial steady flight conditions of 135 knots with a 2000 fpm climb rate. A typical example is shown in Figure 2, designated as Case B-17, which is identical to the case in Figure 1 except for the higher initial climb rate.

Although this case comes closer to meeting the 30 deg attitude / 20 deg climb criteria (28 deg at 3.75 sec), several other considerations combine to eliminate it as a realistic portrayal of potential HC Mk2 maneuver capability in these flight circumstances.

(a) The initial steady flight condition of 2000 fpm climb at 135 knots requires excessive power (97% of torque limit), and is therefore not a likely entry condition for the maneuver.

(b) Engine torque (137%) and rotor speed (90%) at the pitch/climb intersection point fail to meet the maximum torque and minimum RPM constraints listed above.

(c) Backward projection of the flight path profile of this, and most of the other 2000 fpm climb cases, starting with the impact point at 810 ft AMSL, indicates that the helicopter would have previously have struck the ground farther down the hillside.

For these reasons all of the cases in Table 3, with initial conditions of 135 knots with 2000 fpm climb rate, are discarded from further consideration, and the "boxed" cases from Table 2 at 135 knots TAS with 1000 fpm climb rate are retained for further study, to see whether they meet the Additional Maneuver Criteria discussed below.

Additional Maneuver Criteria

The thirteen cases selected for further study are set forth in Table 5, together with their pitch/climb intersection angles and times obtained from Table 2. They are arranged in order of decreasing collective input and increasing longitudinal stick input. The simulation records for these cases are included in Appendix B. For each of these cases, five additional criteria are applied at the time of pitch/climb intersection, in order to determine which cases would most closely represent the final pullup maneuver of ZD576. These criteria are:

(1) DASH actuator extension shall not be less than 23%.

(2) LCT actuator positions shall not be less than 3.65 deg (fwd rotor) and 3.75 deg (aft rotor).

(3) Altitude gain shall not exceed 195+/-50 feet in the 18 sec prior to impact.

(4) Engine torque shall not exceed 125+/-5%.

(5) Rotor speed shall not decrease more than 3%.

DASH and LCT Actuator Extensions

Referring again to Figure 1(a), the third and fourth graphs show the LCT and DASH actuator positions. The criteria for the final actuator positions set forth above are obtained from Reference 1, sections 7.4.6 (DASH) and 7.4.7 (LCTs). These final actuator positions are represented by the horizontal dotted lines in each graph. The LCT actuators attain their final extensions at 4.3 sec (fwd) and 3.9 sec (aft), and the DASH actuator attains its final extension at 4.1 sec. These times do not coincide exactly with the impact time indicated in this record by the pitch/climb intersection point at 3.7 sec. However, this is readily explainable by the strong possibility that the electrical system continued to function for a moment or two after impact, thereby allowing the actuators to continue operating until subsequent loss of power. Hence intersection times in Table 5 occurring prior to the attainment of final LCT and DASH positions do not necessarily exclude these cases from further consideration. However, intersection times occurring after these final actuator positions are attained do eliminate such cases, because the decreasing airspeed and increasing pitch attitude during the pitchup maneuver would preclude the actuators from reversing and moving back in the extend direction. In Table 5, the DASH and LCT overrun times are listed for each case. A positive overrun indicates that the actuator has retracted too far beyond the relevant criterion set forth above, and hence the associated pitchup maneuver does not meet this criterion.

Altitude Gain

SuperTANS data show that, eighteen seconds before impact, an altitude update was made, which established the helicopter height at that point to be 615+/-50 ft AMSL. With the impact point fixed at 810 ft AMSL, this places a constraint on the allowable altitude gain during the simulated final pitchup maneuver. Subtracting the nominal values above, the height gain during the pitchup maneuver cases in Table 5, obtained from the height graphs in at the time of pitch/climb intersection, must not exceed 195 feet.

Engine Output Torque

Data extracted from the DECUs indicate that emergency power of 125% was not exceeded on the engines during (or before) this flight. Allowing 5% for potential error in the engine power output of the simulation model and in the calculated rotor power required in climbing flight, and further allowing approximately 0.5 sec for any such power exceedences to register in the DECUs, the engine output torque at 0.5 sec before the pitch/climb intersections in Table 5 must not exceed 130%.

Rotor RPM Droop

During most of the final pullup maneuver cases considered herein, the computed rotor speed falls below 100%, due primarily to the sudden increase in rotor power required. However, the final rotor tachometer reading has been established at 100.4%. This discrepancy can be resolved by considering that the RPM droop occurs during the pullup maneuver prior to impact, when the engines are supplying power to both rotors, whereas the recovery to 100.4% RPM occurs in the few moments immediately after impact, when the forward rotor drive shaft has failed and all the power from both engines is available to drive the aft rotor only. This would allow the aft rotor to rapidly return to 100% RPM with perhaps a slight overshoot, as found at the final wreckage site. Although no precise criterion is set for the maximum allowable RPM droop at impact, smaller droops before impact obviously allow a more rapid aft rotor recovery to 100% RPM after impact and before complete power loss.

TABLE 5

COMPLIANCE WITH ADDITIONAL MANEUVER CRITERIA

Case Number	Coll. Input	Long. Input	Intersection Angle / Time (deg) / (sec)	DASH Overrun (sec)	LCT Overrun (sec)	Altitude Gain (ft)	Engine Torque (%)	Rotor Speed (%)
A-30	3.00	0.00	26 / 5.1	1.0	0.7	**188**	143	80
50-1K	2.75	0.25	26 / 4.6	0.5	0.3	**161**	142	82
A-28	2.50	0.50	27 / 4.7	0.6	0.4	**171**	141	83
47-1K	2.25	0.75	27 / 4.5	0.4	0.2	**161**	140	85
A-20	2.00	1.00	28 / 4.5	0.4	0.2	**164**	139	86
43-1K	1.75	1.25	26 / 4.0	**-0.1**	**-0.2**	**133**	137	88
40-1K	1.50	1.25	31 / 5.1	1.0	0.6	210	136	89
A-07	1.50	1.50	26 / 3.8	**-0.3**	**-0.3**	**121**	135	90
38-1K	1.25	1.50	30 / 4.5	0.4	0.1	**168**	134	92
36-1K	1.00	1.75	26 / 3.8	**-0.3**	**-0.4**	**121**	132	95
35-1K	0.75	1.75	34 / 5.1	1.0	0.7	222	131	96
A-24	0.50	2.00	26 / 3.7	**-0.4**	**-0.4**	**113**	**122**	**97**
32-1K	0.25	2.00	30 / 4.5	0.4	0.1	**166**	**120**	**98**

Nearly all cases in Table 5 meet the altitude gain constraint of 195 feet. However, only four cases meet the DASH and LCT actuator extension criteria, and only two meet the maximum engine torque requirement of 130%. These cases are designated by the boldface underlined data entries in the table. Arbitrarily selecting 3% as the greatest aft rotor RPM increase that might be generated by both engines momentarily supplying full power to the aft rotor only, the minimum rotor RPM would be 97% with only two cases meeting this requirement.

Considering the data in Table 5 as a whole, it is apparent that the control input combinations at the bottom end of the table tend to more readily meet all the criteria than those at the top. Since collective inputs are decreasing toward the bottom of the table while longitudinal inputs are increasing, it is therefore concluded that the final pullup maneuver most likely involved low collective and large aft longitudinal control inputs. However, the fact that only Case A-24 meets all the criteria in Table 5 does not imply that this is precisely the only control combination that could have done so.

Flight Profile of Final Pullup Maneuver

Since Case A-24 meets all the Additional Criteria discussed above, and also comes close to meeting the impact angle conditions of 30 deg pitch attitude and 20 deg climb angle, it is selected as a typical example of a possible flight path approach to the point of impact. In Figure 3, the flight path profile at the bottom of Figure 1(b) is superimposed on the terrain profile obtained from Reference 1, section 5.5. Note that the vertical and horizontal scales are not identical, and that the helicopter image is not to scale. The common point of coordination is the impact point at 810 ft AMSL, 3930 ft from the MSL coastline, at 18 sec after the last SuperTANS altitude update. The impact point in Figure 1(a) is considered to be the pitch/climb intersection at 3.7 sec. Backplotting from this intersection to the beginning of the control inputs at 1.0 sec in Figure 1, the conditions at 15.3 sec in Figure 3 are:

Airspeed	135 kt TAS	Distance	3224 ft from MSL Coast
Ground Speed	160 kt TGS	Height	713 ft AMSL
Rate of Climb	1000 fpm		

Further backplotting, the 1000 fpm climb rate intersects with altitude 615 ft AMSL at 9.4 sec in Figure 3, at which point the climb rate is assumed to begin from the following steady level flight conditions:

Airspeed	135 kt TAS	Distance	1630 ft from MSL Coast
Ground Speed	160 kt TGS	Height	615 ft AMSL
Rate of Climb	0 fpm		

Backplotting 9.4 sec from this point brings us to the point of last altitude update at 910 ft offshore, 18 seconds prior to impact.

Hence the aircraft maneuver and the flight path profile described in Figures 1 and 3, resulting in the impact with terrain at 810 ft AMSL, are within the HC Mk2 flight capabilities, including the final pullup maneuver which meets all the criteria discussed above within reasonable limits.

Variation of Initial Flight Conditions

Minor variations in the initial flight conditions assumed for the final pullup maneuver in this study could make a significant difference in the overall number of control input combinations in Tables 1 through 4 which meet the simultaneous pitch attitude and climb angle criteria.

Initial Airspeed and Climb Rate

It is assumed in Figure 3 and elsewhere in the study that the level flight airspeed prevailing as the helicopter approached the Mull was maintained unabated in the steady climb phase immediately prior to the pullup. It is possible, however, that in actual fact airspeed would be allowed to decrease and the excess energy converted to additional climb rate, providing a somewhat steeper ascent than the steady climb shown in Figure 3. If both of these modifications to the initial conditions were incorporated, the data in Tables 1 through 4 for the final pullup maneuver would show more cases meeting the pitch/climb angle criteria, because the number of such cases in the tables increases with lower airspeed and higher climb rate. However, initial airspeed can not be much below 135 kt TAS because of the LCT actuator retraction criteria, and climb rate can not be much greater than 1000 fpm because of terrain clearance considerations discussed above.

Assuming initial conditions of 130 kt TAS and 1200 fpm to be the minimum airspeed and maximum climb rate attainable under above LCT and terrain restrictions, interpolation among the four tables indicates that there could be 6 cases which would meet the simultaneous pitch attitude and climb angle criteria within +/-2 deg, and 18 cases which would meet the criteria within +/-5 deg. This contrasts with the 4 cases and 13 cases respectively in Table 2 which meet the criteria at 135 kt TAS and 1000 fpm initial conditions.

Estimated Gross Weight

Throughout this study, the helicopter gross weight has been taken as 37700 lb, per Section 5.8 of Reference 1. If the weight had been taken as 40000 lb, the typical pullup maneuver in Figure 1(a), (b) & (c) would have been somewhat altered as shown on Figure 4(a), (b) & (c), indicating a quicker pitchup response and a slightly reduced climb angle. A slightly larger collective input and a somewhat smaller aft longitudinal input would likely compensate for these response changes, and the results would then be virtually the same as for the 37700 lb condition.

Variation of Control Input Profiles.

Modifications to the timing, rate of input and shape of the control inputs (e.g. a large initial input followed quickly by backing off to a lesser input) can affect the details of the control response of the helicopter. Of these, only the effect of relative timing of the collective and longitudinal control inputs has been systematically investigated, since the effects of input rate have not been found significant, and the effect of input shape can be duplicated by other simpler inputs.

Effect of Phased Control Inputs

The results of the analysis up to now have been predicated on the collective and longitudinal control inputs being applied simultaneously. These results can be significantly influenced by the phasing of the inputs, such that one might be applied before the other. The effects of applying longitudinal inputs one second prior to the collective are displayed in Table 6. With collective inputs of 1.00 in., several closely spaced longitudinal inputs from 1.60 in. to 2.30 in. were applied both simultaneously and pre-phased by one second. The same was done for collective inputs of 2.00 in., with longitudinal inputs from 0.80 to 1.40 in.

For all control combinations, the helicopter responses with longitudinal pre-phasing were significantly different from those with simultaneous inputs, as shown by a case-by-case comparison of pitch/climb intersection angles and times in the table. In addition, for control combinations with 2.00 in. collective, the number of cases which meet the 30/20 deg pitch/climb criteria within +/-2 deg increases from one to three, and the number which meet the criteria within +/-5 deg increases from three to five, as indicated by the number of "boxed" cases in the table. This indicates that, if control phasing had been considered in compiling the data in Tables 1 through 4, a significantly greater number of cases would have been retained for further study. However, when this was done for several of these phased combinations, it was found that most of them were subsequently eliminated by the Additional Maneuver Criteria discussed above. In particular, since the longitudinal input precedes the collective, the pitch attitude diverges earlier in the data record, and the DASH actuator retracts to 23% before the critical pitch/climb intersection angle is attained.

TABLE 6

EFFECTS OF CONTROL INPUT PHASING

TAS = 135 kt R/C = 1000 fpm

Collective = 1.00 in.				Collective = 2.00 in.		
Long. (in.)	Simult. Input	Phased Input		Long. (in.)	Simult. Input	Phased Input
1.6	33 / 5.1	X		0.8	X	X
1.7	**28 / 4.1**	33 / 4.2		0.9	**29 / 4.9**	X
1.8	25 / 3.6	**28 / 3.3**		1.0	27 / 4.5	**30 / 4.7**
1.9	24 / 3.4	26 / 2.9		1.1	25 / 3.9	**29 / 4.3**
2.0	23 / 3.2	24 / 2.6		1.2	24 / 3.6	**28 / 4.0**
2.1	22 / 3.0	22 / 2.3		1.3	24 / 3.5	26 / 3.5
2.2	22 / 2.9	21 / 2.1		1.4	23 / 3.3	25 / 3.2

Variation of Pitch Attitude and Climb Angle Criteria

Minor modifications to the specified pitch attitude and climb angle criteria required to match evidence at the first impact site can have a significant effect on the number of control input combinations in Tables 1 through 4 which simultaneously meet these criteria.

Relaxation of Pitch Attitude Criterion

Based on CAD modeling of local terrain and of the approach of a Chinook model along the appropriate heading, it was concluded in Reference 1, Section 5.9 that, with no allowance for rotor coning and flapping, the fuselage attitude at first impact would have had to be approaching 35 deg nose up, in combination with a 7 deg left roll angle, in order for the forward rotor blades to clear the terrain with no blade strikes. With an estimated allowance for rotor coning and flapping, it was assessed that a fuselage pitch attitude of 30 deg, with 7 deg left roll, would be consistent with the physical evidence at the first impact site.

From Figure 10 of Reference 1, the forward rotor blade at azimuth 135 deg (i.e. in the right front rotor quadrant) would be the first to strike the hillside. The computed flapping angle of the blade at this azimuth is shown in the bottom graph of Figure 1(c). This trace includes the effects of coning, longitudinal and lateral flapping, and the additional coning due to 2G normal force at 3.7 sec. The oscillatory shape is due to higher blade flapping harmonics at high speed, and entry into the blade stall region during the maneuver. The blade flapping angle at 3.7 sec varies between 7.5 and 9.0 deg in this particular simulation case. In other cases, flapping angles as great as 10 to 12 deg were computed for the blade at this azimuth.

The blade flapping angle at this specific azimuth has an influence on the 30 deg pitch attitude requirement assessed in Reference 1, since this impact attitude was inferred from the absence of blade strikes on the ground. If for some reason (e.g. a last second left stick and/or rudder pedal input) this blade angle were increased, the rotor clearance to the ground would be improved, and the corresponding fuselage pitch attitude required to provide this clearance would be reduced. If this pitch attitude requirement, along with the climb angle requirement, were both relaxed by as little as 2 deg, the number of control input combinations in Tables 1 through 4 which would simultaneously meet these relaxed requirements would be more than doubled. Table 7 is an example of how many more of the control combinations in Table 2 would become eligible for further comparison with the Additional Maneuver Criteria of Table 5 if these angular criteria were relaxed.

Systematic Variation of Pitch and Climb Angle Criteria

All preceding work in this analysis has presumed that the pitch attitude at impact is 10 deg greater than the simultaneous climb angle at that point. This was based on the specified requirements of 30 deg pitch attitude and 20 deg climb angle at impact, and was implemented in the analysis by adding 10 deg to the climb angle data in the plotted simulation results, as in Figure 1(a). Those cases in which the pitch attitude and modified climb angle intersected between 25 and 35 deg were then considered to have impacted the hillside at that intersection point.

If allowance is made for a possible systematic variation of +/-2 deg in each of these two criteria, the number of combined criteria variations is nine as follows.

Pitch Criterion (deg)	Climb Criterion (deg)	Criterion Delta (deg)
<u>28</u>	**<u>18</u>**	**<u>10</u>**
28	20	8
28	22	6
30	18	12
<u>30</u>	**<u>20</u>**	**<u>10</u>**
30	22	8
32	18	14
32	20	12
<u>32</u>	**<u>22</u>**	**<u>10</u>**

The three boldface underlined combinations with Delta = 10 deg have already been addressed. But other combinations having Deltas from 6 to 14 have not been considered. If all these possible variations of pitch and climb angle criteria were considered, and especially the combinations involving criteria reductions, new versions of Tables 1 through 4 could be compiled, having many more "boxed" control combinations which meet these criteria and would therefore be candidates for further investigation with respect to the Additional Maneuver Criteria.

Conclusions

The pullup maneuver set forth in Figure 1 and the corresponding flight path profile shown on Figure 3, resulting in the impact with terrain at 810 ft AMSL, are within the HC Mk2 flight capabilities for the most likely initial flight conditions at the start of the maneuver.

The simulated pullup maneuver meets or closely approximates all the requirements and restrictions imposed by the physical evidence at the scene, including:

- Pitch attitude and climb angle at impact.
- DASH and LCT actuator extensions at power loss.
- Altitude gain during the pullup.
- Clearance with local terrain.
- Maximum engine torque output.
- Minimum rotor RPM.

The most likely initial conditions at the start of the pullup maneuver are 135 kt TAS with a 1000 fpm climb rate. These initial conditions agree closely with the flight conditions determined independently in Reference 4 and elsewhere.

The final pullup maneuver probably involved a low to moderate collective control input combined with a larger aft longitudinal stick input.

Minor variations in the following parameters would bring the simulated pullup maneuver even closer to the requirements imposed by the physical evidence:

- Slightly lower initial airspeed (130 kt) and slightly higher climb rate (1200 fpm).
- Phasing of the collective and longitudinal control inputs so that they are not necessarily applied simultaneously.
- Relaxation by 2 deg of the pitch attitude and climb angle criteria at impact.

Variations in gross weight up to 40000 lb would not affect the helicopter responses in any way which would alter the above conclusions.

References

1 - Accident to Chinook HC2 Registration ZD 576 at Mull of Kintyre, Scotland, on 2 June 1994, AAIB Statement to Board of Inquiry.

2 - IOM 8-7431-1-3183, "Response of HC Mk2 Helicopter to Large Collective and Longitudinal Control Inputs", Boeing Helicopters, 30 September 1994.

3 - 352-PJ-808-1, "Flight Test Acceptance Document, HC Mk2 Helicopter", Boeing Helicopters.

4 - Enclosure 4, "Mull of Kintyre - Analysis of Available Data", Boeing Helicopters, 18 June 2002.

TABLE 1

Pitch Attitude / Climb Angle Intersections @ 135 kt
(deg & sec)

Collective Inputs (in.)	Initial R/C (fpm)	Longitudinal Inputs (in.)										
		0.00	0.25	0.50	0.75	1.00	1.25	1.50	1.75	2.00	2.25	2.50
0.00	0										17 / 3.0	
0.25	0								X	X	17 / 2.8	
0.50	0								X	22 / 3.7	17 / 2.7	
0.75	0							X	X	19 / 3.1		
1.00	0						X	X	27 / 4.3	20 / 3.0		
1.25	0						X	X	20 / 3.2			
1.50	0					X	X	**30 / 4.4**	20 / 3.0			
1.75	0				X	X	**30 / 4.8**	20 / 3.2				
2.00	0			X	X	X	23 / 3.5	20 / 3.0				
2.25	0		X	X	X	23 / 3.7	20 / 3.2					
2.50	0	X	X	X	25 / 4.1	20 / 3.3						
2.75	0	X	X	26 / 4.4	22 / 3.5							
3.00	0	X	26 / 4.5	21 / 3.6								

TABLE 2

Pitch Attitude / Climb Angle Intersections @ 135 kt

Collective Inputs (in.)	Initial R/C (fpm)	Longitudinal Inputs (in.)										
		0.00	0.25	0.50	0.75	1.00	1.25	1.50	1.75	2.00	2.25	2.50
0.00	1000							X		X	21 / 3.1	20 / 2.8
0.25	1000								X	**30 / 4.5**	21 / 3.0	
0.50	1000			X		X		X	X	26 / 3.7	22 / 3.0	20 / 2.7
0.75	1000							X	34 / 5.1	24 / 3.3		
1.00	1000			X		X	X	X	26 / 3.8	23 / 3.2		20 / 2.7
1.25	1000					X	X	**30 / 4.5**	24 / 3.4			
1.50	1000	X		X		X	**31 / 5.1**	26 / 3.8	23 / 3.2	22 / 3.0		20 / 2.6
1.75	1000				X	X	26 / 4.0	24 / 3.4				
2.00	1000	X		X	X	**28 / 4.5**	24 / 3.6	23 / 3.2		21 / 2.8		20 / 2.6
2.25	1000		X	X	27 / 4.5	24 / 3.7	23 / 3.3					
2.50	1000	X	X	27 / 4.7	24 / 3.8	23 / 3.5		22 / 3.0		20 / 2.7		20 / 2.6
2.75	1000	X	26 / 4.6	24 / 3.9	23 / 3.5							
3.00	1000	26 / 5.1	24 / 4.0	23 / 3.6		21 / 3.1		20 / 2.8		20 / 2.7		20 / 2.5

TABLE 3

Pitch Attitude / Climb Angle Intersections @ 135 kt
(deg & sec)

Collective Inputs (in.)	Initial R/C (fpm)	Longitudinal Inputs (in.)										
		0.00	0.25	0.50	0.75	1.00	1.25	1.50	1.75	2.00	2.25	2.50
0.00	2000									36 / 5.4	27 / 3.4	25 / 3.0
0.25	2000								X	**30 / 4.1**	26 / 3.3	25 / 3.0
0.50	2000							X	33 / 5.1	**28 / 3.7**	25 / 3.1	
0.75	2000							X	**29 / 4.2**	26 / 3.4	25 / 3.1	
1.00	2000						X	**30 / 4.8**	27 / 3.7	25 / 3.3	25 / 3.0	
1.25	2000					X	**30 / 5.1**	27 / 4.0	26 / 3.5	25 / 3.2		
1.50	2000					X	**28 / 4.3**	27 / 3.8	25 / 3.3			
1.75	2000				X	**29 / 5.1**	27 / 4.1	26 / 3.6	25 / 3.3			
2.00	2000											
2.25	2000											
2.50	2000											
2.75	2000											
3.00	2000											

TABLE 4

Pitch Attitude / Climb Angle Intersections @ 150 kt

Collective Inputs (in.)	Initial R/C (fpm)	Longitudinal Inputs (in.)										
		0.00	0.25	0.50	0.75	1.00	1.25	1.50	1.75	2.00	2.25	2.50
0.00	1000					X		X	X	23 / 3.6	20 / 3.0	20 / 2.8
0.25	1000							X	$\boxed{26 / 4.3}$	21 / 3.3	20 / 3.0	
0.50	1000			X		X	X	$\boxed{\mathbf{29 / 5.1}}$	24 / 3.8	21 / 3.2		
0.75	1000						X	24 / 4.1	22 / 3.5			
1.00	1000			X		X	X	23 / 3.7		20 / 3.0		
1.25	1000				X	$\boxed{26 / 5.0}$	24 / 4.0					
1.50	1000	X		X	X	23 / 4.1		21 / 3.3				
1.75	1000		X	X	23 / 4.3							
2.00	1000	X	X	$\boxed{25 / 4.8}$		21 / 3.5						
2.25	1000											
2.50	1000											
2.75	1000											
3.00	1000											

TABLE 7

Pitch Attitude / Climb Angle Intersections @ 135 kt
(Angular Criteria Relaxed 2 Deg)

Collective Inputs (in.)	Initial R/C (fpm)	Longitudinal Inputs (in.)										
		0.00	0.25	0.50	0.75	1.00	1.25	1.50	1.75	2.00	2.25	2.50
0.00	1000							X		X	21 / 3.1	20 / 2.8
0.25	1000								X	**30 / 4.5**	21 / 3.0	
0.50	1000			X		X		X	X	**26 / 3.7**	22 / 3.0	20 / 2.7
0.75	1000							X	34 / 5.1	24 / 3.3		
1.00	1000			X		X	X	X	**26 / 3.8**	23 / 3.2		20 / 2.7
1.25	1000					X	X	**30 / 4.5**	24 / 3.4			
1.50	1000	X		X		X	31 / 5.1	**26 / 3.8**	23 / 3.2	22 / 3.0		20 / 2.6
1.75	1000				X	X	**26 / 4.0**	24 / 3.4				
2.00	1000	X		X	X	**28 / 4.5**	24 / 3.6	23 / 3.2		21 / 2.8		20 / 2.6
2.25	1000		X	X	**27 / 4.5**	24 / 3.7	23 / 3.3					
2.50	1000	X	X	**27 / 4.7**	24 / 3.8	23 / 3.5		22 / 3.0		20 / 2.7		20 / 2.6
2.75	1000	X	**26 / 4.6**	24 / 3.9	23 / 3.5							
3.00	1000	**26 / 5.1**	24 / 4.0	23 / 3.6		21 / 3.1		20 / 2.8		20 / 2.7		20 / 2.5

Fig. 1(a) HC MK II SIMULATED CONTROL RESPONSE

INITIAL TAS = 135 Kts.
INITIAL ROC = 1000 fpm.

CASE A 24

CYCLIC TRIM: ADVANCED

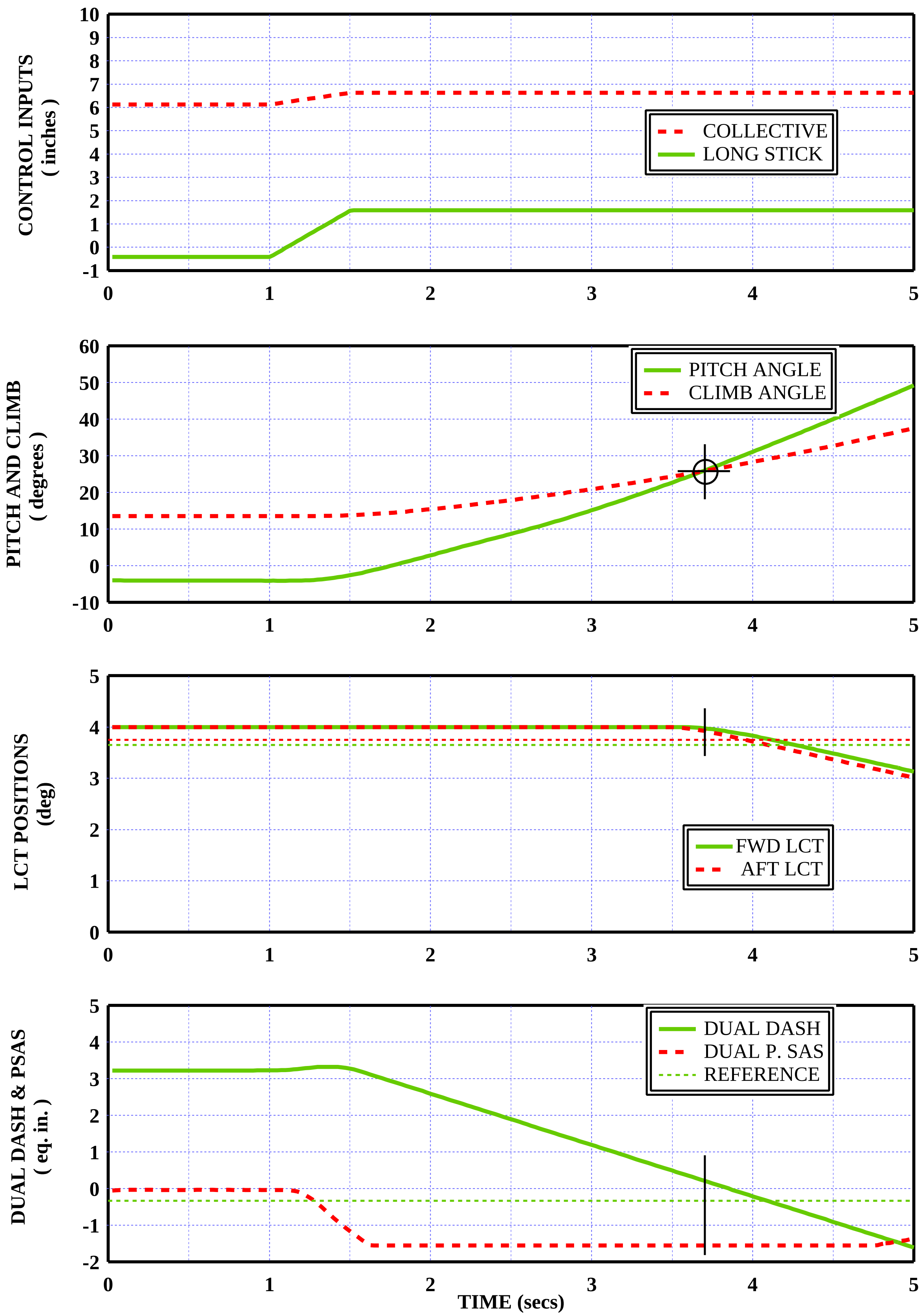
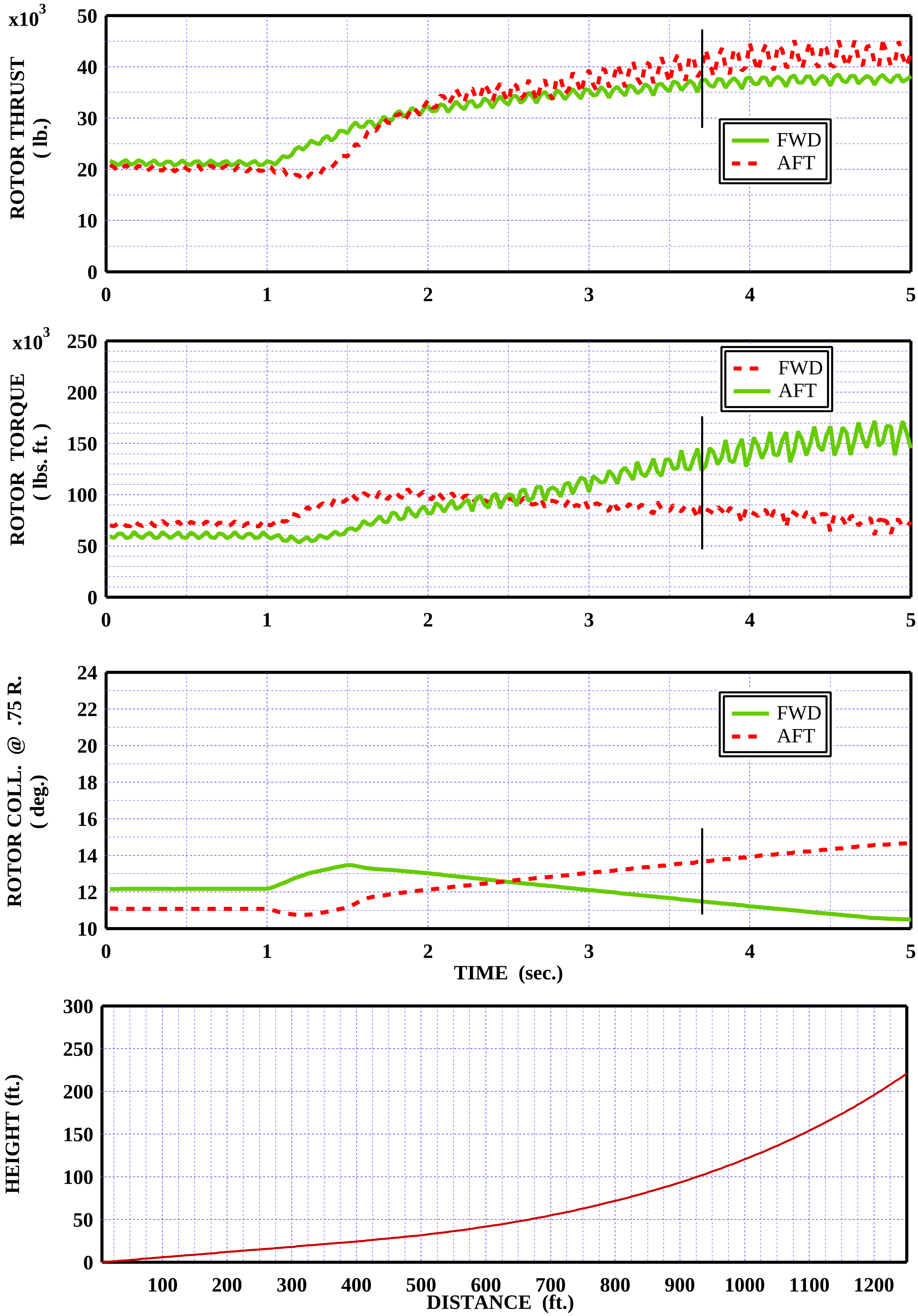


Fig. 1(b) HC MK II SIMULATED CONTROL RESPONSE

INITIAL TAS = 135 Kts.
INITIAL ROC = 1000 fpm.

CASE A 24

CYCLIC TRIM: ADVANCED



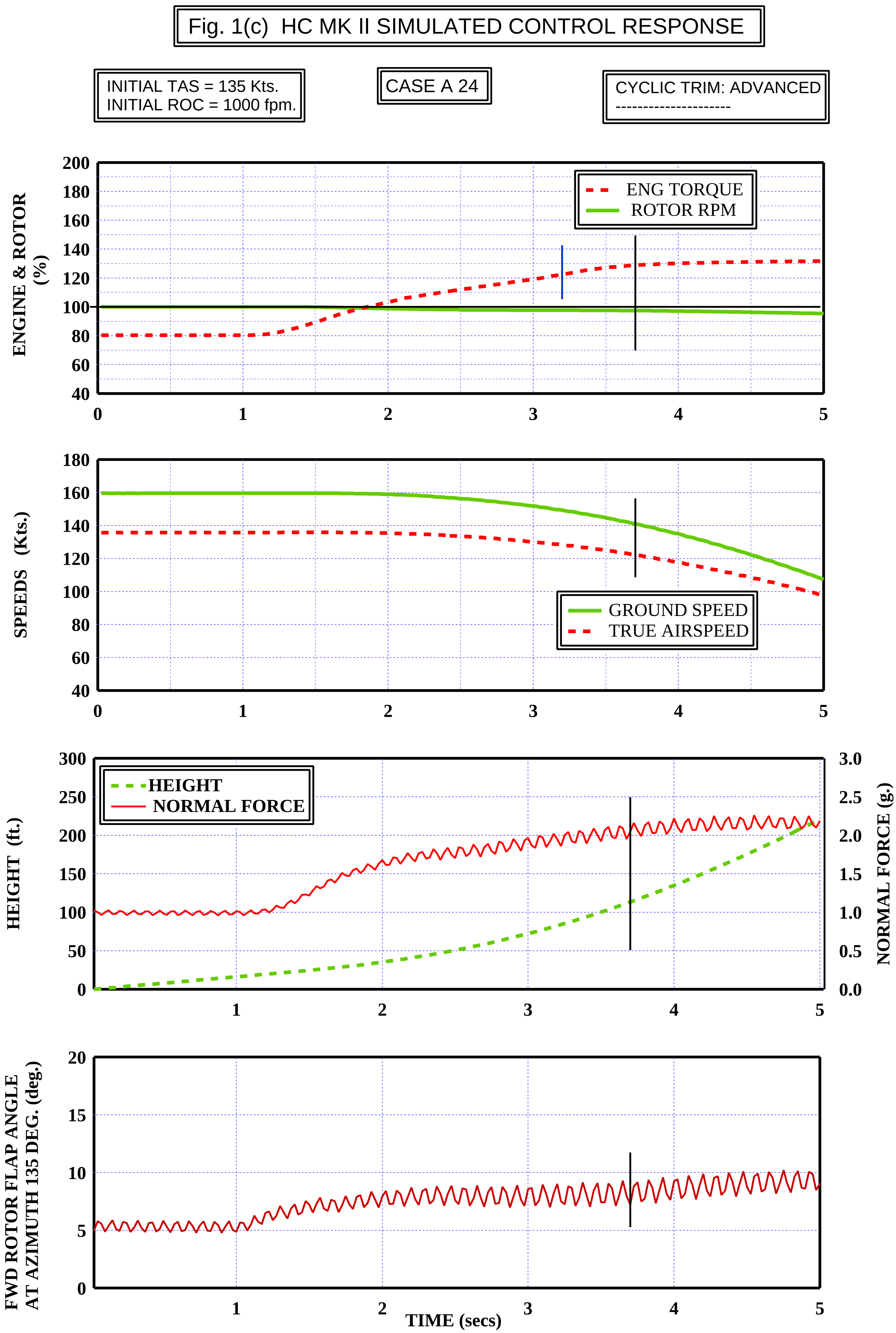


Fig. 1(c) HC MK II SIMULATED CONTROL RESPONSE
INITIAL TAS = 135 Kts.
INITIAL ROC = 1000 fpm.
CASE A 24
CYCLIC TRIM: ADVANCED
ENGINE & ROTOR (%)
ENG TORQUE
ROTOR RPM
SPEEDS (Kts.)
GROUND SPEED
TRUE AIRSPEED
HEIGHT (ft.)
NORMAL FORCE (g.)
HEIGHT
NORMAL FORCE
FWD ROTOR FLAP ANGLE AT AZIMUTH 135 DEG. (deg.)
TIME (secs)

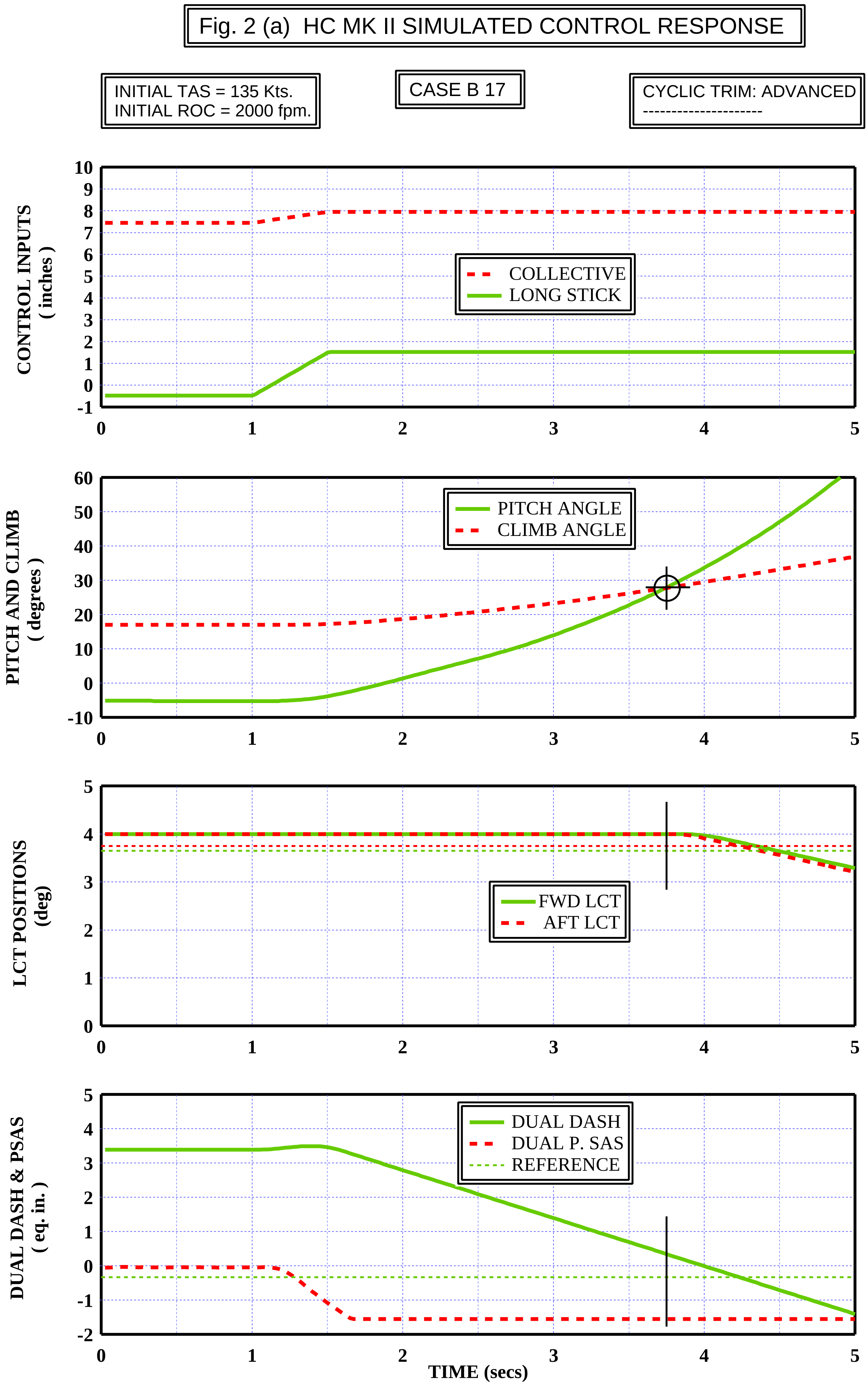


Fig. 2 (a) HC MK II SIMULATED CONTROL RESPONSE
INITIAL TAS = 135 Kts.
INITIAL ROC = 2000 fpm.
CASE B 17
CYCLIC TRIM: ADVANCED
CONTROL INPUTS (inches)
COLLECTIVE
LONG STICK
PITCH AND CLIMB (degrees)
PITCH ANGLE
CLIMB ANGLE
LCT POSITIONS (deg)
FWD LCT
AFT LCT
DUAL DASH & PSAS (eq. in.)
DUAL DASH
DUAL P. SAS
REFERENCE
TIME (secs)

Fig. 2 (b) HC MK II SIMULATED CONTROL RESPONSE

INITIAL TAS = 135 Kts.
INITIAL ROC = 2000 fpm.

CASE B 17

CYCLIC TRIM: ADVANCED

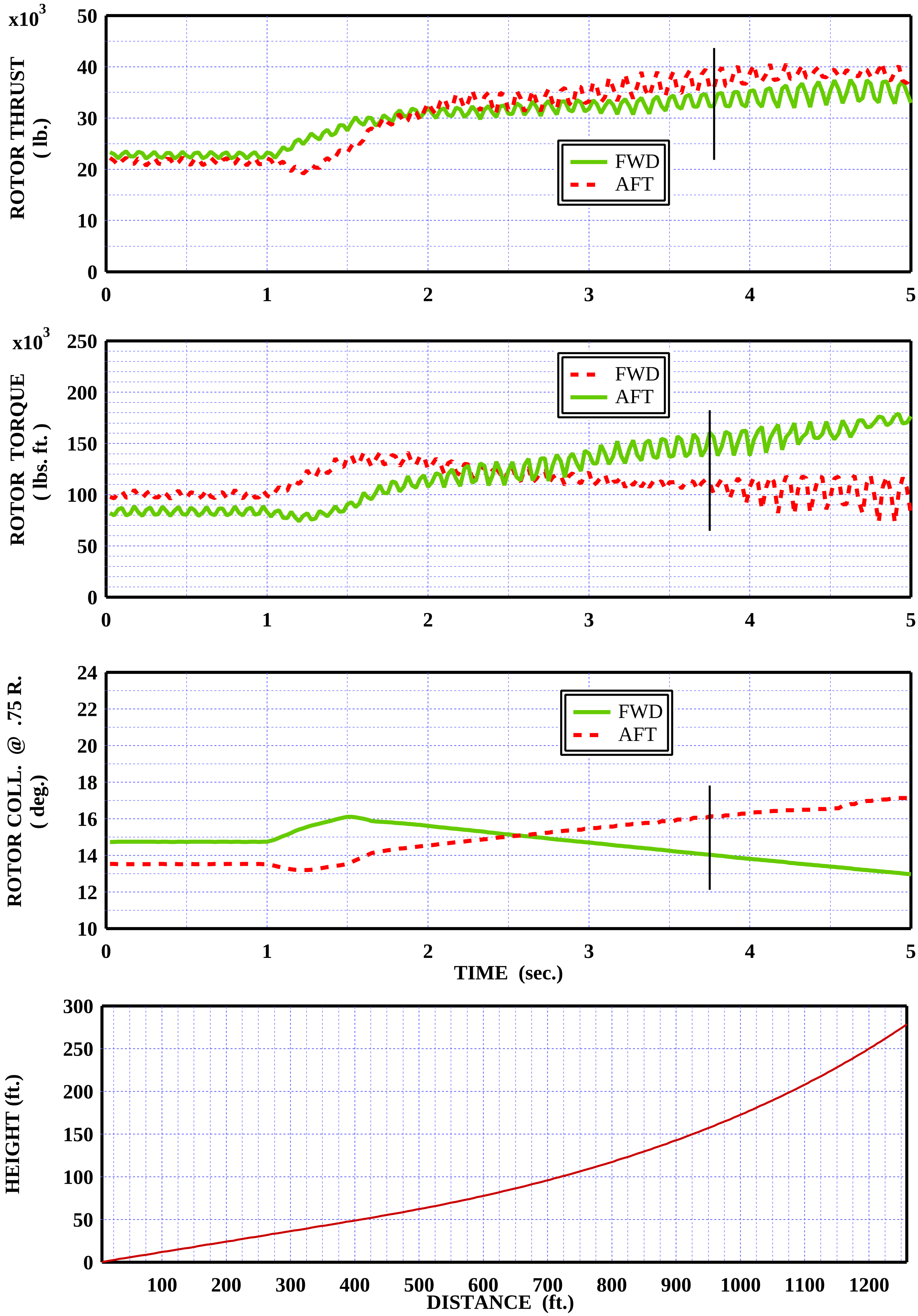


Fig. 2 (c) HC MK II SIMULATED CONTROL RESPONSE

INITIAL TAS = 135 Kts.
INITIAL ROC = 2000 fpm.

CASE B 17

CYCLIC TRIM: ADVANCED

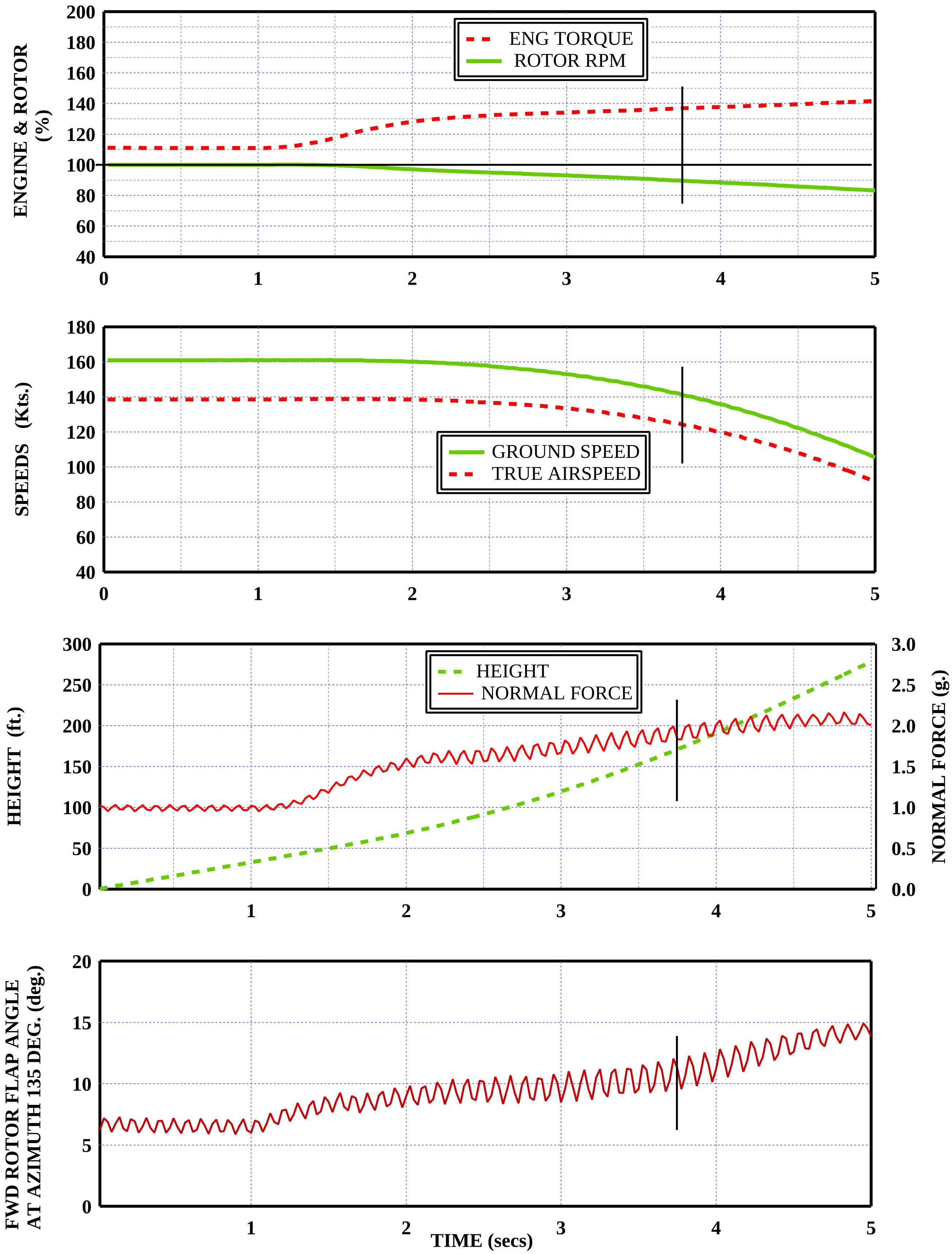
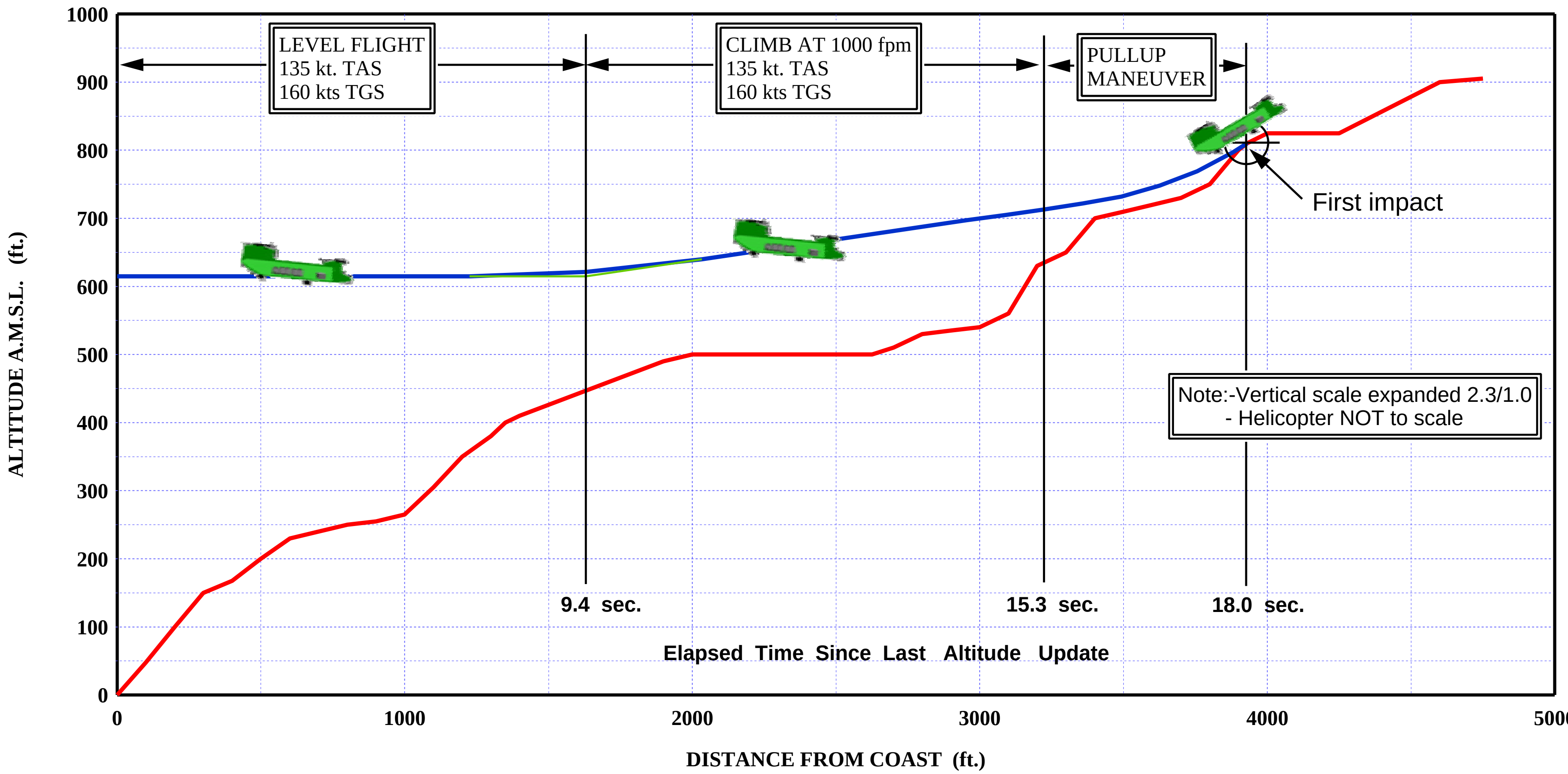


Figure 3
HC MK 2 ZD 576 MULL OF KINTYRE
FLIGHT PATH AND TERRAIN PROFILES (CASE A-24)



LAST ALTITUDE UPDATE
AT - 910 ft.

PULLUP MANEUVER	
Initial A/S	135 kts
Initial R/C	1000 fpm
Collective Input	0.5 in.
Long. Input	2.0 in.

CONDITIONS AT IMPACT	
Airspeed	122 kts.
Altitude	810 ft. AMSL
Climb Rate	4130 fpm.
Pitch Attitude	26 deg.
Climb Angle	16 deg.

Fig. 4(a) HC MK II SIMULATED CONTROL RESPONSE

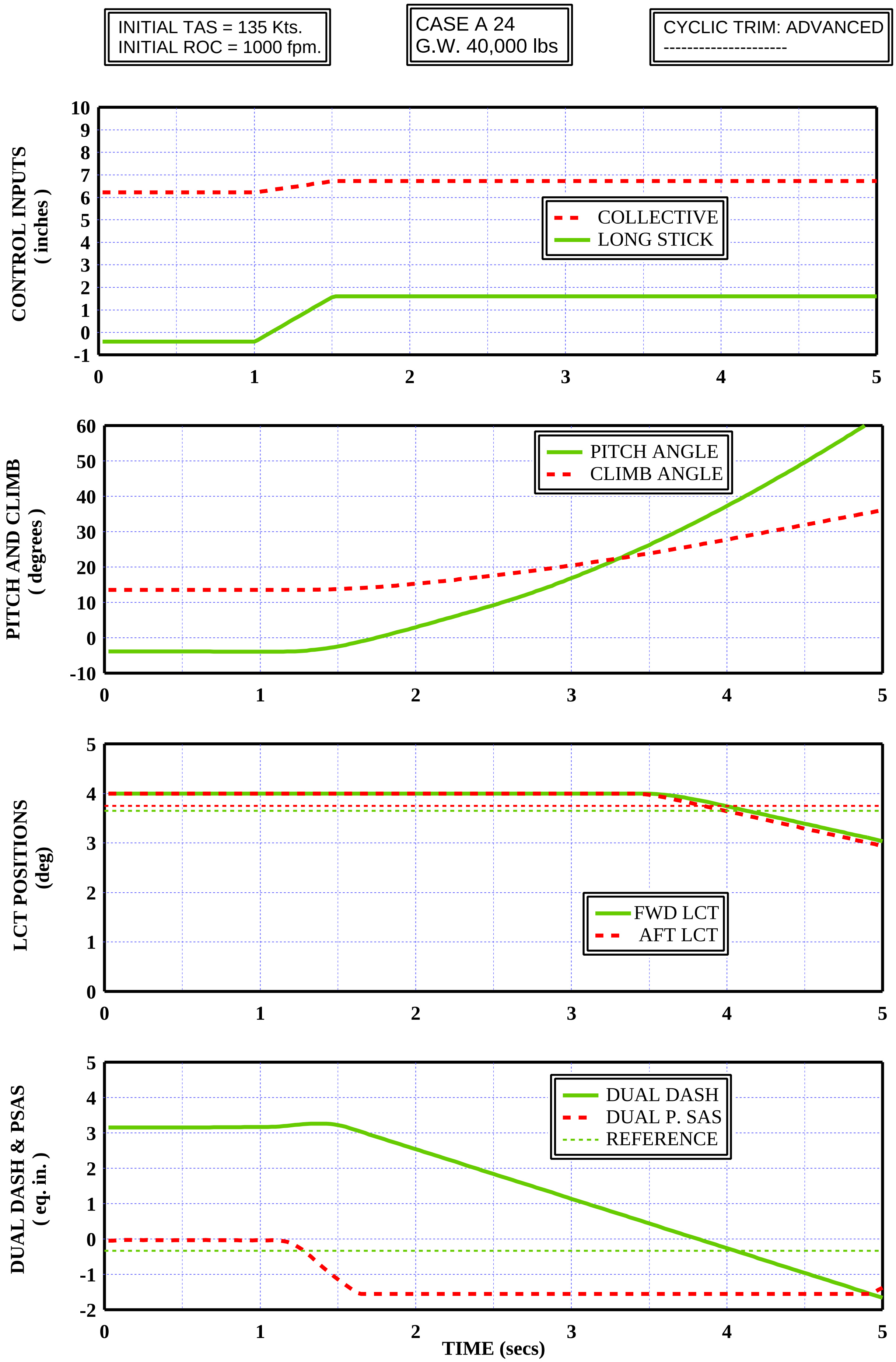


Fig. 4 (b) HC MK II SIMULATED CONTROL RESPONSE

INITIAL TAS = 135 Kts.
INITIAL ROC = 1000 fpm.

CASE A 24
G.W. 40,000 lbs

CYCLIC TRIM: ADVANCED

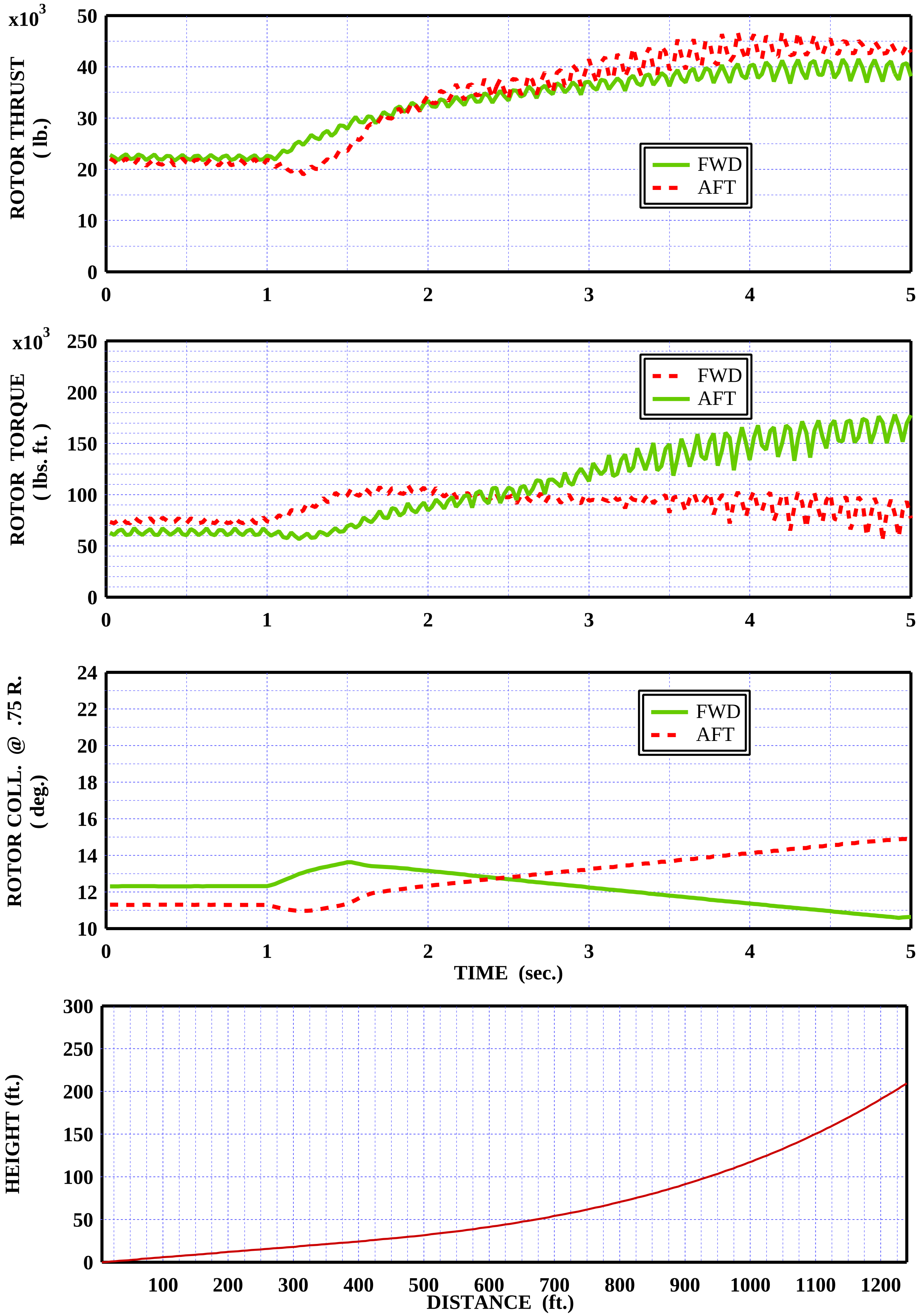


Fig. 4 (c) HC MK II SIMULATED CONTROL RESPONSE

INITIAL TAS = 135 Kts.
INITIAL ROC = 1000 fpm.

CASE A 24
G.W. 40,000 lbs

CYCLIC TRIM: ADVANCED