AIRCRAFT ACCIDENT REPORT 3/87

Accidents Investigation Branch

Department of Transport

Gulfstream Rockwell Turbo Commander EI-BGL at Jevington, Eastbourne on 13 November 1984

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Department of Transport
Accidents Investigation Branch
Royal Aircraft Establishment
Farnborough
Hants GU14 6TD

11 May 1987

The Rt Honourable John Moore
Secretary of State for Transport

Sir,

I have the honour to submit the report by Mr M M Charles an Inspector of Accidents, on the circumstances of the accident to Gulfstream Rockwell Turbo Commander EI–BGL which occurred at Jevington, Eastbourne on 13 November 1984.

I have the honour to be
Sir
Your obedient Servant

D A COOPER
Chief Inspector of Accidents

Contents

Accidents Investigation Branch

Aircraft Accident Report No. 3/87
(EW/C893)

Operator:		Flight Line Ltd
Aircraft:	*Type:*	Gulfstream Rockwell Turbo Commander
	Model:	690B
	Nationality:	Irish
	Registration:	EI–BGL
Place of Accident:		Jevington, near Eastbourne, Sussex
		Latitude: 50° 46′ North
		Longitude: 000° 08′ East
Date and Time:		13 November 1984 at 1841 hrs
		All times in this report are UTC

Synopsis

The accident was notified to the Accidents Investigation Branch by the London Air Traffic Control Centre and the investigation was commenced on the same day. The aircraft was flying from Dublin to Paris (Le Bourget) at a height of 25,000 feet. In the area of Petersfield, Hampshire the aircraft began a gentle turn to the left from a south easterly heading. After the radar controller queried the departure from the expected heading the commander reported that the auto-pilot had "dropped out", and the south easterly heading was resumed. Approximately 7 minutes later, the radar recording shows that the aircraft again began to turn left and started to lose height. After the aircraft had reached a northerly heading it began to lose height rapidly following which secondary radar returns were lost and the primary returns became fragmented before they also disappeared.

The aircraft suffered an in-flight disintegration at approximately 19,000 feet and all 9 occupants were killed. A positive cause of the accident was not determined but there was evidence that a part of the aircraft's electrical supply had been lost. This would have caused the autopilot to disengage and also have resulted in the failure of the commander's flight director indicator. It was concluded that, following the disengagement of the autopilot, the aircraft probably entered a steep spiral dive and that the disintegration of the aircraft occurred as recovery was attempted.

1. Factual Information

1.1 History of the flight

The aircraft was on a charter flight from Dublin to Paris (Le Bourget) with the aircraft commander and eight male passengers on board. The aircraft took-off at 1704 hrs intending to fly via airways at flight level (FL) 250 (ie, 25,000 feet), with a standard arrival procedure at Le Bourget.

At 1726 hrs the commander made radio telephone (RTF) contact with London Control (Air Traffic Control – ATC) reporting his position as Vatry (halfway across the Irish Sea) and was then cleared to climb from FL 230, his reported altitude, to FL 250. At 1733 hrs the aircraft was cleared to fly directly to the Brecon VOR* and the commander reported overhead this beacon at 1758 hrs.

At 1812 hrs the commander reported his position as Lyneham and requested approval to fly directly to the Midhurst VOR. London Control cleared the aircraft to fly directly to the Dieppe VOR, located 150 nautical miles to the south east of the aircraft's position at that time. The commander advised London Control at 1827 hrs of his revised estimated time of arrival at Dieppe as "five zero" (ie, 1850 hrs). Examination of the recording of the ATC radar plot (see Appendix 1) indicates that at about this time the aircraft commenced a very gentle turn to the left, maintaining FL 250. Four and a half minutes later London Control queried the aircraft's heading, which was now north easterly, and the commander apologised, remarking that the autopilot had "dropped out" (become disengaged).

The aircraft was observed turning to the right at a normal rate of turn to intercept a direct track to Dieppe. A magnetic heading of 140° was established, and was maintained at the request of London Control until 1834 hrs when the aircraft was again cleared to fly directly to Dieppe. The recorded altitude on this heading was FL 250.

Shortly after 1839 hrs secondary radar** was indicating the altitude as FL 249, and further study of the radar recording shows that the aircraft had commenced a slow turn to the left. At 1840:04 hrs London Control advised the commander to call Paris Control on a frequency of 132.0 megahertz (MHz) and this was acknowledged by the commander who terminated his transmission with "Good-day sir." at 1840:08 hrs.

The turn to the left, and gradual descent continued until the aircraft was on a northerly heading; the rate of descent then increased and at 1840:34 hrs secondary radar returns ceased at an altitude below FL 222. The primary radar return then fragmented and particles were observed on the screen to be drifting northwards for a further ten minutes.

*VOR : Very High Frequency Omni Range: A radio beacon providing bearing information to the aircraft.

**Secondary radar: A ground based radar capable of receiving and decoding signals from equipment mounted in the aircraft (the transponder) giving identification and altitude information.

Upon observing this, London Control made repeated attempts to contact the aircraft and ascertained that RTF contact had not been established with Paris Control. The Sussex police were then alerted to the possibility of an aircraft accident in the area of Eastbourne. A number of witnesses had already contacted the police, having heard an aircraft, seen a flash in the sky above the cloud, and heard whirring, rushing sounds and a high revving engine. A number also heard the impacts of major components but nobody witnessed them. The weather at the time was low cloud and heavy rain with poor visibility. There were numerous reports of an aircraft flying below the cloudbase with the landing lights being switched on and off as if in trouble, but this was later identified as a light aircraft flying from Goodwood to Southend, attempting to maintain visual contact with the ground and to avoid the cloud. One witness, a reserve lifeboatman, heard an interrupted Mayday call on a marine channel on his home VHF scanner and he assumed it was a vessel in distress.

The police instituted a search for the missing aircraft and at 1855 hrs had arranged a rendezvous point for the emergency services. At 1907 hrs a police officer noticed a fire on a ridge near Jevington and, upon closer inspection, discovered the burning fuselage. The fire and ambulance services had already been dispatched and were en-route to the rendezvous when they were advised of the location of the accident site.

The fire appliances and two ambulances arrived on the B2105 road immediately below the fuselage site between 1913 hrs and 1916 hrs but access to the site was found to be unsuitable. A fire tender and ambulance were then led to the fuselage along a road and a track on the ridge above the accident site. The fuselage fire was extinguished by water spray, and at least two bodies were identified in the forward part of the fuselage, one being in the cockpit area. The search for bodies and wreckage was restricted by the darkness and poor weather. Six of the bodies had been ejected during the break-up sequence and were accounted for by the early hours of the following morning, with the assistance of HM Coastguard parachute flares. The wings and engines were located by first light and at noon a third body was found in the cockpit area of the fuselage, confirming a total of nine.

A concentrated search for wreckage continued until 23 November using a light aircraft, a helicopter, police officers on foot and mounted police. The wreckage trail was found to be spread in a straight line northwards from Jevington over a distance of 10.1 statute miles.

1.2 Injuries to persons

Injuries	Crew	Passengers	Others
Fatal	1	8	–
Non-fatal	–	–	–
Minor/None	–	–	

1.3 Damage to aircraft

The aircraft was destroyed by an in-flight break-up and subsequent ground fire.

1.4 Other damage

Minor damage was caused to agricultural land.

1.5 Personnel information

Commander:	Male, aged 48 years
Licence:	Airline Transport Pilot's Licence (Ireland), valid until 23 March 1985
Aircraft ratings:	Cessna 100 series, DC3, Beechcraft 90, 99 and 200, Shorts SD3–30 and Rockwell 690B
Instrument rating:	1 November 1984, valid until 30 October 1985
Last medical examination:	13 September 1984, Class 1 with no restrictions
Flying experience:	Total flying hours: 10,256 Hours on turbo-prop aircraft: 5,500 (approximately) Hours on Rockwell Commander 690B: 150 as first pilot under supervision or as co-pilot The accident flight was his first as pilot-in-command on this type as a single crew operation
Duty time:	Off duty 11 November 1984. Standby duty 12 November 1984. On duty 1345 hrs 13 November 1984 until 1841 hrs (4 hours and 56 minutes)

1.6 Aircraft information

1.6.1 Leading particulars

Type:	Rockwell Commander 690B
Constructor's number:	11507
Date of manufacture:	November 1978
Certificate of Airworthiness:	Issued by the Department of Tourism and Transport, Ireland in the Transport of Passengers category. Valid until 19 April 1985
Certificate of Registration:	The registered owners were Flight Line Limited
Certificate of Maintenance:	Renewed on 23 October 1984 at 2372.55 airframe hours and valid until 2472.55 hours

Total airframe hours:	2390.15
Engines (2):	Garrett Airesearch TPE 331
Propellers (2):	Hartzell HC–B3TN–5FL
Total engine hours:	Left : 2390.15 Right : 2066.35
Aircraft limitations (Pilot's Operating Handbook)	
Manoeuvring indicated airspeed (V_A):	148 kt
Maximum operating indicated airspeeds (V_{mo}):	sea level to 19,000 feet: 246 kt 21,000 feet: 234 kt 23,000 feet: 225 kt 25,000 feet: 216 kt
Limit manoeuvring load factors of 10,325 lb with landing gear retracted and flaps at 0°	+3.28 g to –1.44g

1.6.2 *Technical defects*

Inspection of the maintenance records for the aircraft showed that there was a deferred defect of "Altitude pre-select unserviceable" at the time of the accident. It was noted that on three occasions over the two years prior to the accident the elevator trim tab was reported for freezing at altitude: lubrication cured the problem. No defects had been reported in the Technical Log since the last maintenance check which was carried out on 19 October 1984 at 2372 airframe hours.

1.6.3 *Weight and balance*

Maximum weight authorised for take-off:	10,325 lb (4,683 kg)
Actual take-off weight:	10,346 lb (4,692 kg) (estimated)
Estimated weight at the time of the accident:	9,469 lb (4,307 kg)
Estimated centre of gravity at the time of the accident:	211 inches aft of datum
Centre of gravity range at estimated accident weight:	210.4 inches to 218.5 inches aft of datum
Estimated fuel remaining at the time of the accident:	800 lb (363 kg)

The loadsheet for the flight was recovered from the vicinity of the wreckage. Standard weights of 165 lb (75 kg) for a male and 143 lb (65 kg) for a female had been used in calculating the weight and balance for this flight. The 4 seats behind the pilot's seats were shown as being occupied by females, whereas all the occupants were male, adding 88 lb (40 kg) to the calculated take-off weight of 10,308 lb (4,675 kg). The 50 lb (23 kg) taxy and take-off allowance was not used by the commander and this brought the actual take-off weight to 10,346 lb (4,692 kg). Due to the nature of the accident it was not possible to weigh accurately the items contained within the aircraft, including baggage. The original loadsheet weight of 100 lb (45 kg) for baggage was therefore used in subsequent calculations.

1.6.4 *Electrical system*

1.6.4.1 *Direct current (DC) system*

The aircraft's electrical system was based on 28 volts DC which was provided by two engine driven generators and two lead acid batteries. The generators and batteries fed 28 volts DC into the main DC bus bar which was situated in the rear fuselage behind the luggage bay. Electrical power from the main DC bus bar was fed forward to the control and distribution bus bars via cables routed along the inside of the right-hand side of the fuselage approximately 18 inches below the bottom of the cabin windows. The DC control and distribution bus bars together with their associated circuit breakers and fuses were located in the cockpit.

In the event of a double generator failure, provided the battery master switch in the cockpit was selected on, the batteries would automatically provide DC power to the main DC bus bar. Both generators had warning lights mounted in the main annunciator panel which, when illuminated, would indicate that the particular generator was not connected to the main DC bus bar. In the event of a DC distribution or control bus bar failure the appropriate bus-off light would illuminate. After isolation of the electrical fault, electrical power could be transferred from one bus bar to another by closing the appropriate bus tie circuit breaker.

1.6.4.2 *Alternating current (AC) system*

The AC electrical power was supplied by two inverters located in the rear fuselage. Each inverter produced both 26 volts and 115 volts to power the AC electrical components. The DC power for the inverters was routed via the same relays which powered the two DC radio bus bars. These relays were controlled by the radio 1 and radio 2 master switches which were located in the overhead panel above the co-pilot's seat. Selection of the radio 1 or radio 2 master switches to off would switch off the DC electrical supply to the corresponding inverter. The control of the AC power from the inverters to the AC bus bars was achieved by the inverter switch in the overhead panel above the co-pilot's seat. This switch allowed the pilot to select either inverter 1 or inverter 2 to power the 26 volt AC and 115 volt AC bus bars or to effectively turn off the AC power and it was not possible to connect both inverters simultaneously, neither was it possible to select off either the 26 volt output or the 115 volt output independently. When the battery master switch was selected on and neither inverter was operating an AC volts light would illuminate in the annunciator panel. The AC volts light would also illuminate following a 26 volt AC failure or if the inverter switch was selected off. In the event of failure of the inverter selected originally, the pilot would have to select the second inverter. Distribution of the AC power from the inverters to the AC bus bars, circuit breakers and fuses (located on the left-hand side of the cockpit) was via cables routed alongside the DC distribution cables.

1.6.5 *Autopilot and flight director system*

The aircraft was fitted with a Rockwell Collins AP 106 autopilot and a FD–112V flight director. The compass gyros were Sperry C14's.

The autopilot operated electric servos on the rudder, aileron, and elevator control circuits. In addition it operated an electric servo driving the elevator trim circuit in a follow-up mode to trim out the elevator control forces. The autopilot controller was common to both the flight director and autopilot functions and the unit contained a computer section which developed the lateral and vertical steering commands. With the autopilot engaged, the computer controlled the primary servos and also the elevator trim servo in its follow-up mode. The FD–112V was a combined instrument with a flight director indicator in the upper half and a horizontal situation indicator in the lower half. With the autopilot disengaged, the flight director presented computed commands to enable the pilot to control the aircraft's flight path manually. With the autopilot engaged, the system automatically controlled the aircraft and the pilot monitored the flight path by observing the information displayed on the flight director.

The autopilot had two modes of operation, attitude and guidance. When engaged with no modes selected on the controller, the autopilot was in the attitude mode and accepted pitch and roll rate commands from the pitch/turn control knobs. When the autopilot was engaged and a mode was selected on the controller, the autopilot was in the guidance mode and accepted the computer developed steering commands. The system could perform a variety of functions including maintaining a barometric altitude and a desired heading.

The autopilot/flight director system had a number of warning flags that were displayed on the flight director when a system malfunction was detected. Amongst the system flags were:

a. Heading flag indicating a failure of the compass system.

b. Attitude flag indicating a failure of the vertical gyro or attitude display.

c. Computer flag indicating a fault in the signal source used to generate the steering signals.

d. Navigation flag indicating a fault in the lateral deviation bar.

A number of annunciator lights were provided to indicate to the pilot the status of the system. A small annunciator panel mounted above the flight director contained fourteen captions, including one which illuminated when the heading mode was selected and another to indicate the selection of the altitude hold mode. The autopilot controller also had annunciator lights to indicate the system status and these included captions which illuminated under the following conditions:

a. Autopilot engaged

b. Autopilot disengaged

c. Autopilot commanding trim-up

d. Autopilot commanding trim-down

e. Selection of heading mode

f. Selection of navigation mode

g. Selection of altitude hold mode

h. Selection of indicated airspeed hold mode

An audio tone was provided which was activated whenever the autopilot operated the electric elevator trim system. The tone was not present when the manual electrical elevator trim was used.

The autopilot disengaged automatically in the event of power failure or gyro monitor failure and would cause the autopilot disengaged caption on the main aircraft annunciator panel to illuminate. There was no audio warning provided to alert the crew that the autopilot had disengaged, however, the signal which was used to activate the above visual warnings also had the capability of operating an audio warning, if fitted. The autopilot could also be disengaged by the operation of the:

a. autopilot release switch (on the co-pilot's control wheel)

b. Autopilot engage/disengage switch (on the controller)

c. Manual electric elevator trim switch (on the pilot's control wheel)

d. Electric elevator trim and autopilot disengage switch (on pilot's control wheel)

e. Autopilot circuit breaker (in the lower right-hand panel)

f. Autopilot master switch (in the overhead right-hand switch panel)

In addition, the operation of the turn control knob, when the autopilot was engaged and either the heading or navigation mode was selected, would cause that mode to release and the autopilot to revert to the attitude hold mode. Similarly, rotating the pitch control knob would release a selected vertical mode and the autopilot would revert to pitch hold mode.

1.7 Meteorological information

An aftercast was prepared by the Meteorological Office for the period 1800 hrs to 1850 hrs covering the aircraft's planned track over southern England. The accident happened at night.

1.7.1 Synoptic features

A waving cold front was orientated north to south over the North Sea to near the Straits of Dover and south southeastwards. A light to moderate southerly flow covered the route.

Upper features

A trough to the west of the UK left the route in a southerly flow.

Winds and temperatures in the east of the route

FL 390	190°	55 kt	Minus	60°C
FL 340	190°	75 kt	”	59°C
FL 300	190°	65 kt	”	48°C
FL 240	190°	60 kt	”	33°C
FL 180	190°	60 kt	”	21°C
FL 100	180°	35 kt	”	05°C
FL 050	180°	25 kt	”	0°C
FL 020	180°	20 kt	Plus	06°C
SURFACE	160°	10 kt		

Maximum wind: FL 350 190° 80 kt
Heights of the ZERO DEG Isotherm: 5,000 feet
Cloud Structure in the east of the route (Heights Above Mean Sea Level)

Broken cumulus, stratocumulus base 2,000 feet tops 6,000 feet locally base 1,000 feet touching highest hills at times, mainly near precipitation. Broken layers of altocumulus and cirrus between 10,000 feet and 25,000 feet: thin cirrus layer was above 34,000 feet.

Weather in the east of the route
Outbreaks of rain, moderate at times. Hill fog patches were likely over highest ground.

Visibility in the east of the route
Mainly 10 kilometres or more but locally 2,000 metres and 400 metres or less in hill fog.

Icing
Generally light or moderate in cloud between 10,000 feet and 20,000 feet.

Turbulence and Clear Air Turbulence (CAT)
Generally slight or moderate in cloud. Moderate CAT could occur between FL 250 and FL 380.

No reports of significant turbulence or CAT were received at Heathrow in the vicinity of the route.

Further Comments

At 25,000 feet the aircraft should have been flying just above the thickest cloud and thus not in danger of any icing, but on descending the aircraft could have encountered moderate icing in cloud down to 10,000 feet and the aircraft may have experienced moderate turbulence during the descent, but severe turbulence seems unlikely.

1.7.2 A radio sonde ascent from Crawley (near Gatwick) was made at approximately 1800 hrs. Ten recorded samples of the wind velocity between 15,000 and 26,000 feet indicate that the wind was consistently from the south at around 60 kt.

1.7.3 A BAe 1–11 which had flown through the area of the accident at FL250 some twenty minutes earlier, had its flight data recorder removed and analysed. There was no evidence of atmospheric turbulence.

1.8 Aids to Navigation

All the navigational aids in the vicinity of the aircraft's track, including Dieppe VOR, were serviceable and radiating normally.

1.9 Communications

Communications throughout the flight had been normal, and the last recorded RTF contact was with London Control on 134.45 MHz. London Control asked the commander to change frequency to Paris Control on 132.0 MHz at 1840:04 hrs, and this was acknowledged by the commander at 1840:08 hrs with an additional, "Good-day Sir". There was no further two-way communication between the aircraft and London or Paris Control. Communications between aircraft and major ATC units are continuously recorded and at 1840:15 hrs a momentary transmission with no speech was recorded on the London Control frequency 134.45 MHz. The source of this transmission was not determined.

At about this time a reserve lifeboatman living in Eastbourne was monitoring a number of marine VHF frequency modulated (FM) radio channels on a relatively unsophisticated scanning monitor. He heard "Mayday, Mayday, Mayday .... Golf," and nothing further. He identified the transmission as being broadcast on Marine Channel 6, 156.3 MHz, and assumed that a vessel was in distress. The only emergency logged by HM Coastguard on that day was the accident to EI–BGL.

Telecommunications experts from the National Air Traffic Services advised that it was possible for VHF amplitude modulated (AM) transmissions from aircraft to be received by VHF (FM) receivers when they are close to the transmitter and operating in adjacent frequency bands. It was not uncommon, for example in the vicinity of major airports, for aircraft transmissions to cause interference to VHF (FM) receptions.

1.10 Aerodrome information

Not relevant.

1.11. Flight recorders

Neither a flight data recorder or cockpit voice recorder was fitted, nor were they required to be.

1.12 Wreckage and impact information

1.12.1 On site examination

The wreckage of the aircraft was dispersed over a distance of 10.1 statute miles on a south to north track across the South Downs in Sussex (see Appendix 2). The area consisted of a mixture of farm land and dense woods interspersed with isolated farm buildings, residential buildings, and two villages of small population. The most southerly items of wreckage were the two aircraft batteries which had impacted the ground with a high speed, approximately 4,000 feet north east of the centre of the village of East Dean. Approximately 2,000 feet to the north west of the batteries was the aircraft fuselage, from the nose cone to frame 234 in the baggage bay (minus the wings and a large section of the cabin roof) which had fallen on its upper right-hand side. The impact caused severe distortion to the fuselage structure. Approximately 1,000 feet and 2,400 feet north of the fuselage the right and left wings respectively, complete with their engines and propellers had impacted the ground inverted and in an approximately 20° root end and leading edge down attitude. Both wings had their ailerons missing. The right wing was more severely damaged in the root and trailing edge areas and the left wing had both flaps attached whereas the right wing had both missing. Evidence indicated that the flaps were retracted at ground impact and both landing gears were attached and retracted. In the general area of the wings were found the pieces of missing ailerons and flaps together with internal cabin furnishing, personal possessions and in-flight consumable provisions.

The right propeller showed good evidence of having struck something while under power whereas the left propeller showed evidence of having impacted the ground at a low rotational speed. Both propellers were at or very near the full feather position. There was no evidence of fuel being present in the area of the wing impact points.

In the area between the wings and Jevington village was the rear fuselage, frame 240 to frame 254, which contained the rear avionics and electrics bay. Details of aircraft stations and frames are shown at Appendix 3. In the field next to Jevington church was the bulk of the baggage bay contents. A small number of these items showed evidence of having been involved in a fire. All of the items had a very strong smell of aviation turbine fuel. Between 1¾ and 3 miles from the fuselage were found the majority of the vertical and horizontal stabilisers, the rudder and elevators, the rear 2½ feet of the fuselage and some small heavy pieces of internal fittings. In the area that was 3 to 10 miles north of the fuselage were found many small light weight items which mainly consisted of interior fittings, personal clothing and belongings, loose aircraft paraphernalia and items of aircraft structure. There was no visible evidence of an explosion having occurred within the cockpit, passenger cabin, baggage bay, nose electrics bay or rear fuselage area.

1.12.2 Detailed examination

1.12.2.1 Airframe

a. Fuselage

The fuselage from the tip of the nose cone rearwards to frame 234 was in one piece. The severe post-impact fire destroyed all the left-hand side and remaining cabin roof area of the fuselage. Examination of the remains of the fuselage indicated damage consistent with the ground impact except in the area of frame 117 on the right-hand side, the picture window on the right-hand side and the nose avionics bay cover. Fire markings around the right-hand side picture window indicated that it had been disrupted prior to the ground impact. Examination of the window mounting points within the fuselage did not indicate any form of pre-impact window fitting failure. A vertical tear had taken place in the plane of rotation of the propeller at approximately frame 117 in the fuselage on the right-hand side just below the propeller ice deflection plate. This tear was 27 inches in length and 8 inches in width at its widest point. A piece of fuselage skinning was recovered approximately 2½ miles downwind from the fuselage impact site and when offered up to this tear mark its fracture matched the upper area of the tear mark. This piece of airframe skin was not burnt and had vertical score marks on its outer surface. This would indicate that penetration of this area of the fuselage by the right propeller had taken place prior to the impact with the ground and prior to any fire. The nose avionics bay cover was found some distance from the fuselage indicating that it had become detached prior to the ground impact. Examination of this panel and the forward upper fuselage area indicated that something had struck the forward nose area and this panel prior to the fuselage's impact with the ground. It could not be determined what had struck this panel and the forward upper fuselage area. There was no evidence to support the possibility of the aircraft having collided with a foreign object.

b. Wings

Both wings had damage of a similar nature, the right wing being more severely damaged than the left. Both wings had failed in upload at station 24, the left wing being a very clean failure compared with the right. The undersides of both wings were heavily sooted in the areas between the engine nacelles and the wing tips. This sooting was streaked in a direction from the nacelles to the tips indicating that it had occurred after the wings had become detached from the airframe. The sooting was caused by fuel from the ruptured bag tanks located at station 24 which had been ignited by the engines.

The left wing leading edge between station 24 and the engine nacelle had suffered damage which was consistent with the ground impact. The upper and lower wing skins were virtually undamaged and there were no indications of the wing having struck anything other than the ground. The upper wing outer skin area had 45° creasing between the leading and trailing edge although there was no impact or other damage to the wing leading edge which would have accounted for this creasing. There was no noticeable visual deformation of the wing spars. The aileron had broken away from the wing prior to impacting the ground. Examination of the aileron attachments and skinning at the wing trailing edge indicated that the aileron had travelled to and beyond its maximum up position pulling rearwards and upwards on the centre hinge attachment. This force was reacted by the aileron control rod mechanism, which was attached to the aileron centre hinge point, causing the
failure of the centre hinge attachment to the wing. This in turn led to the failure of the aileron in the area of the centre hinge point. The aileron end attachments had failed in bending overload caused by the flailing aileron halves as the wing descended to the ground. There was no evidence of the

aileron having contracted anything with enough force, in the static sense, to cause its removal from the wing. Examination of the wing and aileron structure did not show any indications of loose or worked rivets or bolts that would suggest that aileron flutter had occurred. The flaps had remained attached to the wing and there was no evidence of damage other than that caused by the ground impact.

The right wing had suffered severe damage in the area between the engine nacelle and station 24, some damage being caused by the ground impact, some by airborne impact and some by the disintegration. In general the damage was a downward bending of the spars causing a failure in a downward direction of the top front spar midway between stations 39 and 69 and a compression of the lower skins causing skin/rib separation. Leading edge damage between the engine nacelle and station 24, similar to the left wing was also present. The spar, at approximately station 165, had a slight downward bend and there was compression buckling of the lower wing skin in the same area. The aileron had broken away from the wing and fallen in two pieces. There was no evidence of distortion of the wing skin in the area of the aileron except at the inboard hinge point. The trailing edge of the wing from the inboard aileron hinge towards the engine nacelle for an approximate distance of 4 feet had contacted a fairly solid object during the intitial disintegration sequence. The effect of this was to disrupt the aileron and its inboard hinge allowing the aileron to flail and eventually break away from the wing. Examination of the wing and aileron structure did not show any indication of loose or worked rivets or bolts that would suggest that aileron flutter had occurred. In the area of the flaps and rear engine nacelle witness marks provided clear evidence that the wing had struck the right horizontal stabilizer and rear right fuselage side. This damage had caused severe disruption to the flaps and rear engine nacelle which resulted in them becoming detached from the wing.

A microscopic and metallographic examination of the spar failures in the area of station 24 were conducted by the Materials Department of the Royal Aircraft Establishment. Both wings main spars fracture features exhibited fast ductile failures in tension and upward bending except for the left wing front upper spar which exhibited compression and upward bending. The three left wing upper intermediate spars exhibited similar features of fast ductile failures in compression and bending which were consistent with end loading in compression. The right wing intermediate spars fracture surfaces were damaged by the ground impact but they revealed general evidence of upward bending. Sections cut from the main spars revealed microstructures consistent with an aluminium copper 2024–type alloy. The hardness levels were high indicating that the strengths were above the required levels. In neither case was there any evidence that the failures had been influenced by pre-existing defects.

c. Empennage

The empennage, which was all of the aircraft rearwards of station 234, had broken into many pieces. The main failures of the fuselage section of the empennage was between frames 234 and 254 and between frames 366 and 409. Examination of these two failure areas indicated that the tail of the aircraft had moved to the left in relation to the cabin area. Damage to the right-hand side skinning indicated that it had been struck by something large and heavy and witness marks in the area of the fuselage below the vertical stabilizer corresponded with marks at the root end of the right wing. The left horizontal stabilizer had failed at approximately station 53 with the tip moving in an upwards and rearwards direction. The right horizontal stabilizer had failed in two places at stations 92 and 53. Both of these failures were in a tip moving rearwards direction. The leading edge of the right horizontal stabilizer had been heavily impacted in a number of places. The major impacts were in the areas of the stabilizer's failure. Witness marks on the leading edge damaged areas matched witness marks on the rear area of the right wing. The centre section of the horizontal stabilizer between left station 53 and right station 53 was intact.

Examination of the attachment points of the stabilizer to the fuselage indicated a general upward overload failure. Both left and right elevators had failed at the mid-hinge point. These failures were consistent with the failures of the left and right horizontal stabilizers. The vertical stabilizer was intact although superficially damaged. It had separated from the fuselage taking with it the frame located at frame 386. Failures in the area where the vertical stablizer attached to the fuselage indicated that the vertical stabilizer had nodded forward in-line with the centre line of the aircraft. The rudder assembly was in two pieces, a failure having occurred at approximately station 90. Examination of the top of the rudder assembly indicated that it had been struck by something heavy on its forward right-hand side. There were no indications of any premature failures or weaknesses within the empennage area or of an internal explosion having occurred.

d. Cabin door

The bulk of the main cabin door had been destroyed during the ground impact fire. Amongst the debris within the cabin area was found the main cabin door hinges, a number of the lock bolts and the main door handle. The door lock bolts were found in the extended position and the handle was found in the closed and locked position. The presence of these items at the main fuselage impact site indicated that the door was attached and with the main wreckage at the time of the ground impact. The evidence from the cabin door handle and the lock bolts would indicate that the door was closed and locked at impact.

e. Emergency exit window

The emergency exit window located on the right-hand side of the fuselage was found in place and locked.

f. Baggage bay door

The baggage bay door was found some distance downwind from the fuselage impact site. It was undamaged and had no evidence of sooting, fire, or explosion damage. The key lock mechanism was in the unlocked position. Examination of the baggage bay door frame indicated that due to the disruption of this frame area, caused by the disintegration of the aircraft, the door had come open allowing the hinge points to break in overload.

g. Seats

All nine seats were recovered, eight were found within the fuselage and one passenger seat was approximately 1500 feet downwind from the fuselage. The single passenger seat that was found some distance from the fuselage was a rearward facing seat that had been located on the right-hand side of the cabin. The outboard side of this seat (that is the side nearest the fuselage skin) had a vertical slash mark through the lower seat cushion supports which corresponded with the propeller slash mark found on the right-hand side of the fuselage. The upholstered cushions from the rear bench seat had been thrown clear of the cabin before the ground impact. There was no evidence on these cushions or those of the single passengers seat of having been involved in a fire or an explosion. Both of the cockpit seat buckles were found to be undone and all but one of the passenger seat buckles were also found undone. It was assessed that the buckles were unlikely to have been released by impact forces.

1.12.2.2 *Flight controls*

Although much of the airframe had been destroyed by the post-impact fire the flight control cables and fitments had survived. No evidence was found in any of the control circuits of a disconnection or restriction prior to the disintegration. In several areas the cables had cut through structure and this was a clear indication of control circuit continuity at the moment of the in-flight disintegration. All failures were found to be in overload consistent with the disintegration.

It was not possible to determine the positions of the elevator, aileron, and rudder trim controls in the cockpit because of fire damage. The wing mounted electrical actuator for the aileron trim was found in an approximately neutral position and this was assessed as a reliable indication of position when its DC supply was lost. The mechanical actuator of the rudder trim was found in a position corresponding to a very small amount of nose left trim. Part of the elevator trim circuit was not recovered but both of the elevator trim actuators were found displaced two-thirds from the neutral position in the aircraft nose-up sense. The rudder and elevator trims were both susceptible to displacement during the break-up sequence and the "as found" positions were therefore unreliable.

1.12.2.3 *Landing gear*

Both main landing gears were found to be in the retracted and locked position. The nose landing gear was found in the extended and downlocked position. This landing gear was held in the retracted position by hydraulic pressure and when the hydraulic pressure failed it would extend and lock under the action of the extension spring.

1.12.2.4 *Engines*

a. General

Both engines had heavy ingestion of mud and ground debris, and severe disruption of the casings had been caused by the ground impact. Initial examination of the engines indicated that the right engine/propeller drive had failed. There were no indications of a disconnection of any of the engine associated controls or systems prior to the disintegration. Both engines were removed from their mountings within the wings and taken to the manufacturer for a detailed strip examination.

b. No. 1 (left) engine

Strip examination of the No. 1 engine revealed two types of rotational witness marks on both compressor shrouds; one was of high speed rotation which had superimposed upon it marks caused by low speed rotation. The low speed rotational rub marks were indicative of an engine being driven by a windmilling propeller at the time of impact with the ground. The high speed rotational marks could indicate that, at some time prior to ground impact the engine had been twisted rapidly about its lateral axis (end over end) in such a way as to cause the central shaft to nod on its bearings, thus allowing the compressor impeller blades to contact their shrouds while the engine was rotating at high speed. No pre-existing conditions were found which would have interfered with the normal operation or rotation of the engine.

c. No. 2 (right) engine

Strip examination of the No. 2 engine revealed that both compressor shrouds had high speed rotational rub marks similar to those found in the left engine. Superimposed on these were static compressor blade impressions indicating that the engine was not rotating at the time of impact with the ground. It was found that the torsion shaft which connected the high pressure turbine to the propeller gearbox had failed in high speed rotational stress. This type of failure is indicative of the propeller striking something while the engine was developing power. There was evidence of damage within the compressor section of the engine consistent with foreign objects entering the intake while the engine was developing power. Within the engine combustion chamber were found some foreign objects in the form of clear plastic which had solidified from a molten state and pieces of woven fabric, some charred and some partially charred. The condition of these items indicated that they arrived there just as the combustion ceased. There was no evidence of foreign object damage within the turbine section. No pre-existing conditions were found which would have interfered with the normal operation or rotation of the engine.

d. Engine fuel control systems

Both of the fuel control units were taken to the manufacturer's overhaul base for strip examination. The strip examination showed the units to be in a serviceable condition except for damage to the casings which was consistent with the impact damage in the areas where they were mounted on the engines.

e. Cockpit engine controls and instrumentation

Because of the severe damage caused by the impact and post-impact fire within the cockpit area no readings or positions could be obtained from the engine controls or instrumentation. The control runs from the engine control levers to the engines were examined and no evidence of a disconnection or restriction prior to the disintegration was found.

f. Engine mountings

Both engines were recovered attached to their wing mounting points. Examination of these mounting points showed damage consistent with the type of impact the engines had made with the ground and there was no evidence of a pre-impact failure or disruption.

1.12.2.5 *Propellers*

a. No. 1 (left) propeller

At the impact site two of the blades were attached to the hub whereas the third blade was detached and lying on the ground approximately 18 inches from the propeller hub assembly. The two blades attached to the hub were virtually undamaged whereas the detached blade had leading edge damage towards the tip, a number of cordwise score marks, and a small amount of midspan bending in a tip forward direction. Examination of the fracture of the detached blade showed it had occurred in overload. The propeller hub assembly was taken to an authorised overhaul agency for strip examination, which showed that it was in a serviceable condition except for minor damage caused by the ground impact.

b. No. 2 (right) propeller

At the impact site all three blades of this propeller were attached to the propeller hub although one of the blades was free to rotate in pitch. All three blades exhibited good evidence of having struck something while under power. The propeller hub assembly was taken to an authorised overhaul agency for strip assembly. The examination showed that the hub assembly was in a serviceable condition except for damage caused by the ground impact. The propeller blade that was free to rotate in pitch was found to have an overload failure consistent with the ground impact.

1.12.2.6 *Fuel system*

Both fuselage and wing inboard fuel tanks were severely disrupted by the in-flight disintegration and the subsequent ground impact. Examination of the intact wing fuel tanks showed no evidence of a pre-disintegration leak. The fuel pipe system from the central fuel sump area to the engines was examined and there was no evidence of a pre-disintegration disconnect. Small amounts of aviation turbine fuel were found within engine fuel components and this was uncontaminated. Because of the destruction of the fuselage fuel tanks and fuel sump area by the ground impact and the post-impact fire, it was not possible to determine whether there had been a pre-disintegration fuel leak. There was no evidence of a pipe disconnect or seal failure within the fuel sump area.

1.12.2.7 *Electrical system*

Examination of the aircraft wreckage showed that all the AC and DC cables from the rear of the aircraft to the cockpit area had been severed when the right propeller had entered the fuselage on the right-hand side. Because of the severe post-impact fire everything forward of the right propeller cut into the fuselage was destroyed. The main DC distribution bus bar together with its associated relays, contactors, regulators and wiring were examined in detail and electrically tested and, with the exception of the distribution bus bar remote controlled aircraft circuit breaker (RCB), no fault was found.

Examination of the distribution bus bar RCB showed that the contactor bar had deformed in such a way that electrical contact between the two ends of the bar was not possible. Metallurgical examination of the contactor bar indicated that it had, at some time, been severely overheated. Examination of the surroundings of the RCB and the contactor bar showed no indications of external heating. Every indication from the RCB and the contactor bar indicated that the heat had been caused by excessive electrical current passing through the contactor bar. The 115 volt and 26 volt AC wiring from the inverters to the area where the cables were severed by the propeller, together with the inverter 1/inverter 2 selection relay, were examined and tested and no pre-disintegration defect was found.

Both DC starter/generator units with their corresponding voltage regulators and the two AC inverters were taken to approved overhaul agencies for testing and found to be serviceable. The two aircraft batteries were totally destroyed by the ground impact and no evidence could be found as to their condition prior to impact. No cockpit indications or selections could be determined due to the total destruction that occurred during the post impact fire.

1.12.2.8 *Avionics*

a. Radio and navigation equipment

All the cockpit located controls, circuit breakers and fuses for this 28 volt DC powered equipment were destroyed by the post-impact fire. The majority of the units, which were mounted in the rear fuselage, just to the rear of the luggage bay, were recovered intact and undamaged by fire. When bench tested all the equipment was found to function correctly and where there was a failure the cause was found to be as a result of impact damage.

b. Autopilot

The main computer unit and cockpit controls were totally destroyed by the post-impact fire. The programming unit and three of the four servo units which were mounted in the rear fuselage were recovered undamaged. The aileron servo unit was severely damaged in the post-impact fire and the indicated airspeed sensor and altitude controller were very badly damaged during the ground impact. Examination of these three units did not reveal any evidence of a pre-impact defect. The programming unit and the three surviving servo units were taken to the manufacturers for testing and examination. All of the units were found to function satisfactorily when bench tested.

c. Attitude and direction gyros

The attitude gyro was recovered intact but with a large dent on the top of the gyro casing. The unit was taken to a manufacturers authorised overhaul agency and strip examined and tested. Bench testing of the unit as a whole was not possible due to impact damage but no fault was found with individual units. Witness marks on the inside of the dented top cover indicated that the gyro had a large amount of gyroscopic energy available, and that the unit was either at an angle of bank exceeding approximately 70° or the gyro roll gimbal was precessing due to the unit exceeding approximately ± 85° in the pitch axis when the 115 volts AC power was removed allowing the roll stop solenoid to engage. Examination of the fine wire brushes that carried the 115 volt AC power to the gyro assembly showed no evidence of burning or arcing indicating that 115 volt AC power was not present at the brushes at the time they were disrupted by their impact with the dented top of the unit. The two directional gyro units and their systems were examined and no evidence of a pre-impact failure was found. The static inverter integral with the unit which supplied the co-pilot's compass was tested and found to be serviceable.

1.12.2.9 *Instruments*

a. General

Because of the damage caused by the ground impact and the severe post-impact fire very few instruments were recovered and identified. Of those which were, very few reliable impact readings could be determined.

b. Collins Flight Director System – FD 112V

The flight director display unit, which was mounted on the left-hand panel, was recovered damaged and badly burnt. All the warning flags were visible. The artificial horizon section of the unit had frozen at an attitude of approximately 5° nose-down and a very small amount of right wing down. The flight director command bars were in view and appeared to be commanding a small amount of pitch down and a small amount of right roll. Due to impact and fire damage the pitch and roll command indications could not be assessed as being reliable pre-impact evidence. The compass section of the flight director had frozen with the lubber line opposite 154° on the azimuth card. The course arrow was found at 154° and the heading bug at 120° on the azimuth card. The VOR indicator to/from pointer was showing a to indication. The power supplies for the flight director display unit were aircraft generated 26 volts AC and 28 volts DC; failure of either of these would have caused the unit to freeze.

c. Radio barometric encoding altimeter

This unit, which was mounted on the left-hand panel, was recovered very badly damaged and burnt. The barometric digital altitude read-out display was frozen at 18,900 feet. The atmospheric pressure read-out was set at 29.91 inches of Hg and 1012.5 millibars. It was not possible to determine the position of the electrical power failure warning flag. The power supply for this unit was aircraft supplied 28 volts DC, the failure of which would result in the barometric digital altitude read-out freezing at the altitude indicated at that time.

d. Clock

The clock, which was mounted on the left-hand panel, was recovered badly damaged with the hands frozen at 1939 hrs.

e. Barometric altimeter (right-hand panel)

The barometric altimeter was recovered with relatively little impact damage but badly burnt. The atmospheric pressure read-out was displaying 1000 millibars. The barometric altitude analogue display indicated 100 feet.

f. Course indicator (right-hand panel)

The course indicator was recovered severely damaged by both the impact and the subsequent fire. The lubber line was opposite 45° on the azimuth card. The power sources for this instrument were aircraft supplied 28 volts DC and gyro unit generated 26 volts AC. Failure of either of these two power supplies would cause the instrument to freeze.

1.12.2.10 *Environmental system*

a. Cabin pressurisation and air conditioning system

All of the cockpit controls for these systems were totally destroyed in the post-impact fire. The bulk of the pressurisation and air conditioning system equipment was mounted in the rear fuselage area and was recovered moderately damaged. Examination of the various valves and components of the system showed that the selections were consistent with normal operation and no pre-in-flight disintegration defect could be found. Examination of the internal chambers within the equipment showed no indication of contamination of any kind. Remains of both of the outflow safety valves were found within the nose avionics bay area. Both of these valves had been severely disrupted by the ground impact. Examination of the remains of these valves could reveal no evidence of premature failure or defect.

b. Oxygen

The cabin and cockpit oxygen system was almost totally destroyed by the post-impact fire. The oxygen regulator assembly, mounted on the inside forward right-hand cockpit wall, was intact and the altitude adjusting knob system shut-off valve was found to be open. The oxygen supply bottle mounted in the rear fuselage was intact and undamaged. The shut-off valve for the oxygen supply bottle was in the open position but because of the disruption to the supply pipes forward to the cabin and cockpit no oxygen was present within the supply bottle.

1.12.2.11 *Trajectory and wind drift plot*

A wind drift plot was made, using accurate meteorological radio-sonde data and a selection of wreckage items taken from along the whole wreckage trail. The plot indicated that the initial disintegration of the aircraft structure occurred at approximately 20,000 feet, 1 to 1½ miles to the south of the fuselage impact site. There were subsequent secondary disintegrations that occurred between approximately 20,000 feet and 16,000 feet. There was considered to be margin for error in the way the wind drift plot was carried out. The mode of descent of the wings was not known and it was therefore not possible to determine the point at which they separated from the fuselage. The mode of descent of smaller items of wreckage associated with the wing/fuselage interface structure and empennage components were assessed with reasonable accuracy and these plots indicated that the wing centre structure and rear right-hand empennage structure were the initial areas of the airframe that were disrupted. Because it was not possible to assess with any confidence the aircraft's ground speed and attitude at 20,000 feet, a trajectory plot was not carried out.

1.13 Medical and pathological information

The occupants of the aircraft all died as the result of multiple injuries.

The post-mortem examination and report concludes that all the passengers were in normal health and were unlikely to have suffered pre-accident incapacitation. The toxicological report indicates that at least six of the passengers had consumed alcohol before or during the flight, the ethanol analysis for each individual varying from 88 mg/100 ml to 176 mg/100 ml. The ethanol analysis of the passenger recovered from the co-pilot's seat was 107 mg/100 ml.

The commander was in good health and had passed a Class 1 medical examination on 13 September 1984, with no restrictions. Apart from a carboxyhaemoglobin saturation of 5.0% (normal in a smoker) the toxicological result was entirely negative.

There was no medical evidence of a fire or an explosion within the cockpit or cabin area prior to ground impact.

1.14 Fire

After the disintegration, fires occurred on both wings and in the luggage bay area. During the descent both wing fires extinguished themselves. After ground impact, the luggage bay fire spread forward to the cabin area and destroyed most of the cockpit and cabin. The majority of the flight instruments and interior controls were also destroyed.

The East Sussex Fire Brigade was alerted to the possibility of an aircraft accident by the police at 1855 hrs, and despatched appliances to an agreed rendezvous point.

Whilst en-route, the location of the accident site was established and the appliances were re-routed to the B2105 road below the ridge upon which the more substantial items of wreckage were situated.

After an initial assessment of the site of the still burning fuselage, two appliances were directed to the top of the ridge. The fire was rapidly extinguished by the spray application of an estimated 600 litres of water. The fire services maintained a presence throughout the night and the following day, and made a valuable contribution to the search for suvivors and wreckage in difficult conditions.

1.15 Survival aspects

The aircraft suffered an in-flight disintegration at altitude and the accident was non-survivable.

1.16 Tests and research

1.16.1 Radar recording

A recording of the secondary returns to the radars at Ventnor, Isle of Wight and London Heathrow covering the track of the aircraft was available from 1823 hrs to 1840:34 hrs. The recording showed that there was no conflict between the accident aircraft and any other traffic. Appendix 1 shows the track of the aircraft over this period together with the RTF communications between London Control and the commander.

The final portion of the radar information was corrected to ground level datum by the Accidents Investigation Branch and the last 2 minutes 50 seconds was plotted by the Ordnance Survey on a large scale map. From this was derived a track and ground speed from which in turn was calculated the heading and calibrated airspeed (CAS) of the aircraft. Appendix 4 shows the final 1 minute 20 seconds of flight before loss of secondary radar contact, together with the RTF communications between London Control and the pilot.

Following the loss of secondary radar contact at 1840:34 hrs primary radar returns from falling wreckage continued to be recorded for a further 10 minutes. During this time the strong southerly wind carried these lighter items of wreckage as far as 10 nautical miles north of the main accident site before contact was lost.

1.16.2 Analysis of radar plot

At 1826 hrs, after having maintained a steady heading for approximately two minutes, the aircraft began a gentle turn to the left while maintaining height. The turn continued until the heading had changed through about 90° in 4½ minutes, at which point London Control requested the aircraft's heading. The commander reported that the autopilot had "dropped out" and the aircraft began a normal rate of turn to the right until the south easterly heading was regained.

At 1832:30 hrs the track of the aircraft was measured as 126°T with a ground speed of 208 kt. The aircraft heading required to maintain this track was calculated to be 143°M which showed good agreement with the commander's stated heading of 140°. Between 1837:03 hrs and 1839:16 hrs the aircraft was flying on a straight track of 137°T with a ground speed of 204 kt. The heading required to achieve this track was calculated to be between 153° and 154°M. At 1839:16 hrs the aircraft began to diverge left from this straight track into its final turn.

The maximum rate of the final turn was calculated to be less than 4°/second and this was reached immediately after the RTF exchange, as the aircraft was crossing the coast on a north easterly track. The calculations showed that for most of the turn there was little overall change in CAS and that there was no major departure from the cruising speed of 170 kt until the rate of turn had decreased and the aircraft was tracking approximately north. At this stage there was a marked reduction in the height being transponded. This was consistent with a steep dive which would in turn be associated with a rapidly increasing CAS. It was estimated that the maximum permitted airspeed (V_{mo}) for the aircraft was exceeded at a height of approximately 20,000 feet.

1.17 Additional information

1.17.1 Flutter analysis

During the development of the Aero Commander 690 series of aircraft a comprehensive flutter substantiation analysis was carried out by the manufacturer. The report covered, in considerable depth, the flutter susceptibility and characteristics of the airframe and control surfaces. A number of conclusions were made in the report which referred to the control surfaces. During the consideration of this report an accident occurred to a prototype model 690 aircraft which was caused by flutter of the rudder tab due to inadequate tab actuator stiffness. An independent flutter analysis was carried out which established that the production standard aircraft would be flutter free. Subsequent to the prototype accident the manufacturer extended the initial flutter substantiation analysis and the combined conclusions were actioned on the production aircraft.

1.17.2 Previous accidents and structural strength

Appendix 5 lists eight accidents that have occurred to the Aero Commander 690 series aircraft which have involved in-flight structural disintegrations. All of the cases involved wing failures at station 24 and/or station 160 and subsequent tail area failures. Five of the accidents were possibly weather related and two were declared as "cause unknown". In none of the accidents was there a clear explanation except that the aircraft had been overstressed resulting in catastrophic structural failure. The report of the investigation into the accident to Aero Commander 690B D–IKOC in France in 1981 expressed doubts about the structural integrity of the type and questioned the structural analyses. The manufacturer carried out their own investigation and determined that the loading required to fracture the wing at station 24, with the engine on, was 6.3 "g".

1.17.3 Subsequent flight in a similar aircraft type

Following the accident, a flight was made in a similar aircraft type to assess the effect of disengaging the autopilot. With the aircraft in cruising flight at 25,000 feet it was noted that the extent to which the rudder or aileron control circuits were out-of-trim had a marked effect on the response of the aircraft when the autopilot was disengaged. For example, with a moderate amount of left rudder trim applied the aircraft began to diverge to the left rapidly and loose height as if entering a spiral dive, requiring the pilot to intervene within a few seconds. Lesser amounts of rudder out-of-trim produced lesser rates of divergence.

During the flight, while an emergency was being practised which required the pilot to remove his headset, it was noted that a number of the rocker switches in the overhead panel were accidentally knocked and their positions changed.

1.17.4 *Confidential human factors incident reporting system*

The confidential human factors incident reporting system (CHIRP), which has been in operation in the UK since 1983, was interrogated for incidents where a warning or important event on the flight deck had passed unnoticed for a significant period of time. At least 36 incidents were recorded and in some of these the event was eventually noticed by a member of crew other than the handling pilot.

2. Analysis

2.1 General

It was clear that the aircraft suffered a major in-flight structural failure and the wind drift plot showed that the initial disintegration was at approximately 20,000 feet with secondary disintegration occurring between 20,000 feet and 16,000 feet. The southerly wind scattered the wreckage northwards over a distance of some 10 miles. The intense ground fire consumed much of the flight deck and passenger cabin, destroying valuable evidence. Evidence from flight recorders would have greatly assisted the investigation but there were none fitted to the aircraft and neither were they required by law. The latest amendments to ICAO Annex 6 recommend that multi-engined turbine powered aircraft greater than 5700 kg, certificated before January 1987 be fitted with a cockpit voice recorder and flight data recorder. However, it is considered that where possible multi-engined turbine powered public transport aircraft should be fitted with a cockpit voice recorder and flight data recorder.

Detailed examination of the wreckage indicated that there had not been any failure of the airframe before the start of the disintegration sequence. The right wing had been subjected to severe download which had caused sufficient downward bending of the wing section inboard of the engine to allow the right propeller to contact the fuselage side and produce the 27 inch tear at frame 117. During the disintegration sequence which immediately followed there was clear evidence that the wing/fuselage structure had failed in severe upload and that as the right wing broke away it travelled rearwards and struck the rear fuselage immediately forward of the tailplane and the tailplane itself. This had caused major disruption to the empennage which had then broken into many pieces.

Laboratory examination of the material from the areas of the wing root and fuselage showed no pre-disintegration defect or fatigue and that the material conformed to the manufacturer's specifications. There was no evidence of faulty construction or maintenance of the aircraft.

There can be little doubt that the structural disintegration occurred because the aircraft was subjected to vertical loading in the normal axis beyond the design capabilities of the wing/fuselage structure. The possible ways in which the aircraft could have encountered such a regime of flight below 20,000 feet from a cruising condition at 25,000 feet were therefore examined in detail.

2.2 Evidence specific to EI–BGL

There were a number of items of evidence related to the accident aircraft which were considered significant and these are discussed below.

2.2.1 The compasses

The commander's flight director indicator was recovered in a badly fire damaged condition. The compass part of this combined instrument was reading a heading of 154°M and the failure warning flag was showing. This instrument was servo driven and was dependent upon the aircraft's 28 volt DC supply, a 26 volt AC supply from either of the two inverters located in the rear fuselage, and a heading signal from an AC powered directional gyro also in the rear fuselage. The loss of either the DC or AC supply would cause the instrument to "freeze" at the indication showing at the time of the power loss and the failure warning flag to show.

The compass indicator in the right-hand instrument panel was also servo driven and would "freeze" if the 28 volt DC supply was lost. However, the DC powered directional gyro (also in the rear fuselage) which fed this indicator contained its own static inverter and the system was therefore independent of the two main inverters. The instrument was recovered in a badly fire damaged condition and was indicating a heading of 045°M.

During the period between 1837:03 hrs and 1839:16 hrs, immediately preceding the final turn when the aircraft was tracking towards Dieppe, the radar recording enabled the heading of the aircraft to be calculated as between 153° and 154°M which was within 1° of the heading shown on the commander's compass. This cannot be considered as mere coincidence and it strongly suggests that the 26 volt AC supply to the left hand instrument was lost at 1839:16 hrs, just before the aircraft departed from a steady heading and entered its final turn. This would have caused the commander's instrument to remain at the pre-turn heading while allowing the compass in the right-hand panel to continue indicating correctly until DC power was lost at a later point in the accident sequence.

2.2.2 *The attitude indicator and attitude gyro*

The attitude indicator part of the commander's flight director was recovered indicating an aircraft attitude of 5° nose down and a very small amount of right wing down with all of the failure flags showing. This servo driven instrument required a 28 volt DC supply from the aircraft's system, a 26 volt AC supply from the selected main aircraft inverter and attitude signals from the attitude gyro which was located in the rear fuselage. Loss of either of the electrical supplies or the attitude signals would cause the cockpit indicator to "freeze" at the attitude showing at the time and bring all the failure warning flags into view. The attitude gyro was powered by 115 volts AC from the selected main aircraft inverter. Examination of the attitude gyro, in addition to finding no fault with the unit, showed that the unit (and therefore the aircraft) was at an attitude in excess of approximately 70° of bank or ± 85° of pitch at the time of loss of the 115 volt supply. The discrepancy between the attitude gyro position and the cockpit indication suggests that it was the 26 volt AC supply to the flight director which was lost at an early stage, while the aircraft was still in a near normal cruising attitude, but that the 115 volt AC and main 28 volt DC supplies were not lost until later, by which time the aircraft had already adopted an extreme attitude.

The flight director command bars were found "frozen" in the centre of the display. A small amount of right roll was demanded although the mechanism was vulnerable to disruption and the indication was considered to be unreliable. With power on and no mode selected the command bars are driven out of sight, but with power off they "freeze" in the position indicating at the time. The fact that they were visible therefore indicates that, at the moment of power loss, a flight director mode was selected. It is not possible to be certain which lateral mode was selected but with an aircraft heading of 154° on the compass, the heading bug set to 120° and the course pointer set on 154°, it seems more probable that the navigation mode rather than the heading mode was selected.

2.2.3 *The radar track and altitude information*

The ground track of the aircraft shown at Appendix 4 was obtained from a recording of the secondary radar returns from the aircraft's transponder received by the London and Ventnor radars. The recording shows that the DC powered transponder was operating throughout the final turn and that at 1840:29 hrs it transmitted the height of the aircraft as FL 222. There was one final transponder return 5½ seconds after that at FL 222 but this did not contain height information. The height encoding altimeter, which presented the commander with a digital readout of height, was recovered indicating 18,900 feet. This instrument was also DC powered and it would have "frozen" at the height indicated when DC power was lost. This cannot have been later than the point at which the right propeller cut into the fuselage at frame 117 and severed the entire electrical loom. There can, therefore, be little doubt that DC power was available until the moment of structural failure which occurred just below 19,000 feet.

The transponded altitude information on the radar recording indicated that soon after the start of the final turn the aircraft began slowly to lose height. The gradual height loss increased as the turn continued until approximately 1840:23 hrs when it increased sharply. At the time of the transponded height of FL 222 it was estimated that the aircraft was descending at a rate of between 20,000 and 25,000 feet per minute and accelerating rapidly. The airspeed calculations based on the radar recording showed that there was no major departure from the cruising speed of 170 kt until the final rapid loss of height. It was estimated that the maximum permitted operating speed (V_{mo}) for the aircraft was exceeded at approximately 20,000 feet.

2.2.4 *The ATC tape recording*

Examination of the ATC recording showed that at 1840:04 hrs the London controller instructed the aircraft to call Paris on 132.0 MHz. The aircraft VHF communication sets are DC powered and the fact that, at 1840:08 hrs, the commander replied "ONE THREE TWO ZERO GOOD DAY SIR" is further proof that DC power was available during the final turn. Furthermore, the manner of his reply gave no indication that he was aware of any problem or emergency although by this time (52 seconds after the departure from the straight track to Dieppe) the aircraft had already altered heading through approximately 70° to the left and lost 200 to 300 feet of altitude.

At 1840:15 hrs a momentary transmission with no speech was recorded on the ATC tape but it was not determined if this was from EI–BGL or some other source. A truncated distress call was overheard by a reserve lifeboatman at about the time of the accident but this call was not heard by London ATC or recorded on the ATC tape. In view of the lack of any other known emergencies at the time, it is considered that this call could have been made only by EI–BGL. Since normal transmissions had been made by the aircraft up to 1840:08 hrs, the distress call must have been made, when the commander realised the seriousness of his situation, at some time between 1840:08 hrs and the moment of structural failure.

2.2.5 *The loss of the 26 volt AC supply*

The evidence strongly suggests that the loss of the 26 volt AC supply occurred immediately before the beginning of the final turn whereas the DC and probably the 115 volt AC supplies were available until the moment of structural failure. No fault was found in either of the main inverters and after detailed examination of the wreckage no reason for the 26 volt AC power loss was found. Much of the electrical system was destroyed by the ground fire, including most of the cockpit area, and the possibility of a component failure, faulty insulation in the wiring loom or a loose connection could not be eliminated.

It was noted in a similar aircraft that the banks of "rocker" type switches in the roof panels were close to the heads of the occupants of the pilot's seats. In particular the autopilot switch, the switch controlling the inverters, and the radio 1 and radio 2 bus switches (which also control the AC distribution) were located immediately above the head of the occupant of the right-hand pilot's seat. During a subsequent flight on a similar aircraft type, when an emergency was being practised which required the pilot to remove his head-set, a number of the rocker switches were accidentally knocked and their positions changed. The possibility that the AC power loss in EI–BGL resulted from one of these switches being operated accidentally was examined. Unless certain circuit breakers positioned on the cockpit left-hand side were pulled, it was not possible for the crew to have switched off the 26 volt AC supply without simultaneously having selected off the 115 volt supply by means of the inverter control switch. This is inconsistent with the evidence that the 26 volt supply was lost before the 115 volt supply and suggests that action by the commander (or one of the passengers) is unlikely to have been the cause of the electrical failure.

In addition to the failure of the commander's flight director the loss of the 26 volt AC supply would have caused the AC failure warning light on the central warning panel to illuminate. Such AC power failure would also have caused the autopilot to disengage automatically and illuminated the auto-pilot disconnect warning light on the central warning panel. An autopilot disconnect audio warning would have alerted the commander even if the visual warnings had passed unnoticed for whatever reason. Airworthiness Authorities should consider the requirement for an autopilot disconnect audio warning.

2.2.6 *The disengagement of the autopilot*

The disconnection of the autopilot in cruising flight would have permitted the aircraft to respond initially to any out-of-trim forces in the rudder and aileron control circuits which were previously being opposed by the auto-pilot servos. (The autopilot trims in pitch by driving the elevator trim motor and therefore any residual out-of-trim force in this control circuit would have been minimal.) The aircraft would then have followed a flight path dependent upon its natural stability characteristics. During a subsequent flight in a similar type the effect upon the aircraft was observed of disengaging the autopilot with varying amounts of out-of-trim in the rudder and aileron control circuits. With a moderate amount of left rudder trim the aircraft diverged left and began to lose height as if entering a spiral dive. The divergence was sufficiently pronounced that the pilot had to intervene manually within a few seconds. Lesser amounts of rudder trim induced a slower divergence.

Examination of the radar recording of EI–BGL revealed a significant incident at 1826 hrs when the aircraft began a very gradual left turn after having maintained a constant heading for the previous two minutes. The turn continued at a rate of approximately 1/3° per second for some 4½ minutes by which time the aircraft had changed heading through about 90° to the left. When the commander was requested for his heading by London Control at 1830:46 hrs he replied "I'M SORRY THE AUTOPILOT'S JUST DROPPED OUT I'M TURNING BACK RIGHT". Since there is no reason to doubt the commander's explanation it seems likely that the heading hold disengaged, thus allowing a slow deviation from heading while leaving the height hold operative. Alternatively, it is possible that the autopilot disengaged altogether allowing a minimal out-of-trim condition to turn the aircraft slowly to the left at a rate which was not enough to disturb the aircraft in pitch and therefore altitude. In either case accidental operation of the turn control knob or the autopilot disconnect switch by the passenger in the co-pilot's seat could not be eliminated.

It is probable that at 1839:16 hrs when the autopilot disengaged, with the loss of the 26 volt AC supply, a slightly greater out-of-trim condition caused the aircraft to diverge to the left in the manner recorded on radar, possibly developing into a steep spiral dive.

2.3 Other possible causes

Information on other accidents involving structural failure to the Rockwell Commander 690 A, B & C aircraft was obtained from a number of sources including the NTSB and ICAO records. A summary is at Appendix 5. Of the eight accidents listed, a cause was not given for two of them. The causes ascribed to the remaining six accidents, which may be relevant, were assessed against the evidence available in the case of EI–BGL. These together with other possible causes are discussed below.

2.3.1 Turbulence and icing

The meteorological aftercast for the area of the accident indicated that slight to moderate turbulence could have been experienced in cloud up to FL 250 and moderate clear air turbulence between FL 250 and FL 380. No reports of turbulence in the vicinity of the route were received and, furthermore, there was no evidence of turbulence on the flight data recorder of a BAe 1–11 which had flown through the accident area twenty minutes before the accident. The possibility of the aircraft encountering wake turbulence from another aircraft was ruled out by the radar recording which showed that there were no other aircraft in the area at FL 250 immediately before the accident. At FL 250 the aircraft was well clear of the light or moderate icing which existed in cloud between 10,000 feet and 20,000 feet. It is, therefore, most unlikely that EI–BGL encountered turbulence or icing of sufficient severity to cause difficulties.

2.3.2 *Pilot incapacitation*

There was no evidence from the post mortem examination of the commander to suggest that a medical factor might have caused or contributed to the accident.

2.3.3 *Pilot disorientation or distraction*

The possibility of disorientation of the commander could not be excluded, but there was no evidence to support this as being a primary cause of the accident. His apparently normal RTF reply at 1840:08 hrs suggests rather that he was unaware of anything unusual happening, although the aircraft had by then diverged approximately 70° from the intended heading and begun to lose height. It is possible that this was due to his lack of attention or distraction. The previous incident at 1826 hrs when the steady divergence of the aircraft from track went unnoticed for some 4½ minutes also supports this view. On this occasion it is evident that his attention was regained only when the London Controller requested the aircraft's heading.

The scenario of an AC failure as the initiator of the accident sequence would have required the commander to fail to respond to the two warning lights on the central warning panel, changes in the mode annunciator lights and all the failure warning flags on his flight director indicator. It is difficult to imagine these indications of potential trouble passing unnoticed for such a length of time unless the commander had been distracted. The possibility that passenger behaviour might have provided such a distraction was examined.

The pathological evidence showed that the ethanol levels of the passengers ranged between 88 mg/100 ml and 176 mg/100 ml. It is therefore reasonable to suggest that, within the relatively small confines of the cabin, a party atmosphere existed which could have been a considerable distraction to the commander. The aircraft was fitted with seven seats in the passenger cabin and the only additional place available for the eighth passenger was the co-pilot's seat, A passenger was recovered from this seat with an ethanol level of 107 mg/100 ml but it was not possible to say whether he was strapped in or if he was in the process of entering or leaving the seat during the final turn. It was not possible to assess whether or not the seat belts of the pilot's seats were fastened at the time of the structural disintegration. In addition to providing a possible distraction to the commander this could also have afforded potential for the inadvertent operation of cockpit switches. Regulatory Authorities should consider prohibiting the carriage of passengers in the co-pilot's seat of such aircraft.

The only known event during the aircraft's final turn which might have distracted the commander was the RTF exchange with the London Controller following which he could have been involved for several seconds selecting the Paris frequency on one of the aircraft radios. However, the confidential human factors incident reporting (CHIRP) system which has been operating within the UK since 1983 has a record of at least 36 incidents where a warning or important event on the flight deck has passed unnoticed for a significant period of time. In some cases the event was eventually noticed only by a second crew member. The carriage of a second crew member has other advantages and Regulatory Authorities should therefore review the policy of single pilot operation of such aircraft types.

2.3.4 *Engine failure*

No pre-existing condition was found in either engine or their associated fuel control units which would have interfered with normal operation. Rotational witness marks on the compressor shrouds indicated that both engines were turning at high rpm (consistent with normal cruise power settings) at the moment of the structural failure. The left engine in addition had low rpm rotational witness marks suggesting that its propeller was windmilling and driving the engine at the time it impacted the ground. There was evidence that the right engine was not rotating at ground impact, however, the drive shaft between the engine and the propeller gearbox had failed while under load and the propeller was therefore not capable of driving the engine. The contact between the right propeller and the fuselage side at frame 117 clearly was made while the engine was delivering power and this would also have produced sufficient loading to cause the drive shaft failure.

It was concluded that both engines were operating normally until the moment of disintegration.

2.3.5 *The distribution bus RCB relay*

After detailed examination of the aircraft's DC electrical system no fault was found other than within the distribution bus RCB relay. This relay had not suffered external impact or fire damage but the main contactor bar had been deformed to the extent that the relay was incapable of making contact and therefore of passing current. Its appearance and the results of the metallurgical analysis and hardness tests suggested that it might have been considerably overheated. In the absence of external fire damage the only possible way in which such overheating could have occurred was by gross electrical overloading. It was theorised that the bar, in a heat softened condition, could have been deformed by ground impact decelerative forces but there was no conclusive proof of this. The DC distribution bus which was energised by this relay provided power for a number of services including the height encoding altimeter which in turn sent height information to the aircraft transponder. The radar recording shows that height information was transmitted from the aircraft throughout the final turn. In addition, the height encoding altimeter was found frozen at 18,900 feet, so there can be no doubt that the relay was passing current until the time of the disintegration.

Although there was no certain explanation for the 'as found' condition of the RCB relay it was concluded that it occurred as a feature of the structural disintegration and did not cause or contribute to the accident.

2.3.6 *In-flight fire*

Each wing exhibited a pattern of fire and sooting in a direction from root to tip which could only have occurred after the structural disintegration when these items were falling with the heaviest (ie, wing root) end foremost.

Much of the main fuselage structure had been destroyed by a severe fire. A piece of fuselage skin which had been detached from frame 117 by the contact of the right propeller was recovered some 2½ miles from the fuselage and showed no signs of fire damage or sooting. A number of articles from the baggage bay which were found 6,000 feet from the fuselage showed minor evidence of fire damage, however, the bulk of the baggage bay contents were not fire affected although they were heavily contaminated with aviation fuel. The aircraft structure rear of the baggage bay forward door frame (including most of the baggage bay itself) showed no signs of fire or sooting.

There was no evidence in the wreckage to indicate that fire was present before the moment of structural disintegration. This was further supported by the results of the post-mortem examinations which showed that the commander and passengers had not inhaled toxic fumes and that their blood levels of carboxyhaemoglobin were within the normal range.

It was concluded that fire had not been present up to the moment of structural failure but that it was a feature of the subsequent disintegration and had continued in the main fuselage after ground impact.

2.3.7 *Explosive decompression/bomb*

It was suggested at the time of the accident that a bomb might have been responsible. There was no evidence to support this and the wreckage exhibited none of the distinctive features which would be associated with such a device.

The fuselage structure and exits which had not been consumed by fire were examined for evidence of failure which could have resulted in an explosive decompression. The evidence suggests that the main cabin door was closed and locked at the time of fuselage ground impact. The emergency exit on the right-hand side of the fuselage was found properly in place. The right-hand picture window fittings had not failed prior to ground impact although the window itself had been disrupted earlier in the accident sequence. No trace of the left-hand picture window or its fittings was found but this area of the fuselage had been consumed by fire. The components of the cabin pressurisation system were disrupted by the ground impact but there was no evidence of premature failure or defect.

There was therefore no evidence that a sudden loss of cabin pressure had occurred before the structural failure but neither was it possible to eliminate this possibility on the basis of the wreckage analysis. Such an event could not have passed unnoticed by the commander and it is reasonable to suppose that he would have declared an emergency in such circumstances. However, the aircraft began to diverge from track at 1839:16 hrs and no emergency was declared during the next 52 seconds. In addition his final RTF call at

1840:08 hrs was normal and it is therefore considered that an explosive decompression played no part in the accident sequence until the moment of structural disintegration.

2.3.8 *Control failure/flutter*

The examination of the flying control surfaces and their associated control systems gave no indication of a pre-disintegration failure.

Although part of the elevator trim control circuit was never found it was established that the main elevator control ciruit was serviceable prior to the disintegration. As the elevator trim control circuit was duplex in the area of the items that were not recovered it is highly unlikely that both systems would fail at or about the same time. Therefore, it is assessed that a failure of the elevator trim control circuit did not occur prior to the disintegration.

During the investigation consideration was given to the possibility of control surface flutter having occurred. Flutter involves movement of the control surfaces in an oscillatory motion that can become sufficiently severe to cause damage to the aircraft. Associated with flutter, especially on the main control surfaces, would be a buzz or vibration that would be felt through the airframe and the control system which would alert the pilot. There was no evidence on the RTF recording that any buzz occurred or that the pilot was aware of anything untoward happening. The aileron control surfaces did exhibit signs of having travelled to their extreme positions but the damage sustained was characteristic of very low frequency oscillations over an extremely short period of time, which is not consistent with flutter. This damage was assessed as being caused by the wings tumbling after their separation from the fuselage. With no evidence of flutter having occurred on any of the other flying control surfaces it is considered that flutter did not occur prior to the structural disintegration.

2.4 Summary

There was insufficient evidence to enable a definite cause of the accident to be established, however, there was no indication that the following contributed to the accident: manufacturing or structural defects; excessive vertical acceleration caused by atmospheric turbulence or wake turbulence; icing; engine failure; DC electrical failure; 115 volt AC electrical failure; in-flight fire; explosive decompression; an explosive device; control failure; and, flutter.

The radar and RTF recordings, in conjunction with the evidence of the aircraft's compasses and attitude gyro, suggest that the chain of events leading to the structural disintegration was precipitated by the loss of the 26 volt AC electrical supply due to an unknown cause. This resulted in the disengagement of the autopilot which would in turn have allowed the aircraft to diverge progressively from its intended track and altitude. It is not possible to say why this divergence was not corrected by the commander before the aircraft had achieved an extreme attitude. With the aircraft in a steep dive and rapidly gaining speed the structural disintegration probably occurred as the commander attempted recovery.

3. Conclusions

(a) *Findings*

(i) The commander was properly licenced for the flight.

(ii) The aircraft had a valid Certificate of Airworthiness and its documentation was in order.

(iii) The aircraft had been maintained in accordance with an approved maintenance schedule and its defects log contained no outstanding items relevant to the accident.

(iv) The aircraft was marginally overweight on take-off but this had no relevance to the accident.

(v) At the time of the accident the aircraft was within the prescribed weight and centre of gravity limits.

(vi) The aircraft was in cruising flight at FL 250 when it entered a turn to the left and began to lose height.

(vii) The turn continued and the rate of descent then increased rapidly.

(viii) The aircraft suffered a complete structural disintegration at a height of about 19,000 feet.

(ix) There was no evidence of a pre-disintegration fire, engine failure, explosion, or structural or mechanical defect.

(x) There was no evidence of incapacitation of the commander.

(xi) There was no evidence that atmospheric turbulence or icing contributed to the accident.

(xii) There was evidence that the 26 volt AC electrical supply was lost immediately before the start of the final turn.

(xiii) The loss of the 26 volt AC electrical supply would have resulted in the disengagement of the autopilot and failure of the flight director system.

(xiv) It was not possible to determine the cause of failure of the 26 volt AC supply.

(xv) Following the disengagement of the autopilot the aircraft began to turn left, uncorrected by the commander, and probably entered a steep spiral dive.

(xvi) The structural disintegration resulted from overstressing in upload, probably as the commander attempted to recover from the dive.

(b) *Probable cause*

The in-flight disintegration of the aircraft was probably caused by overstressing during an attempted recovery from an extreme attitude in a spiral dive. A probable contributory factor was the commander's lack of awareness of the loss of the 26 volt AC supply to the autopilot and flight director system.

4. Recommendations

It is recommended that Regulatory Authorities should consider:

4.1 Reviewing the policy of single pilot operation for public transport aircraft.

4.2 Prohibiting the use of the co-pilot's seat by passengers.

4.3 Requiring multi-engined turbine powered public transport aircraft to be fitted with a cockpit voice recorder and flight data recorder.

4.4 Requiring public transport aircraft to be fitted with an autopilot disengagement audio warning.

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