DEPARTMENT OF CIVIL AVIATION MALAYSIA

AIRCRAFT ACCIDENT REPORT NO. 03-0282

BELL 212 9M - AWW GENTING HILL RESORT, PAHANG MALAYSIA, 25TH FEBRUARY, 1982

OPERATOR	:	Genting Helicopter Services, Kuala Lumpur.		
AIRCRAFT	:	TYPE	:	BELL HELICOPTER
		MODEL	:	212
		NATIONALITY	:	MALAYSIAN
		REGISTRATION	:	9M - AWW
PLACE OF ACCIDENT	:	1734 meter hill slope 337 meters from the Ulu Kali Helipad, Genting.		
DATE AND TIME OF ACCIDENT	:	25th February, 1982, at 1834 hours. All times in this report are Local (+ 8 hours GMT).		

SYNOPSIS

An Inspector of Accidents was notified of the crash at about 1900 hours on 25th February, 1982.

Notification was made by the Duty Watch Supervisor, Kuala Lumpur International Airport, Subang, Selangor.

Investigation of the accident was conducted by the office of the Chief Inspector of Aircraft Accidents, Ministry of Transport Malaysia and assisted by an ICAO recommended Aircraft Accident Investigator from Canada.

2/...

On the Thursday afternoon 25th February, 1982, a Bell 212 helicopter bearing the registration 9M-AWW with eleven persons on board departed from the high altitude Ulu Kali helipad, (in front of the main Genting Resort hotel) for the Segambut helipad in the Kuala Kumpur city valley. The flight was a scheduled flight that service the Genting Hill Resort on an hourly basis. Though all such flights were to be conducted in Visual Meteorological Condition (VMC), approval were granted for Instrument Meteorological Condition (IMC) transit and arrival for the Kuala Lumpur International Airport. On the day in question the flight was delayed slightly because of strong wind conditions prevailing over and around the high altitude helipad. It finally took off at about 1834 hours OMT and was seen to enter cloud almost immediately. The aircraft crashed on a steep embankment about 337 meters North West of the helipad about 22 seconds after lift off.

The aircraft was totally destroyed and there were no survivors.

1. FACTUAL INFORMATION

1.1. HISTORY OF THE FLIGHT

1.1.1. On Thursday 25th, the pilot was rostered to fly 10 sector shuttles from the low level helipad in KL City to the high altitude pad in the Genting Resort. He was detailed to fly one of the two Bell 212 helicopters operated by Genting Helicopter Services, bearing registration 9M-AWW.

1.1.2. The particular helicopter had already flown 2.1 hours that day. The pilot who flew it had not reported or indicated any unserviceability.

1.1.3. The accident flight was scheduled to be the last flight for the day. It had previously flown in passengers from Kuala Lumpur International Airport and landed onto the highest of the 5 helipads.

1.1.4. It was noted that during its approach for that particular arrival, the wind was from the East and gusting to about 20kts. It was also noted that together with the wind were continuous patches of stratus cloud moving through the helipad.

1.1.5. After landing, the engine was shutdown and passengers were dispatched vide a motor coach to the main Resort Hotel about 66 meters to the South of the helipad. The pilot also took the coach ride to the hotel. It was noted that the pilot voiced his concern of the relatively gusty wind condition and had opted to delay his departure.

1.1.6. At about 1800 hrs. the pilot together with 10 passengers prepared themselves for departure although there was no significant change in the weather condition.

1.1.7. The pilot, being the sole crew of the flight was in the pilot's position i.e. in the right cockpit seat. Seated in the left cockpit seat was an employee of the holding company of Genting Helicopter Services. The said employee was not a pilot thus was not familiar to the technicality of aircraft operations. However, the seat he occupied was a fully functional flying station. The rest of the passengers were seated in the cabin, filling up all the seats in row 3, 3 seats in row 2 and one seat in the starboard facing row.

1.1.8. The start up were noted to be normal by all witnesses. The pilot then lifted off, hovered just off the pad and at the same time turned into wind. It translated into wind on the 075° heading at time 1834. The helicopter was seen to make a left climbing turn and shortly thereafter disappeared into a layer of cloud.

1.1.9. The helicopter was seen by a witness, who was walking along a road track below the helipad, as suddenly appearing out of a cloud base and crashing into a steep embankment just ahead of him. The helicopter was engulfed in flames almost immediately and was almost completely burnt out inspite of efforts by rescue parties to extinguish the flame.

1.2. INJURIES TO PERSONS

Injuries	Crew	Passengers	Others
Fatal	1	10	0
Serious	0	0	0
Minor/None	0	0	0

1.3. DAMAGE TO AIRCRAFT

1.3.1. The basic fuselage was totally consumed by post-impact fire. However, the tail boom aft of station 320 was reasonably untouched by the fire. Some airframe components such as doors were thrown clear of the fire area and therefore only suffered impact damages.

1.3.2. Components manufactured of high melting point materials survived the post-impact fire. In fact, quite a large number of important system components survived and were sufficiently recogniseable for detailed inspection and analysis.

1.4. OTHER DAMAGES

1.4.1. The impact area being fairly confined resulted in an almost negligible damage to the natural vegetation. As the main impact point was on a ledge above a road embankment, soil errosion due to exposure of vegetation was also negligible.

1.5. PERSONNEL INFORMATION

1.5.1. The Pilot-in-Command held an Airline Transport Pilot's Licence No. 490/H which was issued on 19th January, 1980. He was the sole operating crew of the helicopter when the accident occurred. At the time of the accident he had approximately 4020:00 hours total flying time, of which 3066:40 hours were gained from his flying whilst in service with the Royal Malaysian Air Force. A large portion of his Service flying was on the Sikorsky S61A4.

He started flying the Bell 212 on 12th January, 1981 after he had left the RMAF and joined Genting Helicopter Services. The conversion took 8 hours 35 minutes (not including the 1 hour 20 mins. flight test). His performance during the conversion was entirely satisfactory. Up to the day of the accident he had accumulated approximately 954 hours flying time on the Bell 212. He held a current Instrument Rating and was not due for renewal until 9th July, 1982. His ATPL was valid until 28th February, 1982.

1.5.2. For the month of January 1982, he flew a total of 62:15 hours. His last scheduled flight was on 23rd February 1982 when he flew a total of 4:40 hours, during which he also logged 1:30 hours of actual Instrument Flying. In fact, throughout the month of February he had been logging actual Instrument Flying in most of the flying days. He also carried out one night schedule on the 6th of February. Up till the day of the accident and for the month of February, he had 9 non flying days.

1.6. AIRCRAFT INFORMATION

1.6.1. 9M-AWW was a Bell 212 helicopter bearing airframe Serial No. 30898. The helicopter was owned and operated by Genting Berhad whilst maintenance was looked after by Heli Orient PTE Limited. The Certificate of Registration (M 347) was issued to Genting Berhad on 18 December 1978. The Certificate of Airworthiness (M 301) initially issued on 27 December 1978 was last renewed on 27 December 1981.

1.6.2. Two Pratt and Whitney PT6T-3 twin pac engines were fitted. The left hand engine Serial No. 60688 had done 4725.46 hours since new, 757.10 hours since last major overhaul and 25.12 hours since last 100 hours. inspection. The right hand engine Serial No. 60426 had been operated for 4614.28 hours since new, 1747.40 hours since last major overhaul and 25.12 hours since last 100 hours inspection.

1.6.3. The airframe had completed 4625.12 hours since new and 757.10 hours since last major overhaul. The authorised overhaul hours is 1000 hrs. and it does not have a retirement life.

6/...

1.6.4. An examination of the helicopter maintenance document confirmed that the helicopter had been maintained in accordance with an approved maintenance schedule. All significant defects had been investigated and rectified prior to the accident and all applicable Airworthiness requirements had been complied with.

1.6.5. The helicopter's estimated all up weight at the time of the accident was 4283kg. The centre of gravity (CG) position was 341 cm. from datum point. The maximum authorised take off weight was 5090.9 kg.

1.6.6. The fuel in use was Jet AI.

1.7. METEOROLOGICAL INFORMATION

1.7.1. General Situation: On 25th February, 1982 at 1834 OMT, the Genting Highlands area was under the influence of low cloud with reduced visibility. Observations in the area reported an intermittent total cloud cover with a base at ground level. Although wind at the nearest Meteorological Station (Subang) was 270 degrees at 10 knots, the effects of terrain in the area of the accident site could have caused gusty conditions with stronger winds.

1.7.2. Witnesses Report: Statements from witnesses indicated that low cloud, poor visibility and strong winds were present at the time of the accident. It was also reported that the impact site situated at about 337 meters from the helipad was not visible during the take off and the helicopter disappeared in cloud approximately 10 seconds after lift off from the helipad. Although most witnesses agreed that the weather was poor with strong gusty winds, there were indications that visibility was good above the mountain tops.

1.7.3. Report from Meteorological Department: Between 1700 OMT and 2000 OMT the Meteorological Department made weather observations by means of radar and satellite. Rain was reported at Subang International Airport, however, there was no reported thunderstorms over Genting Highlands. It was forcasted that Genting Highlands would be experiencing a temperature of 10°C and a wind condition of significant strength.

1.8. AIDS TO NAVIGATION

1.8.1. Genting Highlands did not have any navigational aid.

1.8.2. Automatic Direction Finder (ADF), DME and VOR/ILS formed part of the navigational equipment available in the helicopter. There was no report of any malfunction to any of these aids before the accident.

1.9. COMMUNICATION

1.9.1. As the Ulu Kali pad was surrounded by several high mountains, clearance for flight departure was normally obtained immediately after lift off.

1.9.2. The helicopter crashed at approximately 22 seconds after lift off. No distress call was heard from the helicopter prior to impact.

1.10. AERODROME INFORMATION

1.10.1. There are 5 helicopter pads situated at and around the vicinity of Genting Highlands Resort. They are the west pad, East pad, Mile one pad, Sri Layang pad and Ulu Kali pad.

1.10.2. The Ulu Kali pad, which was the point of departure, is situated on a saddle approximately 1636 meters above mean sea level. The pad is between the new hotel some 200 meters to the south and 3 micro wave towers some 1840 meters to the north. Both the tops of the hotel/micro wave towers are at approximately 1727 meters above mean sea level. The hilly terrain slopes gradually from the micro wave towers to the north and to the hotel to the south. These physical features also form a funnelling effect for winds.

1.10.3. On 18 August, 1979 the Ulu Kali pad was cleared for day and night operations. Due to its locality, several conditions were imposed so as to ensure a safe operation of the pad. These included visual flight rule minimas of 1 nm. visibility, clear of cloud and in sight of ground.

1.10.4. There was no written standard procedure for departure from the Ulu Kali pad.

1.11. FLIGHT RECORDERS

1.11.1. Not applicable.

1.12. WRECKAGE AND IMPACT INFORMATION

1.12.1. The helicopter took approximately 22 seconds from lift off to impact point. From impact evidences it was determined that the entry heading was 291 degrees and the final resting heading was 350 degrees. The helicopter was also in a severe nose down attitude and from the witness marks found on the vegetation it was estimated that the pitch was 55 - 65 degrees nose down and having a left roll of about 50 - 55 degrees.

1.12.2. A majority of the cabin and fuselage were destroyed by fire. The break-up characteristics were consistent with evenly distributed frontal loading.

1.12.3. The deformation of the crew seats indicated that the primarly impact loads were from the front and left. The door structures had received only moderate exposure to the fire and apparently had separated during impact. Both the passenger-cargo doors were deformed in a similar manner which was consistent with even frontal loading. The left hand door had suffered more extensive damage than the right hand door. The nature of the deformation would indicate slightly more severe impact forces from the front than right. Such loading was consistent with the helicopter contacting the slope nose down and rolled left. Further evidence of impact loading from the front and left was indicated by the deformation sustained by the pitot tubes located on the nose and the right ballast weight.

1.12.4. The impact forces had caused partial separation of the vertical fin from the tail boom. The fin structure was heavily damaged. Separation of the fin had occurred directly forward indicating virtually symmetrical frontal loading on the fuselage. The only evidence of side loading was minor buckling of the right forward spar cap at the area of spar failure. The symmetry of the loading was most clearly shown by compressional buckling at the aft extremity of the tail boom. The inertia of the fin with the tail rotor gearbox, hub and blades, caused deflection of the fin forward at impact. The compressional buckling on both the right and left hand sides of the tail boom was symmetrical. With the exception of the fin separation the tail boom had experienced relatively minor structural damage. The absence of tail boom breakup or structural failures was further evidence of the lack of side loading.

1.12.5. The front cross tube of the skid gear assembly had pulled the attachment fittings away from the fuselage. Both skids remained attached to the cross tube. The direction of the failure indicated the loading was primarily frontal in nature. The aft cross tube was heavily fire damaged and had separated from both the right and left skids. The left skid fractured as a result of forward loading while the right skid failed under the influence of loads aft and from the left. The left skid fractured just forward of the frontal cross tube. The directionality of the fracture was consistent with frontal loading with a component from the left. The right skid failed just aft of the front cross tube. The mode of failure was bending as a result of frontal loads.

1.12.6. One main rotor blade had imbeded in the side of the bank above the impact location. The remaining blade had come to rest over the fuselage and was destroyed by fire. Both blades were still attached to the grips and retained by the drag braces. The induced failure of the blade not exposed to the fire was located approximately 3.048 meters from the root end. The inboard 3 meters of this blade remained intact.

1.12.7. The main retention bolt hole was not deformed nor was the grip plate. There was no significant distoration or deformation in the area of the drag link bolt hole or plate. No delamination or failure of any doublers plates was observed. Some relatively minor compressional buckling of the blade pocket skin was apparent near the root and adjacent to the trailing and leading edge spars. The onsite evidence was that this main rotor blade had experienced a sudden stoppage. The degree of compressional buckling observed is not consistent with a sudden stoppage of a main rotor blade rotating under a high power condition.

1.12.8. A spar failure had occurred approximately 152.4cm inboard from the tip. There were no leading edge impact marks associated with the spar failure. The nature of the spar failure indicated that the loading causing fracture was primarily column loading. Such loading would occur by loading the tip of the blade during impact.

1.12.9 The blade pocket structure on the outboard portion of the blade had separated from the spars and fractured into numerous sections. It is not known how much of this damage occurred during the accident sequence and what damage occurred as a result of the post accident salvage operations.

1.12.10. The second main rotor blade was totally destroyed. Only the root end remained. The blade appeared to have failed approximately 91.44 cm. from the root. The main spar was heavily deformed in a manner which indicated that failure was the result of excessive up bending loads. As was the case with the unburned blade this suggests end-wise or column loading during ground contact.

1.12.11. The main rotor hub and mast assembly were not exposed to excessive temperatures. Protection from the post impact fire was likely provided by a covering of loose earth. A detailed inspection of the main rotor hub and mast assemblies indicated the following:-

(1) The stablizer bar arms while deformed remained intact.

(2) Both mixing levers were attached to the stabilizer bar assembly with no evidence of pre-impact bearing seizure or failure.

(3) The four bolts securing one stabilizer support to the trunnion had failed in what appeared to be in an overload mode. The bearing of the second support had pulled away from the stabilizer assembly. This failure had occurred at impact. It was evident from the damage to the supports that excessive deflection of the stabilizer bar assembly had occurred.

(4) Both pitch change links were intact and had been correctly installed.

(5) The white blade pitch horn was attached to the blade grip however the red pitch horn had separated. The nature of the failure indicated that separation had occurred as a result of impact.

(6) Both damper link tubes had failed. One link had failed in tension while the other had been subjected to excessive compressive stresses. The dampers had remained intact and secured to the mast.

(7) The pitch control tubes between the rotating scissors assembly and mixing levers had failed due to excessive column loading.

(8) The mast nut was properly installed and torqued.

(9) The main rotor trunnion, yoke and pillow block bearing were intact.

(10) The blade grips and drag braces were free of unusual damage.

(11) One scissors arm had pulled away from the swash plate otherwise drive continuity from the swash plate through to the pitch control tubes was maintained.

(12) The swash plate assembly was intact and rotated without interference.

(13) The swash plate support arms which support the ring assembly through the gimbal were fractured in an overload mode. There was no evidence to suggest that failure had occurred prior to impact.

(14) The sleeve assembly was intact.

(15) The main rotor mast assembly remained intact and properly secured to the hub. A heavy indentation due to contact with the static stop of the hub was apparent on one side of the mast. There was no evidence of repeated excessive main rotor blade flapping. The damage observed indicated that one excessive

flapping excursion of the main rotor blade had occurred. This deformation as well as other damage to the main rotor components indicated that one main rotor blade had been subjected to column loading during impact.

1.12.12. The physical evidence indicated that the main rotor hub and mast assemblies were intact and functional at the time of impact. There was no evidence of a premature or in-flight failure of any component in these assemblies.

1.12.13. During the course of the examination of the wreckage, the engines were transported to the Heli-Orient facilities in Singapore for teardown. Both engine assemblies had been exposed to extreme temperatures which had destroyed most of the components fabricated from low melting point alloy. As a result of this damage components such as ignition excitors, fuel pumps, fuel controls, filters, pneumatic lines, fuel and oil lines, oil-to-fuel heaters, etc. were either unavailable for analysis or in a condition which precluded detailed inspection. The accessory gearbox assemblies as well as the combining gearbox were destroyed.

1.12.14. Left Engine. The compressor inlet case had melted away, however, the gas generator case and forward fire seal remained. The No. 1 bearing support and first stage compressor rotor were exposed. The extensive fire damage precluded rotation of the compressor rotor. A visual examination of the compressor assembly to the extent possible indicated that the first stage rotor was intact. No evidence of ingestion of foreign debris was noted. The first stage compressor blades were not deformed indicating no excessive rotational inference with the case. The lack of excessive tip rub would indicate that the integrity of the No. 1 and No. 2 bearings had been maintained with little or no radial or lateral displacement of the compressor rotor assembly. While it was not possible to inspect the second and third axial stages or the centrifugal stage, the observation that the turbine wheels showed no indications of foreign object damage indicated that the compressor components were intact.

13/...

1.12.15. Right engine. As was the case with left engine, the right engine had been exposed to servere temperatures during the post impact fire. The engine external components were either destroyed or not recovered. The axial section of the compressor had been completely destroyed. The damage in this area was more severe than that noted with the left engine. The nature of the damage and lack of suitable tooling prevented removal of the centrifugal compressor from the engine. The gas producer turbine wheel was undamaged. There was no evidence of foreign object damage or operation with severe overtemperature condition. The lack of foreign object damage would indicate that the axial compressor had been intact at impact. This was supported by lack of debris within the combustion lines. No significant blade tip rub was observed. The gas producer turbine wheel was seized precluding removal to allow inspection of the first stage nozzle or vane assembly. The power turbine wheel was intact with no evidence of foreign object damage or operation with excessive gas path temperatures. The turbine blade outer shrouds were lightly rubbed as were the trailing edges of the blades adjacent to the shrouds. The power turbine shaft/main input driveshaft assemblies were intact with no evidence of pre-impact distress. The Nos. 3 and 4 bearings were functional at the time of impact. The power turbine shaft was displaced aft approximately 0.0254cm. An analysis of the right engine indicates that the compressor and turbine section were mechanically capable of normal operation when the helicopter impacted. The lack of mechanical damage and ingestion of debris through the compressor and into the turbine and combustion sections suggests a low power condition. There was however no method of making an accurate determination of the power level at the time of impact.

1.12.16. The engines are fitted with individual accessory gearboxes which provide gas producer drive for the fuel pump, fuel control, starter-generator, techometer generator and

The bleed valve assembly had melted and thus was not available for analysis. The gas producer turbine wheel was in excellent condition with no evidence of foreign object damage, tip rub, or other significant mechanical damage with the exception of some minor rub at the blade roots on the training edge side. The rub was the result of contact with the hub of the power turbine nozzle assembly. The splined coupling between the gas producer wheel and impellor was free of abnormalities. There was no evidence of abusive operation or that an overtemperature condition has been experienced. A non-uniform staining pattern was noted on the gas producer wheel. The most likely source of the staining was burning of pooled fuel while the turbine wheel was not rotating. While there was no loss of turbine blades the blade tip shrouds showed heavy rub. This rub pattern was continuous and even around circumference of the wheel. The corresponding rub on the stationary shroud was assymetric and fairly localized at a location corresponding to the bottom sector of the engine. There was no significant rub 180°to location of maximum rub. Evidence of rub was also observed at the training edge blade tips and leading edge blade platform areas. The power turbine nozzle was intact with no evidence of foreign object damage. No vane degradation that would be indicative of a prolonged overtemperature condition was observed. Some minor rub was apparent on the exit side hub. The gas seal was distorted. Localized carbonaceous deposits had accumulated on some vanes through an arc of approximately 180°. The carbon build-up was likely the result of combustion of pooled oil during the post impact fire. The power turbine shaft, main input driveshaft, No.3 and No. 4 bearings were intact with no evidence of distress. The No.4 bearing flange had been pulled away from its normal position adjacent to a shoulder on the power turbine shaft. The physical evidence indicated that the left engine did not experience an in-flight failure of the **turbine** or compressor sections. A detailed assessment of the engine accessories could not be completed due to fire damage or unavailability of components. The rotational damage observed, particularly in the power turbine indicates the engine was rotating at impact. It was not possible to accurately determine the level of power being developed at the time of impact.

engine oil pump. The accessory gearbox cases from both engines had been completely destroyed. The gearbox components were located loose at the crash site. The components were not matched with specific engines. One idler gear and oil pump drive had been exposed to temperatures sufficiently high so as to cause significant melting. An analysis of the geartrain components and associated bearings indicated that drive continuity likely existed at the time of impact. There was no evidence of a premature gear or bearing failure. The external components driven by the gearboxes (i.e. fuel control oil and fuel pumps, etc.) were either destroyed or not recovered in a condition that would allow analysis. Power output from both engines is used to drive a single reduction gearbox (combining gearbox). The combining gearbox has a single output shaft which is secured to the main driveshaft to transmit power to the transmission. Turbine speed reduction is accomplished by means of a 3 stage gear train, the third stage of which meshes with the output gear shaft. One-way or spag clutches are incorporated in the third stage to allow decoupling of an inoperative engine from the gear train should an engine failure be experienced. The combining gearbox case being fabricated from a low malting point alloy was destroyed. Only a portion of the gear train was available for analysis. One first and one second stage gears were not recovered. The power turbine accessory gear train elements including the Nf governor, Nf techometer generator, and one blower drive were not recovered. The section of the gear train which were recovered were free of mechanical damage other than fire damage. There was no evidence of gear or bearing failure that could have been a casual factor in the accident.

1.12.17. The main driveshaft assembly transmits engine power from the combining gearbox to the input quill of the transmission. The driveshaft was slightly bent, however, there was no evidence of torsional buckling as one might expect with a sudden stoppage at high power. The combining gearbox and male and female couplings were intact. The combining gearbox side male coupling (aft male coupling) remained attached to the driveshaft by the lock nut. The centering spring and lock clip were missing. Both the male and female combining gearbox and couplings were free of mechanical damage. The male coupling at the transmission end had fractured through the flange in an overload mode. The coupling teeth were mechanically damaged, but not sheared or smeared in a manner to support operation at excessive temperatures. The transmission end female coupling had failed with only one small remnant remaining on the input quill. None of the female coupling segments were recovered. There was little evidence of rotational scoring on the driveshaft itself indicating little post impact rotation. The lack of rotational interference damage on the driveshaft, the failure of the male coupling, the lack of torsional deformation of assembly and the failure of the transmission end female coupling indicates that the assembly was in place and capable of transmitting power at the time of impact. The physical evidence suggests that the engines were not operating at high power.

1.12.18. The purpose of the main transmission assembly is to provide speed reduction of engine power, change the direction of power transmission and provide drive to the main rotor mast and tail rotor drive systems. The transmission external components include the main rotor tachometer generator, two hydraulic pumps and the transmission oil pump. Impact forces had caused failure of the transmission main case. The loads causing case failure were from the front left. The top case remained intact however the sump and support cases were destroyed in the post crash fire. There was no evidence of in flight distress within the transmission assembly. The main spiral bevel gear and planetary assemblies were intact. The input quill was removed, cleaned and inspected. The input pinion was free of tooth breakage or abnormal meshing patterns. There was no evidence of bearing distress or clutch malfunction. The four main transmission mounts were recovered at the accident site. There was no evidence of a premature mount failure. The geartrain normally located in the sump case including the tail rotor output quill, No. 1 hydraulic pump drive quill, off-set gear and oil pump drive were fire damaged. No abnormal gear wear or degradation (except fire damage) of the bearings was noted indicating that these components were functional at the time of the accident. The mast bearing showed no evidence of overheating or any other discrepancies.

1.12.19. The tail rotor drive train consists of an output quill at the transmission, six driveshaft segments, an intermediate (42^{o}) gearbox. As had been pointed out the tail rotor drive output quill showed no evidence of a premature failure. The three most forward tail rotor driveshaft segments were destroyed with the exception of the hangarbearing at the aft end of the second segment. Loose couplings from the destroyed shaft segments were recovered. No evidence of in-flight distress of these couplings was observed. The clamps, hangar bearing assemblies, etc. associated with the fourth and fifth driveshaft segments were intact and rotated without interference. It was significant that there was no evidence of rotational interference between the driveshaft segments and the tunnel. No torsional buckling of the driveshaft segments was observed. Torsional twisting of the driveshaft commonly occurs with a sudden stoppage of either the main drive train or tail rotor system when under power. There was no evidence of main rotor contact with the driveshaft tunnel or shaft segments. The intermediate gearbox (42^{o}) was free of significant mechanical damage. The input coupling to the 42^{o} gearbox was intact. The driveshaft segment between the 42^{o} and 90^{o} gearboxes had fractured just above the 42^{o} output. This fracture was due to excessive compressive forces which were developed by the forward deflection of the tail fin at impact. The compressive loading sheared the coupling rivets allowing separation of the curvic coupling from the shaft. There was evidence of very little post failure rotation of the failed drive-shaft. The tail rotor gearbox (90^{o}) was intact. The assembly rotated without unusual interference indicating the absence of internal distress.

1.12.2 . The tail rotor blades remained attached to the hub and had suffered relatively minor damage. Paint scuffing indicated some rotation was occurring at impact. One blade tip had separated. The second blade had buckled in a spanwise direction. There was no evidence of tail rotor blade contact with the vertical fin structure.

1.12.21. The Bell 212 is fitted with a cyclic control system for lateral and fore/aft control, a collective system for vertical control and an anti-torque or tail rotor control system for directional control. These three systems are operated through hydraulic servo actuators to minimize control forces. A synchronized elevator is coupled with the cyclic control system to enhance pitch controability.

1.12.22. An analysis of the cyclic system was limited by the availability of components and the condition of those which were recovered. Most of the underfloor controls were either destroyed by fire or not recovered. Continuity of the cyclic controls from the forward side of the servos through to the swash plate assembly was confirmed. Both the pilot and co-pilot cyclic control stick assemblies separated from the fusealage at impact. The pilot's cyclic fractured adjacemt to the friction adjustment. All observed link failures appeared to be overload in nature and impact related. The gimbal assembly moved freely as did the remaining available components. An analysis of the stick assembly was made to determine if any witness marks were present to indicate the cyclic position at impact. One witness mark was noted in the full-aft cyclic position. This was not considered to be an accurate indication of the cyclic position at impact. The co-pilot's cyclic was sheared at the support assembly due to excessive forces forward. Witness marks and mechanical damage corresponding to a full aft cycle and a full right cyclic position were observed. It could not be determined if these impact marks reflected the control position at impact or were the result of cyclic control position during the aircraft break-up.

19/...

1.12.23. Both the pilot and co-pilot collective sticks had separated and were recovered. The controls between the collective sticks and collective servo were either destroyed or not recovered. There was control mechanical continuity through the servo to the collective lever on the main rotor mast. It was not possible to determine the collective position at the time of impact.

1.12.24. As was the case with the other control systems, only portions of the tail rotor control system were recovered. The under floor controls as well as those in the forward section of the tail boom were destroyed. The control pedals and adjuster assembly separated from the aircraft at impact. Both left pedals had fractured while the right pedals remained intact. The control tubes from the pedals to the adjustment assembly failed in column loading during impact. An inspection of the available anti-torque controls in the section of tailboom which survived the post impact fire failed to reveal any evidence of pre-impact faulure. The tail rotor hub, pitch control tube, pitch control lever and pitch control link were all free of significant damage and properly installed.

1.12.25. The main rotor servos were heavily damaged limiting an analysis to one of confirming that the mechanical linkage was intact. The anti-torque or tail rotor servo had suffered less severe fire damage. The actuator piston position corresponded to a full right pedal input.

1.12.26. As was the case with the other controls most of the synochronized elevator controls were destroyed. Only those in the unburned portion of the tail boom were available for analysis. During the crash sequence the left hand elevator contacted a tree creating a large imprint on the elevator adjacent to the tail boom. As a result of this impact the elevator bulkhead contacted the tail boom skin leaving a depression which was utilized to determine the elevator and thus cyclic stick position. Matching the imprint on the tail boom with the elevator established that at the time of tree contact the cyclic was in the full forward position.

20/..

1.12.27. The complete aircraft fuel system was virtually destroyed including the fuel cells, all lines, boost-pump, cross feed valves, shut-off valves, etc. Miscellaneous fuel system components which were recovered provided no useful information.

1.12.28. The engine power turbine governors are linked mechanically and electrically to the collective stick. A droop compensator helps control the power turbine speed by modifying the power turbine governor setting as collective pitch changes are made. No section of this section of the engine control system were recovered.

1.12.29. Each engine has an associated power lever control system which are controlled by independent twist grips on the collective stick. The mechanical linkage extends from the twist grips through the collective stick to sector gears. Additional mechanical linkage running under the cabin floor connects the sector gears to the engine fuel controls on the engine accessory gearboxes. The twist grips are designed such that the pilot can twist the grips from a full power position to a flight idle position. A mechanical stop is incorporated into the system to prevent an inadvertant shutdown of the engines due to fuel starvation. A shut-off position can only be accomplished if a solenoid is electrically activated to control a plunger release to allow further twisting of the grip and thus reductions of engine rpm below flight idle. The upper twist grip is the left engine control while the lower grip controls the right engine. The left engine twist grip had been twisted past the idle stop and into the cut-off position. Both the pilot and co-pilot twist grips were in this position. The right engine twist grip had been rolled by to the flight idle stop. The stop release switch was not recovered.

1.12.30. Portions of the Instrument panel were heavily damaged or destroyed by impact forces or the subsequent fire. A number of light bulbs from the warning systems incorporated in the aircraft were recovered and inspected for filament stretch. If the impact forces are of sufficient magnitude, shock loads may be imparted to the light bulb filaments. The reaction of the filament to these impact loads will be a function of the operational state of the bulb at the time. If a light bulb is illuminated the filament would be very ductile with a low yield strength by virtue of the high filament temperatures. Shock loads would then cause significant distortion and stretch of the filaments. Thus if extensive filament stretch is observed then this is reasonable evidence that, that particular bulb was illuminated at the time of impact. Filament failure without significant stretch is normally an indication that the failure was brittle in nature and indicates that the filament was cold at the time of failure. Low main rotor rpm caution lights are installed in both the pilot and co-pilot panels. These bulbs were not recovered. The master caution warning indicators from both the pilot and co-pilot panels were recovered as were both the left and right engine out warning indicators. The engine fire warning lights are located on the fire extinguisher pull handles. Neither handles was recovered and thus filament analysis was not possible. It should be noted that each warning indicator incorporates two bulbs. The results of the visual inspection of the panel light bulb filaments was as follows:-

Identification	Observation & Bulb Bulb Condition	Filament Condition
(a) Master Caution-Pilot Palen	1) Bulb broken	filament stret
	2) Bulb broken	filament missing
(b) Master Caution-Co Pilot Panel	1) Bulb intact	filament stretch
	2) Bulb intact	filament stretch
(c) Engine Out Left Engine	1) Filament intact	no visual stretch
	2) Filament intact	no visual stretch
(d) Engine Out Right Engine	1) Bulb broken	unable to determine
	2) Bulb broken	unable to determine

The following observations were made during a visual examination of the light bulbs successfully removed from the annunciator panel:-

Id.ntification	Filament Condition
(a) No.2 Oil Pressure	1) cold break 2) cold break
(b) No.2 D.C. Generator	1) cold break 2) cold break
(c) No.2 Fuel Boost	1) possible stretch 2) possible stretch
(d) N_2 Fuel Filter	1) cold break 2) cold break
(e) Fuel Low	1) possible stretch 2) possible stretch
(f) No.2 Governor Manual	1) cold break 2) cold break
(g) Right Engine Chip	1) cold break 2) cold break
(h) No.2 Fuel Valve	1) cold break 2) cold break
(i) No.2 Generator overheat	1) cold break 2) cold break
(j) No.1 D.C. Generator	1) unable to determine 2) unable to determine
(k) No.1 Engine Oil Pressure	1) cold break 2) cold break
(l) No.1 Fuel Boost	1) cold break 2) cold break
(m) Common Hydraulic Pressure	1) cold break 2) cold break
(n) Inverter No.2	1) cold break 2) cold break
(o) Combining Gearbod	1) cold break 2) cold break
(p) Transmission Chip	1) cold break 2) cold break
(q) Common 42° & 90° gearbox	1) unable to determine 2) unable to determine
(r) Common Heater Air	1) unable to determine 2) unable to determine

23/...

1.12.31. Portions of the pilot, co-pilot and center instrument panels were recovered. In addition, a section of the overhead panel was available. Analysis of the instruments indicated the following:-

(a)	i)	No.1 Hydraulic System	Temperature	- off scale high
	ii)	No.1 Hydraulic System	Pressure	- 105 psi
(b)	i)	No.2 Hydraulic System	Temperature	- unable to determine
	ii)	No.2 Hydraulic System	Pressure	- unable to determine
(c)	i)	Gas Producer Techometer Indicator	Left Engine	- approx.30%
	ii)	Gas Producer Techometer Indicator	Right Engine	- unable to determine
(d)		Fuel Quantity Indicator	No useful information	
(e)	i)	Turbine Inlet Temperature	Left Engine	- unable to determine
	ii)	Turbine Inlet Temperature	Right Engine	- 200°C
(f)	i)	Left Engine	Oil Pressure	- unable to determine
	ii)	Left Engine	Oil Temperature	- unable to determine
	iii)	Right Engine	Oil Pressure	- 50 psi
	iv)	Right Engine	Oil Temperature	- unable to determine
(g)	i)	Left Engine	Fuel Pressure	- unable to determine
	ii)	Right Engine	Fuel Pressure	- unable to determine
(h)	i)	Left Engine	Fuel Flow	- unable to determine
	ii)	Right Engine	Fuel Flow	- 191 lbs. per hour
(i)	i)	Pilot Torque Indicator	i) Combined Torque	- 0
			ii) Left Engine Torque	- 57%
			iii) Right Engine Torque	- 52%

(j) Co-Pilot Torque Indicator	i) Combined Torque	-	80%
	ii) Left Engine Torque	-	unable to determine
	iii) Right Engine Torque	-	unable to determine
(k) No.1 Dual Voltmeter (No.1 Bus)	i) AC Voltage	-	greater than 130 volts
	ii) DC Voltage	-	23 volts
No.2 Dual Voltmeter (No.2 Bus)	i) AC Voltage	-	0
	ii) DC Voltage	-	0
(l) Dual DC Ammeter	unable to determine		
(m) Pilot Airspeed	15 knots		
(n) Pilot Triple Techometer	i) Rotor RPM	-	unable to determine
	ii) No.1 Engine RPM	-	greater than 120%
	iii) No.2 Engine RPM	-	unable to determine

(o) i) Pilot Vertical Speed Indicator - unable to determine

ii) Pilot Altimeter - 4800 feet

(p) Co-Pilot Altimeter - 7000 feet

(q) Co-Pilot Vertical Speed Indicator - unable to determine

(r) Co-Pilot Airspeed - unable to determine

(s) Co-Pilot Triple Tachometer - unavailable

(t) Both Fire Pull T handles had failed and were not recovered. This precluded an analysis of the Fire Warning Lights. Inspection of the T-Handle machanism indicated that the fire extinguisher activation switch for the FIRE 1 PULL (left hand) handle was in the activated position. The FIRE 2 PULL (right hand) switch was not deployed.

25/...

(u) A fire extinguisher system selector switch is located near the top of the center panel. This toggle switch has three possible locations, i.e. OFF, MAIN & RESERVE. Normal practice is to select this switch to the OFF position. If it is required to utilise the fire extinguisher system the toggle switch is selected to MAIN or up. The selector switch was in the MAIN position. There was substantial impact damage in this area of the instrument panel.

1.13. Medical and Pathological Information

1.13.1. All of the aircraft's occupants were killed by multiple severe injuries with extensive burns. The injuries sustained were as a result of the deceleration of the helicopter.

1.13.2. A review of the Captain's medical records disclosed no evidence of pre-existing physical problems which could have affected his judgement or performance. Post mortem and microscopic examination did not reveal any abnormalities in the Captain.

1.14. FIRE

1.14.1. Statements from witnesses and evidence from the wreckage indicated that there was a severe post impact fire which destroyed a major portion of the helicopter. The fire also destroyed some surrounding vegetation.

1.15. SURVIVAL ASPECTS

1.15.1. The accident was unsurvivable.

1.16. TEST AND RESEARCH

1.16.1. A test on the fire extinguishing system similar to the helicopter was carried out at Heli Orient Singapore. The result of the test indicated that the extinguishing system produced a significantly loud bang when triggered.

1.17. ADDITIONAL INFORMATION

NIL

1.18. NEW INVESTIGATION TECHNIQUES

Not applicable

2. ANALYSIS

2.1. Aerodrome and Weather

2.1.1. Ulu Kali pad which was a point of departure is situated on a saddle approximately 1636 meters above mean sea level. The physical features surrounding the pad form a funnel which is aligned with the path of the North Easterly and South Westerly wind. This funnelling has the effect of increasing the prevailing wind strength at the pad. Meteorological report indicated that on the day of the accident, Genting Highlands could have experienced a gusty condition of wind strength well in excess of 10 kts. This condition was also reported by witnesses who observed the departure of the helicopter.

2.1.2. It was also reported that there was no thunderstorm over Genting Highlands. However, due to mountain effects, there were presence of low cloud with reduced visibility.

2.1.3. On 18th August 1979, the Department of Civil Aviation Malaysia approved the Ulu Kali pad for night take-offs and landings. Together with the approval, several limitations were imposed which included night VFR minimas of 1 nm visibility, clear of cloud and in sight of ground.

2.2. Final Flight Path

2.2.1. From analysis of the physical evidence at the impact site, instrument analysis and witnesses reports, the flight path of 9M-AWW were constructed and the following were noted:-

(1) The elapsed time from take off to impact was approximately 22 seconds.

(2) The helicopter departed on a heading estimated to be 070° mag. and climbed to an altitude approximately 300 ft. above the pad level.

(3) Following departure, 9M-AWW was observed to enter a left turn.

(4) 9M-AWW impacted while on a heading of 291° mag. which was approximately at right angle to an earth embankment.

(5) The helicopter impacted in a severe nose down left roll attitude. The nose was estimated to be pitched down $55^{\circ} - 65^{\circ}$.

(6) The slope of the embankment which the aircraft impacted was $70^{\circ} - 80^{\circ}$.

(7) An additional 50 vertical feet of altitude would have allowed the helicopter to clear the embankment.

(8) There was no radio communication with 9M-AWW following departure.

2.3. Wreckage

2.3.1. The on-site evidence indicated that the helicopter impacted with a nose down, left roll attitude. Impact was followed by an immediate and intense fire which destroyed most of the helicopter.

2.3.2. There was no evidence of an in-flight structural failure or damage due to contact with trees further back on the flight path. The extent and nature of the break-up, particularly the front sections of the aircraft, indicated that impact occurred at fairly high speed.

2.3.3. The physical evidence indicated that the main rotor blades were rotating at a fairly high rpm but with minimal torque. Such a condition would be consistent with an autorotation descent. The primary evidence of this condition was the lack of compressive damage to the blade structure, the nature of the leading edge damage, the lack of break-up or failure of many of the main rotor mast components.

2.3.4. The mechanical condition of the turbine and compressor sections of both engines indicated that these sections were intact and capable of normal operation at the time of impact. In fact, the lack of mechanical damage or evidence of ingestion of foreign debris is consistent with low power at the time of impact. Substantial power turbine blade top rub was observed in the left engine. This condition was not attributed to blade stretch but rather due to displacement of the rotating component due to high "g" forces at impact. Only fairly minor rub was observed on the power turbine wheel of right engine. The accessory drive train from the engines indicated that the accessory gear train was intact. The analysis of the combining gearbox components which were recovered failed to reveal any evidence of in-flight or preimpact distress.

2.3.5. It is considered significant that no extensive scoring damage was noted on the driveshaft. This would indicate that the female coupling failed at impact and further there was little or no postcrash rotation of the driveshaft. Under high power conditions and with a sudden stoppage torsional twisting of the driveshaft may occur. None was observed. The lack of rotational interference damage and torsional deformation is a further indication of a low engine power condition.

2.3.6. Thus an analysis of the helicopter drive train indicated that it was intact and mechanically capable of normal operation at the time of the accident. The nature and extent of damage sustained by the drive train components is consistent with a low power condition at the time of impact.

2.3.7. Some indications of control positions such as the full right pedal position suggested by the tail rotor servo were indentified. It was not possible to determine if the indications observed were indicative of the control positions just prior to or at impact as opposed to positions which were the result of impact.

2.3.8. The full nose down cyclic indication as given by the synchronized elevator position was probably the result of the cyclic stick being forced forward at impact and therefore was not an indication that full forward cyclic was a pilot input at the time of the crash. Continuity of the main rotor controls from the servos through to the main blades was confirmed. All damage and failures which occurred were the result of impact. There was no physical evidence to indicate an in-flight control failure. Nevertheless, the possibility that a control failure was experienced cannot be completely eliminated as much of the system was not available for inspection. It was not possible to determine the serviceability of any of the hydraulic system components at the time of the accident.

2.3.9. A visual analysis of the light bulbs removed from the various warning systems indicated that the MASTER CAUTION warning lights on both the co-pilot and pilot panels were illuminated at the time of impact. The filament in the bulbs from both of these locations had clearly stretched. This is an indication that an in-flight problem was experienced but gives little information as to the nature of the problem as inspections of the bulbs removed from the warning panel did not disclose any bulbs in which the filaments had clearly stretched.

2.3.10. The pilot and co-pilot collective sticks both became separated and were deformed and heavily damaged by fire. In both cases the left engine throttle twist grip was in the cut-off position, indicating that the pilot had elected to shut down that engine prior to impact. The right engine twist grip was in flight idle position, indicating that the power was reduced by the pilot. This is consistent with the physical evidence observed throughout the engines and drive train indicating a low power condition at impact.

2.3.11. The FIRE 1 PULL fire extinguisher system switch was in the deploy condition. In addition, the fire extinguisher selector switch was in the MAIN position. The fire warning lights are located in the Fire Pull T-handles and were not recovered. Therefore, no light bulb filament analysis was possible to determine if the fire warning lights were on. However, if a fire warning light was illuminated the MASTER CAUTION lights would also be illuminated. The evidence is that the MASTER CAUTION lights were illuminated at impact.

2.3.12. The Bell 212 Flight Manual indicates if the FIRE 1 PULL (fire warning) light illuminates then the following corrective action should be taken:

(1) Pull No.1 handle;
(2) Select MAIN bottle;
(3) Close No.1 twist grip.

An analysis of the wreckage indicated that the three steps above had been taken by the pilot. This was an indication that following take-off an in-flight problem which the response was to an indication of a fire in the left engine compartment was experienced. However, the severity of the post-crash fire prevented any determination of the nature or extent of an in-flight fire if indeed one was experienced.

2.3.13. The Bell 212 Flight Manual indicates that a specific procedure should be followed if a Fire Warning Light illuminates during take-off or landing. The Manual indicates that flying the helicopter to a landing should be the primary function with the fire corrective action as a secondary consideration.

2.5.14. Following take-off, 9M-AWW apparently descended rapidly. It is not known if the high descent rate was induced by the pilot to enter autorotation or was the result of an emergency condition which prevented the aircraft from maintaining flight and returning to the take-off location. Observations made during an analysis of the wreckage indicated that the pilot was in the process of performing the emergency procedures corresponding to an inflight fire in the left engine compartment at the time of impact.

3. CONCLUSIONS

3.1. FINDINGS

(i) The helicopter had been properly equipped and maintained in accordance with an approved schedule and its documentation was in order.

(ii) The pilot was properly licenced and qualified for the flight.

(iii) The aircraft's all up weight and centre of gravity were within permissible limits.

(iv) The flight encountered low cloud and strong gusty wind conditions immediately after departure from the Ulu Kali pad.

(v) An indication of a fire in the left engine compartment was experienced immediately after the departure following which the helicopter descended very rapidly.

(vi) The Bell 212 Flight Manual indicates that, if a Fire Warning Light illuminates during take-off or landing, flying the helicopter to a landing should be the primary function with the fire corrective action as a secondary consideration.

(vii) The pilot was in the process of performing the emergency procedures corresponding to an inflight fire in the left engine compartment at the time of impact.

(viii) The severity of the post crash fire prevented any determination of the nature or extent of an inflight fire.

(ix) The Ulu Kali pad was approved for night take-offs and landings, subject to minimas of one nautical mile visibility, clear of cloud and in sight of ground.

3.2. CAUSE

3.3. The cause of the accident was the inability of the helicopter to maintain height whilst in cloud following an indication of an inflight fire in the left engine compartment.

4. SAFETY RECOMMENDATIONS

4.1. It is recommended that: Policies, procedures, practices and training should be reviewed towards increasing crew efficiency and competency particularly in the handling of emergencies.