Department of Transport and Regional Development

Bureau of Air Safety Investigation

INVESTIGATION REPORT
9304023

De Havilland DH-104 Series 5 Dove VH-DHD
1 km south-west of Essendon, Victoria
3 December 1993

BASI
Bureau of Air Safety Investigation

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CONTENTS

ABBREVIATIONS

AGL	Above ground level
AOC	Air Operator's Certificate
AMSL	Above Mean Sea Level
ASDA	Accelerate and Stop Distance Available
Avgas	Aviation gasoline
CAA	Civil Aviation Authority
CAO	Civil Aviation Order(s)
EAS	Equivalent Airspeed
ESuT	Eastern Summer Time
ft	Foot/Feet
g	Gram(s)
hPa	Hectopascal(s)
IAS	Indicated Airspeed
imp gal	Imperial Gallon(s)
ISA	International Standard Atmosphere
KEAS	Knots Equivalent Airspeed
KIAS	Knots Indicated Airspeed
km	Kilometre(s)
kt(s)	Knot(s)
L	Litre(s)
lb/in^2	Pound(s) per Square Inch
m	Metre(s)
min	Minute
mm	Millimetre(s)
MAP	Manifold Air Pressure
MTOW	Maximum Take-off Weight
NOSAR	No Search and Rescue
OAT	Outside Air Temperature
OEI	One Engine Inoperative
oz	Ounce(s)
PA	Pressure Altitude
QNH	Altimeter subscale setting to show altitude above mean sea level
RPM	Revolutions per Minute
TEL	Tetraethyl Lead
TODA	Take-off Distance Available
TORA	Take-off Run Available
V_2	Lowest speed at which aircraft complies with required handling criteria for climbout following engine failure at takeoff
V_{MCA}	Minimum Control Speed (Air) with OEI
V_{TOSS}	Take-off Safety Speed

Note 1 The Civil Aviation Authority (CAA) was replaced in July 1995 by Airservices Australia (AA) and the Civil Aviation Safety Authority (CASA). CASA is the aviation safety regulator.

VH-DHD following crash into houses near Essendon Airport 3 December 1993.

SYNOPSIS

The flight was planned as a night charter dinner flight over Melbourne. On board the De Havilland DH-104 were the pilot, a dinner hostess and eight passengers.

After carrying out engine runups and pre-takeoff checks, including selecting 20° of flap, the pilot initiated takeoff from Essendon Airport runway 17, some 23 minutes before last light. Wind conditions were light and variable. Just as take-off safety speed was attained, somewhere between liftoff and 50 ft, the right engine lost power and the aircraft yawed to the right. The pilot momentarily assessed the problem as a partial engine failure and selected the landing gear up, but the landing gear failed to retract. However, by recycling the gear selector he was then able to successfully retract the landing gear. By this time the airspeed had decayed to a point well below take-off safety speed, and continued decreasing to the minimum control speed (air). In order to maintain directional control the pilot reduced power on the left engine, but did not manage to raise the flaps or feather the right propeller before the aircraft crashed into a residential area adjoining the aerodrome. About one minute had elapsed from initiation of takeoff until the accident.

Investigation revealed that the vertical drive shaft, which drives the right engine fuel control unit, had failed in torsional overload when the geared fuel pump drive shaft in the fuel control unit had seized on its surrounding bush. Following the vertical drive shaft failure, the right engine failed due to fuel starvation.

Analysis of DH-104 performance indicated that, at the time of the right engine failure, it was possible for the aircraft to achieve a positive rate of climb, assuming that the engine failure drills were performed promptly and correctly, and proceeded without interruption. However, when the landing gear failed to retract on the first attempt, any possibility of the pilot being able to attain the required aircraft performance was lost. As a result, he was probably forced to abandon completing the emergency procedures in order to maintain control of the aircraft.

1. FACTUAL INFORMATION

1.1 History of the flight

The pilot had planned to conduct a night charter flight over Melbourne and Port Phillip Bay, starting from and returning to Essendon Airport. Dinner was to be served in flight.

The pilot gave a safety briefing to the passengers before starting the engines. He completed engine runups and pre-takeoff checks, including selecting 20° of flap. At 2036 ESuT, in daylight, the pilot initiated takeoff on runway 17 using standard take-off power setting of 7.5 lb/in^2 of boost and 3,000 RPM. Wind conditions were light and variable, visibility was about 10 km and the temperature was 19°C.

The aircraft became airborne and, just as it achieved the take-off safety speed of 84 kts, at a height not above 50 ft, the right engine lost power. The aircraft yawed right. The pilot reported to the investigation team that he briefly noticed a reading of 3 lb of boost on the MAP gauge and assessed the problem as a possible partial right engine failure. He then selected the landing gear up but it did not retract. He cycled the landing gear selector once and the gear then retracted. By this time several seconds had elapsed and the airspeed had decayed to 76 kts. The pilot then assessed the airspeed as too low to retract the flaps and left them at 20°. The airspeed continued to decay until V_{MCA}, 72 kts, was reached. When indicated airspeed had further decayed to 68 kts, the pilot reduced power on the left engine to avoid an uncontrollable roll to the right. He was able to maintain wings level and attempted to track the aircraft toward a street but was unable to maintain height. The aircraft collided with powerlines and then struck the roofs of several houses before coming to rest, on its left side, against the front wall of a house. About one minute had elapsed from initiation of takeoff until the accident.

The pilot and all but one of the passengers remained conscious throughout the accident sequence. All occupants were evacuated, some without assistance and others with the assistance of the pilot, other passengers, emergency services personnel or bystanders.

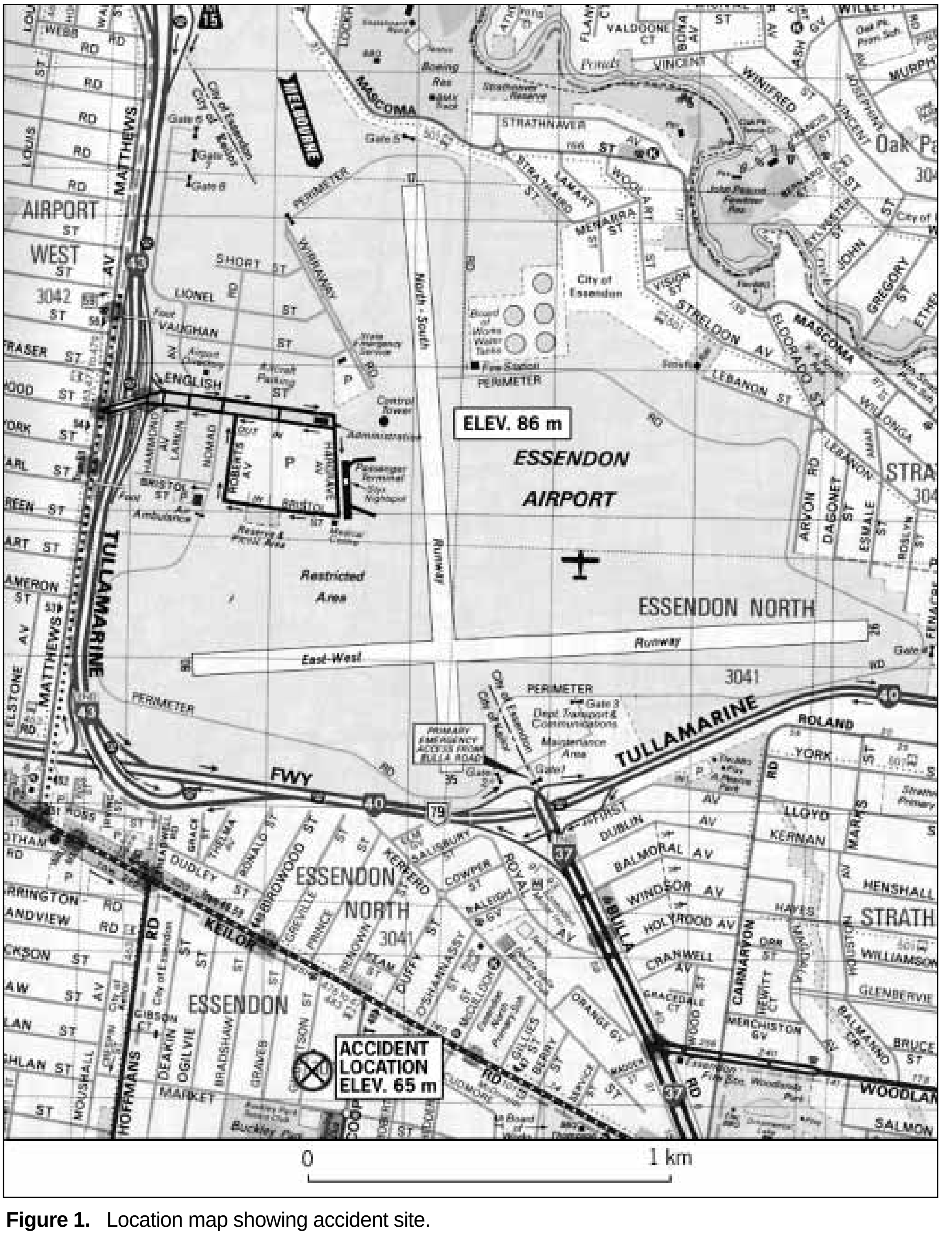

Figure 1. Location map showing accident site.
Map copyright Melway Publishing Pty Ltd. Reproduced from *Melway Street Directory* Edition 21 with permission.

1.2 Injuries to persons

The pilot and four passengers were seriously injured. The other five passengers and one resident of a damaged house received minor injuries.

1.3 Damage to aircraft

The aircraft was destroyed.

1.4 Other damage

Powerlines and six houses sustained damage.

1.5 Personnel information

The pilot in command was the operator/chief pilot. He held an air transport pilot licence (aeroplanes), a current command instrument rating, a valid endorsement on the DH-104 and a valid class-1 medical certificate. His total aeronautical experience was 18,154 hours including approximately 1,500 hours on the DH-104. He was approved by the Civil Aviation Authority to conduct pilot endorsement training on DH-104 aircraft.

For most of his career the pilot in command had been an airline pilot flying heavy jet aircraft, principally the Boeing 727.

1.5.1 Pilot's DH-104 training

The pilot was endorsed on the DH-104 in 1987. He was flight tested in July 1993, by an authorised testing officer, in a DH-104 for a command instrument rating renewal. The authorised testing officer simulated an engine failure in the initial climb after takeoff, but not before the landing gear and flaps had been retracted and the aircraft had accelerated to 10 kts above V_{TOSS}. The pilot performed competently.

1.6 Aircraft information

1.6.1 Aircraft

Manufacturer	De Havilland Aircraft Company, Hatfield, Herts., UK
Model	DH-104 Series 5 'Dove'
Serial number	04104
Registration	VH-DHD
Year of manufacture	1948
Certificate of airworthiness	MBN00589/01 dated 10 March 1988
Certificate of registration	MBN00589/01 dated 4 March 1988
Maintenance release number	03432, valid to 5 August 1994 or 21,266.37 hours
Airframe hours	21,259.13
Maximum take-off weight	3,991 kg
Actual take-off weight	3,688 kg
Weight at impact	3,680 kg
Centre of gravity range	93.9 mm forward of datum to 182.7 mm aft of datum
Centre of gravity actual	62.4 mm aft of datum

1.6.2 Engines

Manufacturer	De Havilland Aircraft Company, Hatfield, Herts., UK
Model	Gypsy Queen 70 Mk2
Serial numbers	Left 66856; right 66863

Approved overhaul period	1,500 hours
Time to next overhaul	Left 334 hours; right 181 hours
Date of last inspection	6 August 1993
Time since last inspection	92.36 hours

1.6.3 Propellers

Manufacturer	De Havilland Aircraft Company, Hatfield, Herts. UK
Type	Metal, 3-blade, feathering
Model	PD/143/312/7
Approved overhaul period	1,600 hours
Time to next overhaul	Left 370 hours; right 27 hours
Date of last inspection	6 August 1993
Time since last inspection	92.36 hours

1.6.4 Fuel control unit—right engine

Manufacturer	Hobson Fuel Controls, Wolverhampton, UK
Model	MK 103/B
Serial number	3371
Year of installation	1981
Approved overhaul period	1,500 hours
Time to next overhaul	753 hours
Date of last inspection	6 August 1993
Time since last inspection	92.36 hours

As it is part of the fuel control unit, the fuel pump does not have its own serial number.

The fuel control unit was last overhauled by an overseas maintenance contractor in 1981.

1.7 Meteorological information

The wind was light and variable, tending to a light north-westerly; QNH was 1009, and temperature was 19°C. There were 2 octas of cloud at 1,000 ft. Last light was at 2059.

1.8 Aids to navigation

Not relevant.

1.9 Communications

The aircraft was in communication with Essendon Tower on 125.1 MHz. After acknowledging his take-off clearance, the pilot made no further radio transmissions.

1.10 Aerodrome information

Runway 17 at Essendon is asphalt, 1,585 m long, 45 m wide, 282 ft AMSL and has a 0.9% slope downward to the south. Both TORA and ASDA are 1,585 m.

1.11 Flight recorders

The aircraft did not carry, and was not required to carry, a flight data recorder or a cockpit voice recorder.

1.11.1 Recorded radar information

Recorded radar information showed the aircraft from a groundspeed of 20 kts in the take-off roll until the impact sequence. The aircraft transponder was set to code 4000 mode C.

The highest altitude recorded was 300 ft AMSL and the highest ground speed recorded was 87 kts. From the time that radar first detected the aircraft, 29 seconds elapsed until the aircraft attained 87 kts. A further 37 seconds elapsed until the aircraft return disappeared from the radar screen.

1.12 Wreckage and impact information

The aircraft came to rest lying on its left side against the front wall of a house about 1 km south of the southern boundary of Essendon Airport.

The impact sequence began when the aircraft collided with powerlines about 0.5 km south of the airport. Continuing south, the aircraft then overflew several houses before striking the roofs of six houses and coming to rest.

The aircraft initially struck the roofs with the underside of the left wing before the leading edge of the left wing, outboard of the engine, struck a house. This impact caused sections of the left wing leading edge to break up and tore away most of the left wing structure at the left main wheel position. The impact of the left wing with the second-last house slowed the aircraft and yawed it to the left. At impact with the ground, the aircraft was moving rearwards and downwards.

1.12.1 Fuselage

The aircraft nose was crushed and displaced left and rearwards to the rudder pedals. The nose remained joined to the fuselage only by control cables. The connections to both pneumatic system air bottles fractured when the nose was crushed.

The fuselage came to rest on its left side against the front wall of a house and was facing in the direction of the airport.

1.12.2 Wings

The main spar was broken at both wing roots. Both engines had separated from their mountings and had moved to the left.

The flaps were substantially damaged. Impact marks adjacent to the flap hinge points were consistent with the flaps being at 20° at impact.

1.12.3 Landing gear

All three landing gear units were found to be retracted.

1.12.4 Control systems

There was no evidence to indicate any fault in the rudder, elevator and aileron systems prior to impact. The control cables showed no evidence of abnormal wear. Trim control cables were intact.

The flap control lever was in the 20° detent.

Throttle, propeller pitch, alternate air and cowl flap cables were intact apart from pulley damage sustained during the impact sequence. Throttle and propeller pitch lever friction assemblies operated normally.

The rudder pedal torque tube mounting bearings and the links between the left and right systems were fractured. The appearance of all fractures was consistent with their having occurred at impact. The rudder gust lock was disengaged.

1.12.5 Propellers

Neither propeller was feathered. The left propeller was more severely damaged, consistent with power on the left engine at impact. The right propeller had experienced less damage, consistent with the right engine not producing power at impact.

1.12.6 Airframe fuel system

Each fuel system selector was found selected to its respective wing tanks. These were likely to have been pre-impact selections.

The airframe fuel system includes rigid metal fuel lines and flexible hoses. The metal fuel lines were fractured in several places whereas the flexible lines had experienced only minor damage. The fuel tanks were not ruptured.

Both left and right fuel systems were examined and tested. Both electric boost pumps were serviceable and capable of producing pressure in excess of the required minimum. However, these pumps provide only a head of pressure of 8–10 lb/in^2 to the fuel control units which require 23–27 lb/in^2 to operate. The engines will not run on electric boost pump pressure alone.

The fuel vents were clear. Each fuel cap was serviceable. All fuel filters were clean. No water was found in the fuel system.

Only about 1 L of fuel remained in each wing tank system, the remainder having escaped through the ruptured lines. An examination of the fuel records revealed that the aircraft would have had approximately 132 L of avgas in each of the main fuel tanks at impact. Tests on the fuel showed it to be uncontaminated avgas 100/130.

1.12.7 Fuel control unit—fuel pump

The vertical drive shaft to the fuel pump within the fuel control unit of the right engine was fractured.

1.12.8 Electrical system

The battery was dislodged from its mount during the impact sequence and remained connected only by one terminal cable. This would have resulted in the airframe electrical systems being deactivated by the time the impact sequence was complete.

1.12.9 Seats and seat belts

The left cockpit seat, occupied by the pilot, remained attached to the airframe at all attachment points. However, the bulkhead, to which the seat was attached, was the point of separation of the cockpit from the cabin. The right cockpit seat, occupied by a passenger, had failed structurally and the seat support mounting had separated from the bulkhead.

The unoccupied right rear passenger seat was the only seat still in position. All other passenger seats had detached due to impact forces.

None of the seat belts had failed. All occupants had remained strapped into their seats throughout the impact sequence.

1.12.10 Detailed examination and testing of components

1.12.10.1 Engines and propellers

After the accident, the right engine was recovered and mounted in a test stand. The engine was primed and started several times but, on each occasion, ran only briefly on priming fuel at low RPM before stopping due to fuel starvation. Examination revealed that the fuel control unit was not being driven by the accessory gear train. The vertical drive shaft had failed, rendering the fuel control unit inoperative.

The left propeller was dismantled to determine blade angles at impact and to check for any pre-impact faults. Blade number one angle was 45° 15', blade number two was 45° 20' and blade number three was approximately 50°. Impact damage to blade number three prevented a more accurate determination of angle. The blade angles confirmed that the left propeller was not feathered at impact. No pre-impact faults were found.

The right propeller was dismantled. No pre-impact faults were found. As the right propeller had sustained much less damage than the left propeller, the blades were straightened and the propeller reassembled and refitted to the right engine in the test stand. No fault was found with the propeller.

1.12.10.2 Fuel control unit vertical drive shaft

Examination of the fuel control unit vertical drive shaft fracture showed it to be consistent with torsional overload. The fracture had occurred on the plane of maximum shear stress. (see fig. 2). No pre-existing condition of the shaft which might have contributed to the failure of the shaft was found.

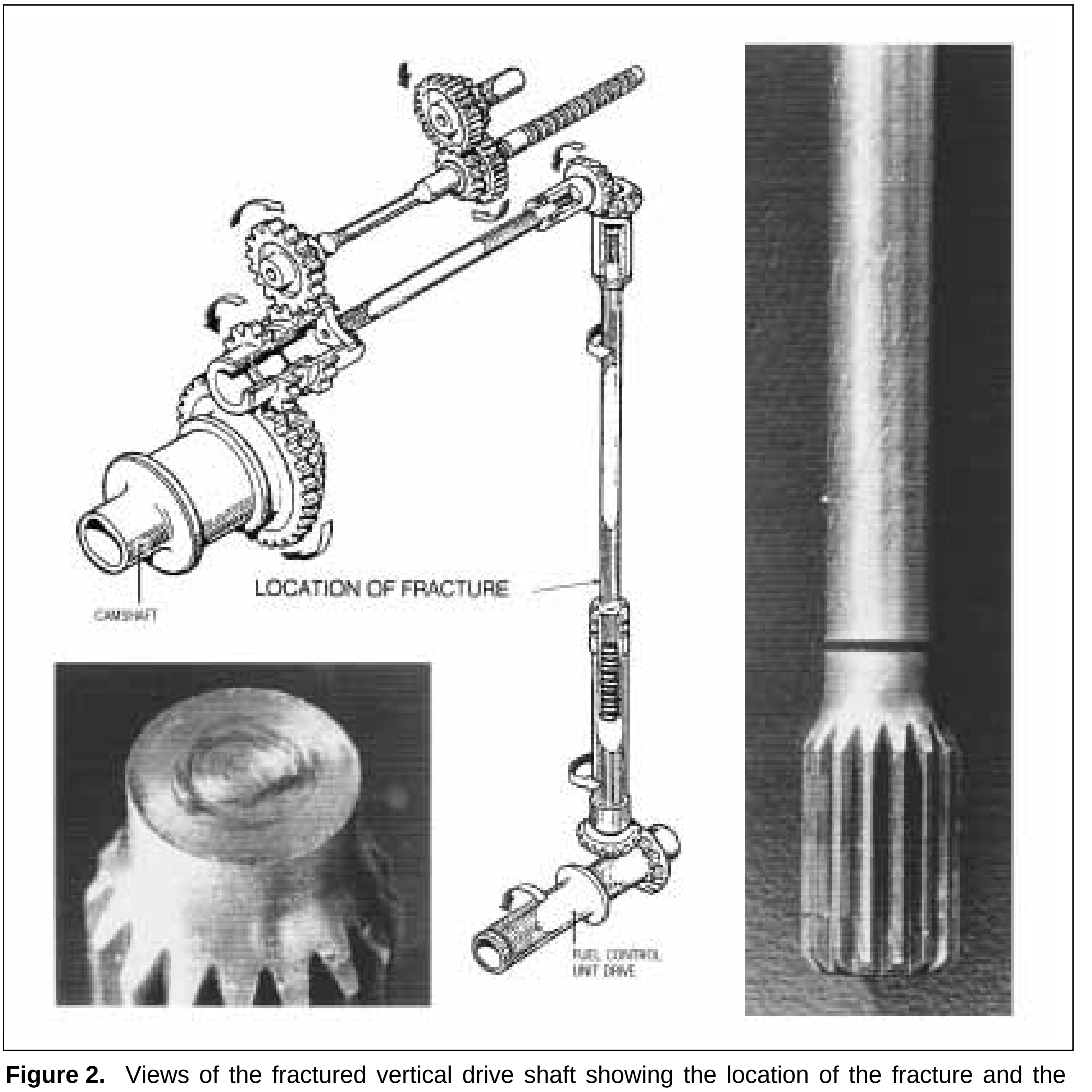

Figure 2. Views of the fractured vertical drive shaft showing the location of the fracture and the fracture surface. The schematic shows the relationship of the drive shaft with the engine camshaft and fuel control unit drive.

1.12.10.3 Failed fuel control unit fuel pump

Examination of the right fuel control unit revealed that the driving spindle of the integral geared fuel pump had seized in its surrounding bush. The seizure resulted in torsional overload of the vertical drive shaft causing it to fracture. The vertical drive shaft, being only 7 mm in diameter, was the weakest component in the fuel control unit drive.

The pump drive spindle is fitted into a flanged bronze bush in the fuel pump assembly. The spindle and bush thrust faces form a thrust bearing. Metallographic examination of the bush revealed the material to be typical of leaded bronze. Heat generated between the thrust faces caused lead to melt and exude from the bronze alloy bush. The bush and spindle ultimately seized.

The moving components in the pump are lubricated by fuel. Two radial V-grooves are machined into the bush thrust face to introduce fuel between the thrust faces. During normal operation a small quantity of fuel flows through holes in the spindle, and through the grooves in the bush, into the clearance between the spindle and bush thrust faces, providing lubrication. The degree of wear between the thrust faces had resulted in the V-grooves of the bush virtually disappearing. A trace of the grooves was revealed only under high magnification.

Modifications Nos. 2239 and 2185 had been carried out. These were introduced to improve lubrication of the fuel pump. Included was a new spindle incorporating lubricating holes and grooves.

Both the bush and spindle thrust faces contained deep circumferential wear marks. Deposits of iron oxides were present on the spindle face. All fuel passages were free of any blockage.

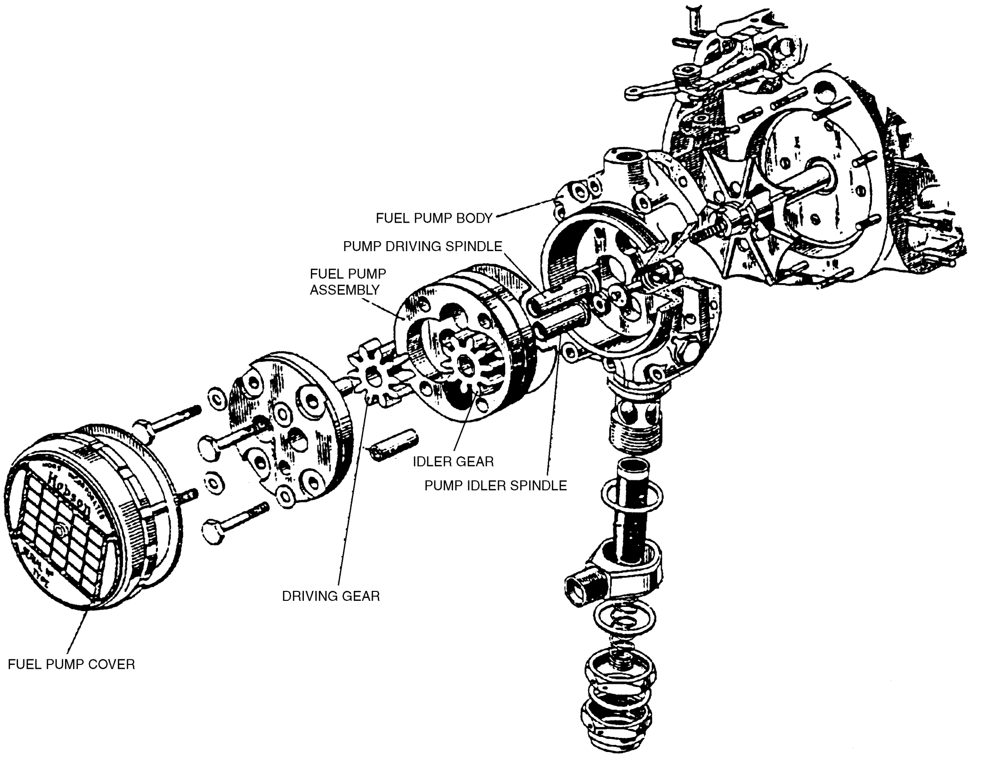

Figure 3. Schematic showing an exploded view of the fuel pump contained within the fuel control unit.

For comparison, the left engine fuel control unit was also disassembled and inspected. The bush and spindle thrust faces from the left engine fuel control unit fuel pump were examined. The bush face contained clearly defined and adequately deep V-grooves.

Throughout the life of the fuel control unit type, the spindles had been subject to a number of modifications designed to improve lubrication and thus to prevent dry running and seizure. Markings on the pump cover of the right engine fuel control unit indicated that all required modifications had been implemented.

The design of the fuel control unit utilises a fuel pump drive shaft comprising two sections. The drive spindle contained within the fuel pump is coupled to a longer shaft that runs through the body of the fuel control unit. Coupling is achieved by a tongue on the drive spindle engaging a corresponding slot in the drive shaft and a spring interposed between the two shafts to take up any end play. The action of the spring creates an axial thrust load on the flange of the bush installed in the end plate of the fuel pump housing. The flange of the bush acts as a thrust washer bearing surface.

Measurements of the spindle, bush, pump assembly and spring revealed that all components were within manufacturer-specified tolerance limits, as was the spring constant. However, the springs from each fuel control unit fuel pump showed differences in appearance. The spring from the right engine fuel control unit was right wound and looked as if it was relatively new. The spring from the left engine fuel control unit was left wound (a right wound spring is specified) and dull in appearance as if it had been in service for a long time. The spring could be further compressed and was assessed as having been correctly installed.

The fuel metering impeller from the right engine fuel control unit had been assembled differently to that from the left engine fuel control unit. The effect of this was that the shaft of the right engine fuel control unit was 0.2 mm longer than that from the left. This would have increased the compression of the end play spring, adding about 3 oz (85 g) to the load between the spindle and bush thrust faces.

The examination also revealed some minor circumferential gear-to-fuel pump assembly contact marks. Apart from the seized spindle, no other abnormality likely to interfere with the pump's operation was found.

1.12.10.3.1 Bush material

The metallographic examination did not reveal any evidence that the leaded bronze material was deficient or unsuitable for the application.

1.12.10.3.2 Contact load

The performance of a thrust washer depends upon the magnitude of the axial load, the presence of any surface roughness or undulations, and the effectiveness of surface lubrication.

The small additional load of 3 oz (85 g), mentioned above, is not considered to have had any significant effect on the bush face wear.

1.12.10.3.3 Fuel as a lubricant

The thrust faces are lubricated by fuel which is a poor lubricant. Lubrication of the thrust faces determines their operational life and some hydrodynamic lubrication, in which metal-to-metal contact is prevented by a thin film of lubricant, is vital. Operational speed and lubricant viscosity are critical. A minimum speed, depending on lubricant viscosity, must be exceeded to create hydrodynamic conditions. For avgas, hydrodynamic lubrication is developed only at high sliding speeds. Examination of the drive spindle and bush from the left engine fuel control unit showed that wear does not normally occur on the thrust surfaces during engine operation.

The engine type data sheet requires 100/130 grade avgas with a maximum lead content of 5.5 mL Tetraethyl Lead (TEL)/imp gal (1.21 mL/L). TEL is added to improve the fuel's anti-detonation characteristics and for engine top end lubrication.

At some time a fuel of a different grade, such as 100 LL, might have been used. The difference between the fuels is the TEL content, 100 LL containing about 2 mL/imp gal (0.44 mL/L). This would not have affected the basic lubricity of the fuel.

Assuming that all fuel tanks had been fuelled from the same sources, any significant reduction in fuel lubricity should have been demonstrated equally in both FCUs.

1.12.10.3.4 Wear conditions

The metallurgical inspection concluded that the excessive wear on the right fuel control unit bush was consistent with sliding, abrasive wear. However, no evidence of abrasive particles was found and the fuel filter screen was clean.

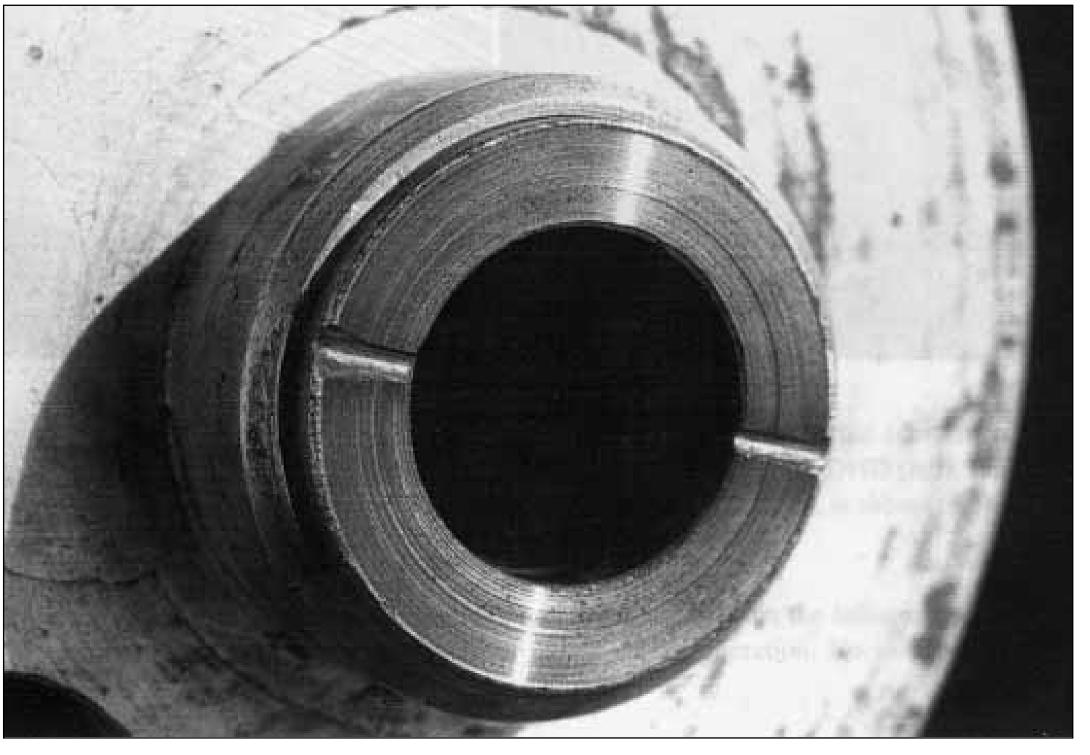

Figure 4. The thrust face of the bush fitted to the FCU fuel pump, left-hand engine.

Hydrodynamic lubrication requires the contact surfaces to be as smooth as possible. Any asperity would penetrate the lubricant film, resulting in metal-to-metal contact and surface wear. As the spindle can move radially within the limits of the spindle-to-bush tolerances, the metal-to-metal contact and wear area would increase.

The marks on the thrust faces from the right engine fuel control unit appeared to have been the result of dry metal-to-metal contact. The thrust faces from the left engine fuel control unit showed some circumferential wear marks consistent with abrasive particles having been present between the surfaces at some time. These particles were probably introduced in the fuel. The contractor who had previously repaired the fuel control unit reported that it is normal for those received for repair to have wear marks on the thrust faces.

Abrasive particles similar to those that produced the marks in the left engine fuel control

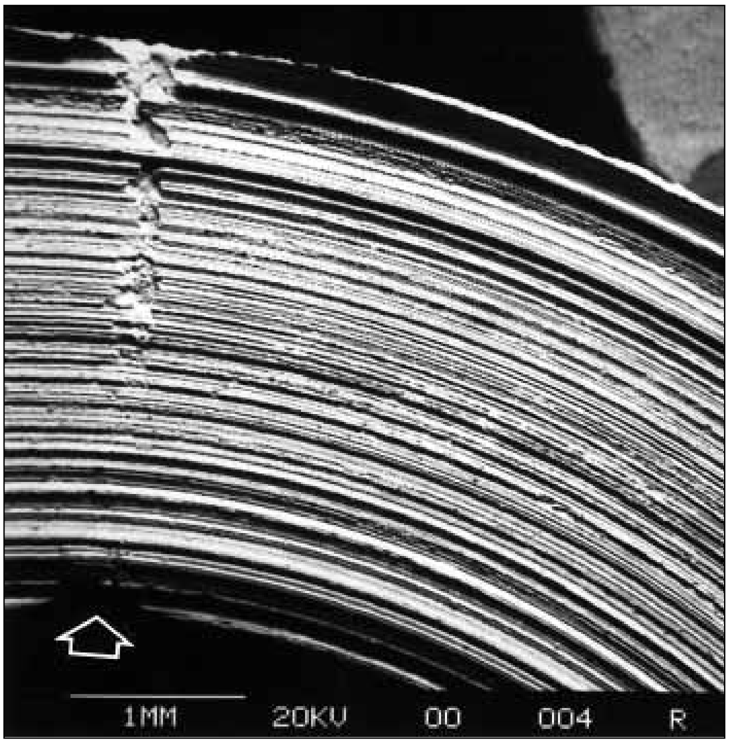

Figure 5. Scanning electron micrograph showing the remnant of the radial lubrication groove on the thrust surface of the fuel pump bush, right-hand engine.

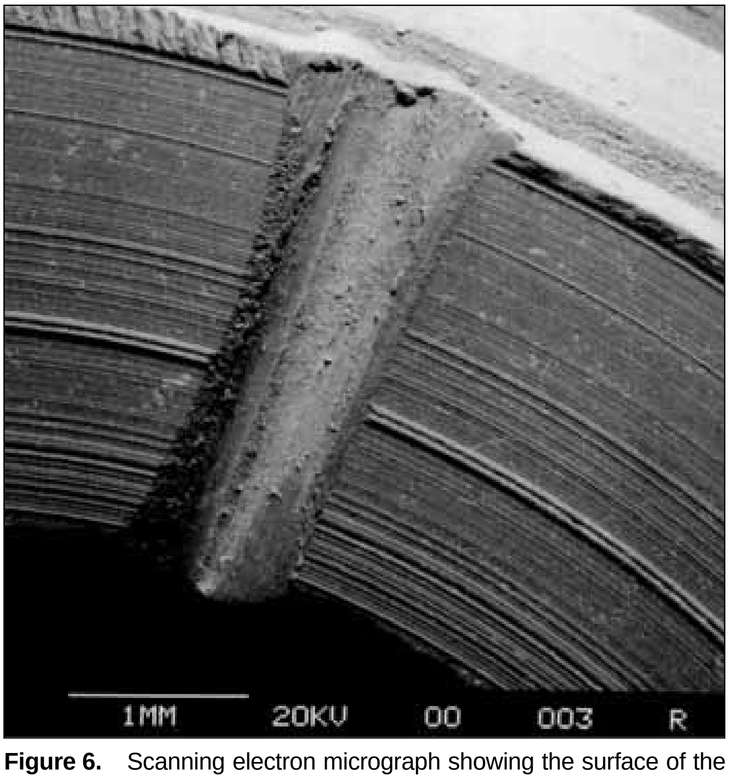

Figure 6. Scanning electron micrograph showing the surface of the bush from the left-hand engine

unit were probably present simultaneously in the right engine fuel control unit. However, the degree of damage to the thrust faces in the right engine fuel control unit would have required a significant quantity of particles over a prolonged period of time. This is unlikely as not only were all fuel filter system screens clear but the investigation did not reveal any history of fuel system contamination. Further, there was no evidence of abnormal pump component wear that was likely to produce such particles.

1.12.10.3.5 Operating conditions

During hydrodynamic lubrication the lubricant film thickness depends upon the lubricant viscosity, the bearing load and the sliding velocity. The bearing load is determined by the physical component dimensions which were not significantly different for the two fuel control unit pumps. The same fuel was used in both engines. The only variable remaining is engine RPM.

During normal cruise, engine RPM differences would be small. During aircraft ground operations, both engines would have run at low RPM. Maintenance test running might have involved different operating conditions for each engine but there was no evidence that the right engine had operated for extended periods at low RPM.

1.12.10.3.6 Fuel control unit maintenance

Maintenance records for the right engine fuel control unit showed that the unit was last overhauled by an overseas maintenance contractor and was returned to service on 18 May 1981. At that time the bearing housing assembly, drive spindle and spring were replaced. However, there is no record of the bush having been replaced.

The bush is an interference fit in the bearing housing. The bush face is then machined to the combined housing and bush assembly thickness of 0.600 ± 0.005 inches. Two grooves to a depth of 0.01–0.02 inches are then machined into the bush face. The combined housing and bush assembly thicknesses were 0.589 inches for the right engine fuel control unit fuel pump and 0.601 inches for the left engine fuel control unit fuel pump.

Advice from the maintenance contractor was that, provided the combined housing and bush assembly thickness was within specification and there was no evidence of excessive wear, overheating, or picking between the thrust surfaces, the bush could be released for a further 1,500 hours of service.

1.12.10.3.7 Total time in service

At the time of the accident the right engine fuel control unit had 753 hours to run to the next 1,500-hour overhaul. It had been last inspected 92.36 hours before the accident.

All pre-1989 records were taken from the original UK logbooks. The operator’s logbooks dated from 1989. In the latter, the fuel control unit installation is recorded in relation to engine or aircraft hours—no specific fuel control unit total time in service is recorded.

When the right engine fuel control unit was released for service after overhaul in 1981, all dimensions should have been within specification. The grooves in the bush face should have been at least 0.01 inches in depth. Unfortunately, no record of the housing and bush assembly thickness at that time was available. Similarly, no corresponding records for the left engine fuel control unit were available. Hence no comparison of bush wear against total time in service could be made.

Anecdotal evidence indicated that, typically, the bushes last for two overhaul periods.

The extent of the wear on the tongues of the fuel control unit drive spindle for each engine indicated that the spindles had been subjected to similar loading conditions. Although the right engine fuel control unit spindle tongues showed slightly more wear, the difference was insufficient to explain the wear on the bush thrust face.

1.12.11 Landing gear retraction

1.12.11.1 Landing gear selection

The pilot's first attempt to raise the landing gear was unsuccessful, but the gear was successfully retracted after deselection and reselection. According to the operator's DH104 operating instructions, failure of the gear to retract because of baulking of the lever movement is a known problem following rapid selection of the up-position instead of a positive and unhurried selection.

1.12.11.2 Baulking of the landing gear retraction lever

The manufacturer's maintenance and repair manual MRM104 indicates that a known cause of the failure of the landing gear to retract is baulking of the selector lever during up-selection of the landing gear because of maladjustment of the lever lock system. A pedestal switch within the selector lever lock system limits the period in use of the lever lock solenoid to prevent the solenoid overheating and burning out. The pedestal switch is adjusted to ensure that the solenoid interrupts the selector movement only when the aircraft landing gear is on the ground. If the switch or solenoid plunger are out of adjustment and during flight the movement of the selector lever is rapid, the solenoid plunger may not fully withdraw and the plunger may be trapped by the lever stop block, thereby inhibiting landing gear retraction.

1.12.11.3 Adjusting and checking the lever lock system

A detailed instruction for adjusting the pedestal switch and solenoid plunger is contained within MRM104. There is also a statement within that instruction to the effect that if the system is not adjusted in accordance with that instruction, the solenoid plunger may not fully withdraw, as described above.

MRM104 also calls up a check for: 'Correct *functioning* of selector lever lock plunger assembly [italics ours]' during landing gear retraction tests every second check 2. There is no requirement to check for the correct *adjustment* of the plunger assembly. Adjustment is only required when necessary, such as when a problem is detected during *functional* checks.

The operator was checking the *functioning* of the landing gear solenoid at each landing gear retraction check during maintenance prior to each maintenance release issue every 100 hours or 12 months. The functioning of the solenoid was last checked five months prior to the accident.

1.12.11.4 Regulatory requirements

The operator had elected, in accordance with the relevant regulations, to maintain the aircraft to maintenance schedules contained within the CAR Schedule 5. These schedules require the landing gear system to be functionally checked but do not require the operator to specifically check the adjustment of the selector lever lock system.

The methods by which operators select a system of maintenance for their aircraft is defined in the CARs. The operator of this aircraft elected to use, as a part of his system of maintenance, the CAA Maintenance Schedule. He advised the CAA of that election by submitting a document known as Log Book Statement Part 1. This document was placed on the aircraft file without any need for the CAA to be satisfied that the maintenance schedules and system of maintenance were appropriate for that class and type of aircraft. The CAA did not have a method of formally advising operators what criteria should be used for assessing the acceptability or otherwise of a maintenance system.

1.13 Medical information

No evidence was found to indicate that the pilot or any of the passengers suffered from any medical condition which might have contributed to the accident.

1.14 Fire

There was no pre-impact or post-impact fire.

1.15 Survival aspects

1.15.1 Pre-flight safety briefing

The pilot had given the passengers a pre-flight safety briefing. Seven of the passengers later stated that they remembered the briefing; one did not remember the briefing but remembered reading a safety briefing card; and one, who was rendered unconscious during the impact sequence, could not recall a briefing or reading a card.

1.15.2 Evacuation

The pilot and four passengers were able to escape from the wreckage without assistance. The pilot helped one passenger to evacuate, while three other passengers were assisted by a bystander, an ambulance officer or another passenger.

The passenger occupying the right cockpit seat was ejected during the impact sequence. He remained strapped into the seat which had separated from the airframe.

1.15.3 Location of accident site

The aircraft was not equipped with an emergency locator beacon. An air traffic controller in the Essendon Tower noted the vicinity of the accident site and relayed this information to the crew of a police helicopter which quickly located the accident site. Emergency services arrived at the site several minutes later.

1.16 Tests and research

1.16.1 Theoretical aircraft performance

The investigation included an analysis of the theoretical performance capability of the aircraft. DH-104 Series 5 performance documentation was studied to assess the aircraft's theoretical capability to:
(a) reject the takeoff and stop within the remaining runway available, or
(b) maintain height or climb safely, OEI following an engine failure during takeoff.

Data

1. Aircraft gross weight at takeoff, 3,689 kg. (Maximum permissible gross weight at takeoff is 3,992 kg.)
2. Takeoff from runway 17, which is sealed, and for which:
   TORA 1,585 m
   TODA 1,645 m
   ASDA 1,585 m
   Slope 0.9% down to south
3. Weather conditions included a light north-westerly wind, temperature 19°C and QNH 1008.

General assumptions for the study

1. Take-off power of 7.5 lb/in^2 boost pressure at 3,000 RPM was set on both engines at the beginning of the take-off roll.

2. Wing flaps were selected to 20° and remained at that setting throughout the sequence.
3. A tailwind component of 3 kts was applied from beginning of the take-off roll until impact.
4. There was a zero instrument error correction. (This must not be confused with position error correction of IAS which was applied.)
5. The aircraft used 60 m of runway length to line up before takeoff.
6. The aircraft attained a height of 50 ft above the runway and an IAS of 84 kts, with both engines operating, prior to the right engine failure.

1.16.1.1 Reject the takeoff (from 50 ft AGL landing back on runway)

Specific assumption

Following the engine failure the pilot took no action for two seconds, then retarded the throttle of the operating engine and allowed IAS to decrease from a take-off safety speed of 84 kts to the nominated landing approach speed, selecting 60° of flap when appropriate, for the aircraft gross weight. From here a landing straight ahead was carried out in accordance with the provisions of the landing chart.

Conclusions

The aircraft would have required 754 m to line up, take off and attain a height of 50 ft above the runway and a take-off safety speed of 84 kts. Following the engine failure and the two-second reaction time, the aircraft would have travelled a further 175 m in the time required to reduce IAS from 84 kts to the nominated approach IAS of 80.5 kts. From this point a further 643 m would have been required to land and stop. Distances quoted are actual distances excluding any safety factors.

The total distance required is 1,572 m compared to an ASDA of 1,585 m. Given the assumptions, the aircraft would theoretically have been able to return to the runway and stop. However, if the pilot had been prepared to overrun the end of the runway, there was an overrun distance of 61 m of level grass to the airport fence available.

Notes

The conclusions of a theoretical model are only as useful as the assumptions upon which the model is based.

The assumption of a two-second reaction time for a pilot who is not expecting a failure (even though he has self-briefed for this contingency) might be unrealistically short. A 1991 research paper titled ‘Response times to simulated aircraft emergencies’ by R. G. Green, J. W. Chappelow and R. J. Skinner of the Royal Air Force Institute of Aviation Medicine reported that, in a series of flight simulator tests in which the subjects were professional pilots in current flying practice on the simulated type, response times to emergencies averaged 3.4 seconds (range 1.8–7.0 seconds) in one series and averaged 3.1 seconds (range 1.5–5.5 seconds) in the other. Any delay in lowering flap to 60° would increase the air distance to touchdown.

The performance data upon which charts and tables are based are usually derived from a manufacturer’s test flight program using both new aircraft and test pilots who are expert on the aircraft type and who may be expecting the failure. The approved landing charts included a factor to compensate for a variety of variables including pilot handling techniques.

In the light of the above, the theoretical results should be used as a guide only. A safe return to the runway followed by a stop within the confines of the airport was possible, but not necessarily before the end of the runway.

1.16.1.2 Continued safe flight OEI

The following calculations were carried out using the performance information available. Although not specifically identified as such, the material appeared to be that produced for, and approved by, the UK Air Registration Board at the time that initial type approval was sought.

Basic climb performance information was available for the aircraft in three configurations:

(a) one engine inoperative, operating engine at maximum continuous power, flaps and landing gear retracted;

(b) both engines operating at maximum continuous power, flaps and landing gear retracted; and

(c) both engines at take-off power, flaps at 20°, landing gear extended, IAS of 84 kts.

Conclusions

With the right engine inoperative and its propeller feathered, the left engine at take-off power, 20° of flap, 84 kts IAS and landing gear extended, the information predicted a performance between level flight and a descent of 20 ft/min.

From the point of engine failure to ground contact, VH-DHD travelled 1,700 m. Given an IAS of 66 or 67 kts (the IAS reported by the pilot), the time from engine failure to impact was 41 seconds. As the impact point was approximately 80 ft below the height at which the engine was thought to have failed, the information predicted that the mean rate of descent over the flight path was approximately 300 ft/min. This rate of descent is consistent with that of an aircraft with a failed engine and the propeller of the failed engine windmilling.

Although data were not available, the analysis included an estimation that, with landing gear retracted, the aircraft would be able to maintain a rate of climb of 80–100 ft/min. Although the pilot was unable to retract the landing gear at his first attempt and had to reselect the gear up before it was successfully retracted, the landing gear was likely to have been fully retracted for the last 20 seconds of the flight. Flight tests in the UK recorded retraction times of 5–6 seconds.

With a failed engine, windmilling propeller, flaps at 20°, and landing gear extended, airspeed would decay. Under ideal conditions, allowing for the two-second reaction time, landing gear retraction and propeller feathering should be completed within 6–7 seconds. Landing gear drag would remain almost constant during the retraction cycle as frontal area varies little, but propeller drag would reduce as the blades moved toward feather. Under these conditions, a speed decay of some 4 kts is likely, requiring an altitude reduction of about 30 ft to regain 84 kts IAS. However, if the speed decay took place over a period of 8–10 seconds, an altitude loss in excess of 60 ft to regain 84 kts IAS could be expected.

The IAS for best climb performance, in the take-off configuration, OEI, is not known, but the nominated take-off safety speed of 84 KIAS is probably representative. Consequently, any reduction in IAS below 84 kts would progressively reduce climb capability as speed reduced. Although the precise relationship between reduction in rate of climb and reduction in IAS is unknown, an IAS reduction to 74 kts could nullify the advantage gained by retracting the landing gear.

Bank angle also significantly affects both rate of climb and the minimum speed at which directional control can be maintained. An angle of bank toward the inoperative engine degrades aircraft OEI performance.

Notes

The above analysis was carried out using the best available data. While it cannot be guaranteed to be precise, the analysis can be regarded as a useful guide to the performance of a DH-104 aircraft, OEI, during this critical phase of flight.

Australian airworthiness performance certification requirements (in common with those of other countries), are applicable to aircraft such as the DH-104 and do not consider the effects of engine failure during takeoff and climb. OEI climb performance cannot be guaranteed, nor are performance guarantees required for normal-category certification until landing gear and flaps have been retracted and the aircraft is flying at or above the optimum climb speed. During takeoff there is an accepted risk period within which, should an engine failure occur, an accident might result. This period will vary according to aircraft type, gross weight and meteorological conditions, but can be up to 25 seconds.

The following extract from *Aviation Safety Digest* 105 (1979) illustrates the philosophy adopted in the certification of this category of aircraft:

> The light aeroplane design standards have been evolved over the years to the point where modern aircraft have a safety record which, from an engineering point of view at least, is very good indeed; but these standards do not, nor are they meant to, provide as high a level of safety as the transport category rules.
>
> Why not design a multi-engine light aeroplane to the transport rules and take full advantage of the extra safety? It can be done and has been done, but like everything else it must be paid for. The price is high, not only in terms of initial purchase and subsequent maintenance costs but also in relation to operating economics. To realise the engine-failed performance of the large aeroplane, the average light aircraft would be so payload-limited it would be virtually unusable. If light aircraft are to be operated in a realistic manner, a level of safety lower than that present in large aircraft must be tolerated.

VH-DHD encountered its engine failure at, arguably, the most critical phase of the takeoff under the prevailing conditions. The aircraft could possibly have been brought to rest within the airport boundaries, even though this may have required a runway overrun.

1.16.2 Actual aircraft performance

The investigation included test flying to assist in assessing the performance OEI of the DH-104. The tests were conducted by the National Test Pilot School, Mojave, California, USA, using an instrumented DH-104 Series 6A aircraft, Serial Number 04401, Registration N11XW.

Series 5 and 6 aircraft were built for operations under British registry and differ only in cabin layout and fuel tank capacity. The differences between N11XW and VH-DHD were cabin layout, cockpit instruments and switch labels. These differences were not considered to be significant for the test objectives.

1.16.2.1 Test objectives

The broad objectives of the test flight program were to determine:

(a) single engine rate of climb with:
   - (i) right propeller windmilling, landing gear retracted, 20° of flap;
   - (ii) right propeller feathered, landing gear retracted, 20° of flap;
   - (iii) right propeller feathered, landing gear retracted, flaps up;

(b) distance to complete a rejected takeoff following an engine failure at 50 ft AGL;

(c) time history of airspeed following an engine failure while maintaining constant altitude, without raising the flaps or feathering the propeller of the failed engine;

(d) landing gear and flap actuation times with both engines operating and OEI;

(e) time for completion of propeller feathering from fuel shutoff, using normal response times; and

(f) engine indications (RPM and manifold pressures) with OEI, propeller windmilling, full throttle and 84 KIAS.

1.16.2.2 Test aircraft data

N11XW had accrued 5,981 total airframe hours. Operating hours since last overhaul were:
(a) left engine – 669.5 hours
(b) right engine – 1,584.8 hours
(c) left propeller – 236.4 hours
(d) right propeller – 236.4 hours.

N11XW was instrumented for the purposes of flight test training. The calibrated instruments applicable to this test program included airspeed indicator and altimeter calibrations plus airspeed and altimeter position error corrections. Aircraft performance was recorded by the flight test engineer at the test station 'Console 2' using a specially mounted calibrated 'wingtip' pitot-static system.

1.16.2.3 Test flight program

1.16.2.3.1 Program

The test flight program consisted of five flights totalling 6.6 flying hours. Test sequences included landing gear cycle time measurement, OEI climb data collection and position error correction tests.

1.16.2.3.2 Method

A series of constant airspeed climbs and descents were flown from a reference altitude of 3,000 ft PA. Each climb or descent was flown for a minimum of one minute after the aircraft was judged to be in a stable condition. Climbs were performed with wings level and constant heading.

One climb/descent was carried out with zero slideslip, requiring 3° of bank toward the operating engine.

Headings selected were at 90° to the surface wind. There was no obvious wind gradient effect.

Maximum continuous power, 2,700 RPM and +6.0 lb/in^2 boost, was used for climbs. Boost was maintained at +6.0 lb/in^2 up to the maximum throttle position which occurred at about 3,000 ft AMSL. The test data were corrected to higher power settings by use of technical data held by the National Test Pilot School.

N11XW weight and balance was maintained as close as practicable to that of VH-DHD, gross weight 8,113 lb (3,680 kg) and centre of gravity 2.5 inches (62.4 mm) aft of datum. Gross weight of N11XW varied from a minimum of 8,069 lb (3,660 kg) to a maximum of 8,204 lb (3,721 kg). Centre of gravity of N11XW was within 0.5 inches (12.7 mm) of that of VH-DHD. All climb results were corrected to a standard weight.

1.16.2.3.3 Limitations

Limitations that directly affected the test program were as follows:
(a) Maximum take-off power was limited to 5 minutes at 3,000 RPM at a boost pressure of +7.5 lb/in^2. (This was the power setting at the time of the VH-DHD engine failure.)
(b) Maximum take-off weight was 8,800 lb (3,990 kg).

In addition, unique limitations for N11XW were imposed due to fuel differences from the fuel specified in the operator's flight manual. The flight manual specifies 100/130 grade fuel whereas the fuel used in N11XW was 100LL. This is an acceptable substitute, but due to concern about possible detonation, the National Test Pilot School limited the take-off power to 3,000 RPM and a boost pressure of +6.0 lb/in^2. All data were gathered under these limitations and corrected to the higher power settings based on engine data held by the School.

The pilot's manual includes a warning concerning flap position:

> Warning: With flaps extended, climb performance on one engine is limited. It is essential, therefore, when early engine failure has occurred (or is simulated for practice or crew training purposes) that the flaps are raised immediately after alighting gear retraction.

1.16.2.3.4 Specific objectives, results and discussion

1.16.2.3.4.1 OEI climb performance

The specific test objectives were to determine:

(a) OEI rate of climb/descent with the right propeller feathered, landing gear retracted and flaps up;
(b) degradation on OEI climb performance due to leaving flaps at 20°;
(c) degradation in climb performance due to leaving flaps at 20° and not feathering the propeller of the failed engine;
(d) variation in OEI climb performance over a range of airspeeds from 84 KIAS (take-off safety speed) to 95 KIAS (flight manual recommended single engine climb speed); and
(e) effect of sideslip on climb performance.

The results, corrected for the condition of VH-DHD at impact (right engine failed, landing gear retracted, 200 ft PA, OAT 19°C, gross weight 8,113 lb (3,680 kg))were:

Flaps	*Right propeller*	*RPM*	*Left engine powerboost*	*EAS*	*Rate of climb*
Up	Feathered	3,000	+7.5	93 kts	+230 ft/min
20°	Feathered	3,000	+7.5	91 kts	+60 ft/min
20°	Windmilling	3,000	+7.5	89 kts	-150 ft/min

The rate of climb decreased with flap extended and a windmilling propeller. The speed for maximum rate of climb decreased with increased drag. Feathering the failed engine and raising the flaps are essential to maximise rate of climb.

Most of the wings-level climbs resulted in approximately 6° of sideslip. The climb at 95 KEAS, with zero sideslip, resulted in approximately 5° of bank toward the left engine. Climb performance (corrected to 8,800 lb (3,990 kg), maximum power on the left engine, right engine shut down and propeller feathered, landing gear retracted, flaps up, standard day) increased from 123 ft/min to 153 ft/min. Although the increase was only 30 ft/min, it represented almost 25% improvement.

The data were also corrected to sea level in ISA conditions, at a gross weight of 8,800 lb (3,990 kg). The data indicated that N11XW could climb at 160 ft/min under these conditions, compared to the pilot's manual figure of 220 ft/min. This 27% difference was probably attributable to the age of the aircraft and engines.

1.16.2.3.4.2 Rejected take-off distance

This test was not carried out successfully. On all test days the wind exceeded the pre-planned maximum of 10 kts for valid data. The combination of pilot technique variations and local engine limitations made achievement of accurate results unlikely.

1.16.2.3.4.3 Deceleration in level flight

The specific objective of this test was to determine a time history of speed following engine failure while maintaining constant altitude, with the right engine inoperative, propeller windmilling, 20° of flap and landing gear retracted.

The test was carried out at 3,000 ft PA with a gross weight of 8,057 lb (3,655 kg). Initially, both engines were set to 3,000 RPM and +6.0 lb/in^2 boost. The aircraft was established in a climb below 3,000 ft, 20° of flap was set, and the landing gear was retracted. At 3,000 ft, fuel to the right engine was selected off and the right propeller allowed to windmill. The pilot then maintained a constant altitude with wings level and rudder applied until full rudder deflection was reached. The aircraft speed was monitored.

The test resulted in a speed reduction from 84 KEAS (81 KIAS) to 76 KEAS (73 KIAS) in 27 seconds. Due to a lack of knowledge of the dynamics of the engine power during wind down to steady windmilling plus the small differences in power settings, weights and OAT between N11XW and VH-DHD, the data were not corrected to VH-DHD flight conditions.

1.16.2.3.4.4 Landing gear and flap actuation times

This test was carried out with both engines operating and OEI with a minimum pneumatic pressure of 400 lb/in^2 before each test point.

The aircraft was flown at 90 KIAS, ± 5 kts, the variation assumed to be insignificant compared to differences between aircraft or variations in pneumatic pressure. Landing gear and flaps were raised and lowered and the time between activation and indication of complete movement was measured. The tests were carried out with both engines operating and repeated with OEI and its propeller feathered.

Landing gear actuation times were consistently 4–5 seconds for extension and 3–4 seconds for retraction as long as pneumatic pressure was above 400 lb/in^2. Flap actuation times were consistently 9–10 seconds for raising and 6–7 seconds for lowering. Times were unaffected by the number of operating engines.

Note: One anomaly occurred during the first attempt to measure landing gear retraction time. When the pilot selected landing gear up on the engineer’s call, the gear did not fully retract. External observation confirmed the landing gear to be partially retracted. The pilot probably moved the handle more quickly than normal in an attempt to obtain accurate test data.

1.16.2.3.4.5 Propeller feathering times

The aircraft was cruising at 90 KIAS at 8,500 ft with both engines operating at 2,700 RPM. The co-pilot shut off the fuel. The pilot feathered the propeller, by retarding the propeller lever to feather and pressing the feathering button, three seconds (assumed normal reaction time) after recognising the failure. Time from fuel shutoff to the propeller fully feathered was 12 seconds, including the three-second reaction time.

1.16.2.3.4.6 Engine instrument indications

With the right engine failed and the propeller windmilling, RPM was 2,200 and the boost was +2.0 lb/in^2 at 90 KIAS. At the lowest airspeeds tested, RPM decreased to 2,100 and boost decreased to +1.0 lb/in^2. When the throttle was retarded, RPM remained constant but boost decreased, similar to an engine operating on partial power. Other indications of the engine failure were cylinder head temperature, which decreased to off-scale, and the substantial left rudder force required to maintain the slip indicator central.

1.16.2.3.4.7 Position error corrections

Although not included as an initial objective of the test series, anomalies between airspeeds observed at Console 2 (wingtip pitot-static system) and airspeeds observed by the pilot resulted in an additional test objective. This was to determine the airspeed position error corrections for the pilot's pitot-static system and the wingtip pitot-static system down to speeds approaching the stall.

For the test, the aircraft was fitted with a trailing bomb. At 5,000 ft PA the aircraft was stabilised at speed increments of 10 kts from the maximum speed attainable with power 2,100 RPM, +1 lb/in^2 boost, down to stall buffet. At each stabilised point, airspeed, altitude, temperature plus static and dynamic pressure differences between the test pitot-static system and the remote ambient conditions, were recorded. The test was carried out in two configurations:

1. landing gear and flaps retracted; and
2. landing gear retracted, 20° of flap.

Data were then reduced in accordance with documented procedures.

At low speeds the position error in the pilot's airspeed indication was 4–8 kts. If a pilot is attempting to fly the aircraft at 91–93 kts, the aircraft flies 4–8 kts faster than desired, resulting in a less than optimal OEI climb performance. At the optimum OEI climb airspeed of 93 KEAS, with flaps up, the position error was 5 kts.

1.16.2.3.5 Conclusions

Raising the flaps and feathering the failed engine are essential to achieve a safe climb following an engine failure during takeoff.

The instrument indications following an engine failure can be misleading. RPM stays high with the engine windmilling at 2,100–2,200 RPM. Boost pressure also remains at 1–2 lb/in^2. The boost pressure varies with throttle position and decreases when the throttle is retarded.

Landing gear and flap actuation times were constant regardless of whether one or two engines were operating. Landing gear extension required 4–5 seconds and retraction required 3–4 seconds. Flap extension to 20° required 6–7 seconds while retraction required 9–10 seconds.

The time required in the test flights to fully feather a propeller following an engine failure was 12 seconds. This included a reaction time of 3 seconds.

Position error at low speeds is 4–8 kts. Optimum OEI climb performance is achieved at 93 KEAS which is 88 KIAS, assuming that the ASI instrument error is zero.

1.16.3 Performance requirements

Civil Aviation Orders (CAO) section 20.7.4, subsection 7, specifies the statutory minimum climb performance requirements for aircraft such as the DH-104 used for private, aerial work or charter operations (i.e. normal category), and states:

> 7.1 – In the take-off configuration with landing gear extended, an aeroplane must have the ability to achieve a climb gradient of 6% at take-off safety speed, without ground effect, and with all engines operating at take-off power.

CAO 20.7.4, subsection 8, specifies en route OEI climb performance requirements. Included is the requirement that an IFR aircraft must be able to maintain altitude or to climb at a 1% gradient, with the critical engine inoperative, at 5,000 ft in ISA, given that:

(a) the operating engine is at maximum continuous power;
(b) the propeller of the inoperative engine is feathered;

(c) landing gear and wing flaps are retracted;
(d) airspeed is that for best climb angle with the critical engine inoperative; and
(e) the aircraft is banking up to 5° toward the operating engine.

Although not specified in the CAO, CAA staff reported that, in practice, the CAA required this capability to be demonstrated at maximum gross weight.

There is no requirement for climb performance in the take-off configuration with OEI. In addition, there is no requirement for operators to recheck the climb capability of their aircraft.

1.16.4 Zero-flap takeoff

The pilot believed that he was not permitted to carry out takeoffs with zero flap as a zero-flap take-off performance chart had not been included in VH-DHD's CAA-approved flight manual. Consequently, all takeoffs were carried out with 20° of flap in accordance with that flight manual.

1.17 Organisational and management information

The chief pilot was the owner, manager and the only full-time pilot of the company. The company employed part-time staff, including a flight attendant, clerical officer and an aircraft maintenance engineer. Other pilots were employed when required.

The operator had been subjected to its most recent annual surveillance by a CAA flying operations inspector on 26 February 1993. This surveillance covered all aspects except a ramp check. In all aspects checked, the operator was found to be satisfactory.

1.18 Additional information

1.18.1 Effects of aircraft ageing

Demonstrations of the capability of each aircraft type to satisfy the certification requirements are usually performed by the manufacturer, under ideal conditions and using new aircraft.

Airframes, engines and propellers exhibit performance deterioration with increasing time in service. Such a deterioration was possibly a factor in the adjusted data indicating that N11XW could climb at 160 ft/min at 3,990 kg gross weight, at sea level in ISA compared to the performance chart figure of 220 ft/min.

VH-DHD had accrued 21,259 airframe hours since new. Although the airframe was in good clean condition, some performance loss compared to N11XW, with the right engines inoperative, cannot be dismissed. The left engine of VH-DHD had 334 hours to run and the left propeller had 370 hours to run to the next overhaul. The corresponding figures were 870 hours and 1,362 hours for N11XW.

1.18.2 Operator certification

The operator held a valid air operator's certificate for charter and aerial work operations, excluding flying training. The certificate was valid for land-based aeroplanes not exceeding 5,700 kg MTOW. It excluded turbine powered aeroplanes, pressurised aeroplanes, aeroplanes with a seating capacity to carry ten adult passengers or more and amateur-built aeroplanes. However, the certificate specified that Aero Commander 690, Cessna 340, DH-104 and GAF Nomad N 24 were included.

The operator also held a valid certificate of approval to carry out maintenance on the airframe and engines of the DH-104 and had been maintaining VH-DHD before the accident.

1.18.2.1 Maintenance history

The CAA's and the operator's maintenance records were examined. There was no record of defects or damage that may have contributed to the accident.

1.18.3 Operator and manufacturer emergency procedures

Part B of the operator's operations manual contained the following emergency checklist primary actions. The procedures specified for engine failure during takeoff were:

Engine failure on takeoff before V_2

Actions

1. Throttles – Closed
2. Flaps – Up
3. Brakes – Apply as required

Engine failure on takeoff at or above V_2

Actions

1. Flight technique – Airborne not below V_2
2. Landing gear – Up
3. Flaps – Up if airspeed above 96 kts, 20° if airspeed below 96 kts
4. Flight technique – Climb to 400 ft at 84 kts if flaps 20°, or 96 kts if flaps retracted
5. Pitch lever – Feather
6. Feather button – Press
7. Flight technique – At 400 ft accelerate to 96 kts
8. Flaps – Up
9. Power – Reduce to rated (+6 lb, 2,700 RPM) when possible.

The operator was using 84 kts as V_2. This speed is also V_{TOSS}. The procedure applies regardless of flap selection; this will affect only the value of V_2.

With the emergency actions listed in this order, the procedure could be interpreted as: 'Maintain V_2, retract the landing gear then attempt to climb the aircraft to 400 ft before feathering the propeller of the failed engine or raising the flaps'.

The manufacturer's *Crew's Notes for the Dove Aircraft Series 5, 5A, and 6, 6A, Second Edition*, which was approved by the UK Air Registration Board and the US Federal Aviation Administration, required the following action if an engine failed:

> Emergency
>
> 4. Engine failure during takeoff:
>
> Action
>
> (1) If engine fails before takeoff safety speed is reached, the other engine should be throttled and the aircraft landed (if airborne) and brought to rest.
>
> (2) If an engine fails after reaching takeoff safety speed, takeoff should be continued on one engine and the following procedure adopted:
>
> (a) Retract the alighting gear THEN IMMEDIATELY retract the flaps (see WARNING below) and feather the dead propeller.
>
> (b) Once clear of obstructions, allow the speed to build up to 105 to 110 m.p.h. (91 to 95 kt.) IAS. At this speed, with an aircraft weight of 8,800 lb., the rate of climb on one engine is approximately 220 ft./min., on takeoff power under I.S.A. sea level conditions.
>
> WARNING: WITH FLAPS EXTENDED, CLIMB PERFORMANCE ON ONE ENGINE

IS LIMITED. IT IS ESSENTIAL, THEREFORE, WHEN EARLY ENGINE FAILURE HAS OCCURRED (OR IS SIMULATED FOR PRACTICE OR CREW TRAINING PURPOSES) THAT THE FLAPS ARE RAISED IMMEDIATELY AFTER ALIGHTING GEAR RETRACTION.

The manufacturer's publication nowhere implies that the aircraft is capable of climbing with a windmilling propeller.

The operator's emergency checklist was based upon that previously used by Trans Australia Airlines (TAA) when it was an operator of the DH-104. TAA had submitted the checklist to the then Department of Civil Aviation for review probably in May 1971. Other procedures from the TAA emergencies section are dated from 1962 to 1974. No records were found to indicate whether TAA's engine failure procedures had ever been approved by the regulators.

The operator of VH-DHD included the TAA checklist as part of its operations manual when the application for an AOC was submitted to the CAA.

The section of the checklist that could have been misinterpreted as requiring the feathering of the propeller of a failed engine to be delayed until 400 ft was not amended when TAA submitted its checklist in 1971. Similarly, it was not amended when the operator's chief pilot applied for endorsement training approval, or when the operator submitted its proposed operations manual with its original application for an AOC.

2. ANALYSIS

2.1 Introduction

The aircraft crashed into houses approximately 1 km south of Essendon Airport following the failure of the right engine. The engine failure was the result of fuel starvation, in turn resulting from a fuel control unit failure. The fuel control unit failure was the result of a internal fuel pump seizure which led to torsional overload failure of the fuel control unit vertical drive shaft.

Once the engine had failed, the pilot had two courses of action available to him. The first was to attempt to land the aircraft on the remaining available runway. The second was to carry out the appropriate drills and attempt to fly the aircraft away on one engine.

This section of the report will analyse why the engine failed, why no attempt was made to land and stop the aircraft on the aerodrome, and why the pilot was unable to fly it away safely on one engine.

To this end, the analysis will consider, among other things, the system defences which may have prevented the accident, and organisational factors which may have contributed to the breaching of these defences.

2.2 Fuel control unit fuel pump failure

Detailed examination of the fuel control unit did not reveal a specific reason for the excessive wear between the thrust faces of the right engine fuel control unit drive spindle and its surrounding bush, wear which resulted ultimately in overheating and seizure.

The principal defence mechanisms protecting the functionality of technical systems are:
1. operation of the systems within specified limitations;
2. scheduled maintenance; and
3. inspections to detect faults as they develop.

There was no evidence of abnormal operation of the right engine fuel control unit. Scheduled maintenance had been carried out by a maintenance facility and the unit was about halfway through its time between overhauls when it failed. It had been inspected 92 hours before the accident. The seizure occurred between the fuel pump spindle and bush, two totally-enclosed components that would have required removal of the fuel pump from the fuel control unit and disassembly of the fuel pump to inspect. Scheduled maintenance does not include removal of the fuel pump from the fuel control unit. Without removal and disassembly of the fuel pump, it is unlikely that an excessively worn bush could be detected.

2.3 Fuel boost pumps

Although both electric fuel boost pumps were serviceable and capable of producing pressure 8–10 lb/in^2 in accordance with specifications, the engines cannot operate on boost pump pressure alone. Thus, a failure of fuel control unit-supplied fuel will result in an engine failure.

The defence in the form of an alternative supply of fuel in the event of a fuel control unit failure was not designed into this system. Such a feature was not required by the airworthiness regulations applicable at the time of design and certification of the aircraft type.

2.4 Post-engine failure actions

The result of the failure of the fuel control unit was a total loss of power from the right engine. Available evidence indicates that this loss of power occurred just as take-off safety speed was attained, somewhere between liftoff and 50 ft above the runway.

2.4.1 Engine failure in twin piston-engine aircraft—indications

Immediately following an engine failure in a twin piston-engine aircraft with wing-mounted engines, the aircraft yaws or 'swings' towards the failed engine and the pilot has to use rudder to regain balanced flight. The severity of the 'swing' depends on the power output of the engines at the time of failure, the rate of power loss, the degree of power loss and the airspeed. The 'swing' is most pronounced when the aircraft is airborne, following a total and rapid failure of one engine, and at low airspeed when both engines have been operating at high power e.g. immediately after takeoff.

The 'swing' is the only totally reliable immediate symptom of an engine failure. Identification of the failed engine is initially guided by the 'dead foot, dead engine' rule after the pilot has applied rudder to regain balanced flight. In a piston-engine aircraft, instrument indications can be misleading.

The National Test Pilot School flight test program revealed that, following an engine failure in a DH-104 with the propeller windmilling, RPM reduced to 2,100–2,200 with the corresponding boost pressure being 1–2 lb/in^2. Boost pressure varied with throttle position, decreasing to off-scale when the throttle was retarded. An engine fitted with a constant speed unit, assuming that the constant speed unit has not been affected by the failure, will remain at relatively high and constant (i.e. governed) RPM if the aircraft maintains a speed that holds windmilling RPM in the governed range.

The National Test Pilot School test revealed that, following the engine failure, the cylinder head temperature instrument indicated an unambiguous reduction in power. However, there is little value in this information, as a pilot faced with an engine failure and a windmilling propeller has far more pressing concerns and is unlikely to be able to absorb information to this degree of detail.

2.4.2 Diagnosis of engine failure

Initially, the pilot diagnosed the loss of power as a partial engine failure and decided to continue with the takeoff. He made this decision, in pressing circumstances, after momentarily observing a MAP gauge indication of 3 lb/in^2 of boost. A factor reinforcing the pilot's perception of a partial failure would have been that 3 lb/in^2 of boost and 2,600 RPM is the operator's recommended power setting for a two-engine power climb.

In the light of his past experience, principally in heavy jet aircraft in which engine instruments are reliable indicators of engine operation, the pilot's initial reaction may have represented a reversion to previous habit patterns.

2.4.3 Option to land back on airfield

Post-accident analysis revealed that, under ideal circumstances, a safe return to the runway followed by a stop within the confines of the airport would have been possible, but not necessarily before the end of the runway. However, at the time of the engine failure, no performance data were available to the pilot to determine the viability of the option to return to the runway, other than pilot judgement at the time.

In the pilot's judgement there was not enough distance to land straight ahead safely within the confines of the airport. Attempting such a manoeuvre might have resulted in the aircraft crashing onto a busy freeway beyond and several feet below the level of the runway. Also, having initially diagnosed the engine failure as a partial power loss, the pilot decided to attempt to fly the aircraft away.

2.4.4 Option to fly away

Based on the assumption that both aircraft were capable of similar performance, extrapolation of results obtained during flight testing at National Test Pilot School indicated that VH-DHD, at its take-off weight, should have been able to achieve a positive rate of climb, provided that the landing gear and flaps were retracted and the propeller of the failed engine was feathered as detailed in the manufacturer's emergency procedures. The testing indicated that with 20° of flap set, a rate of climb of about 60 ft/min was achievable; this increased to about 230 ft/min with 0° of flap set. To achieve this performance the appropriate airspeed would have had to be achieved and maintained.

However, on the accident flight, the engine failure occurred at probably the most critical phase of flight. Prompt completion of all of the engine failure drills would therefore have been necessary to assure the best chance of establishing a positive rate of climb. On this occasion, vital time and aircraft performance were lost after the engine failed, mainly due to the landing gear failing to retract on the initial attempt. The failure of the landing gear to retract was possibly due to trapping of the selector lever solenoid plunger by the lever stop block during rapid movement of the selector lever.

By the time the pilot had deselected and reselected the landing gear and retraction had been achieved, the aircraft's performance had decayed to the extent that the pilot's full attention was then diverted to preserving control of the aircraft. He was thus committed to attempting to land the aircraft as best he could, before control was lost. Consequently, he did not continue with the engine failure emergency drills with the result that the flaps were not retracted and the propeller of the failed engine was not feathered.

2.5 Organisational factors

During the course of the investigation, organisational factors involving the maintenance of the Dove landing gear retraction system and deficiencies in manuals and procedures were identified.

2.5.1 Landing gear maintenance

The manufacturer was aware that an out-of-adjustment lever lock solenoid switch could cause baulking of the landing gear lever during rapid movement of the lever. The operator was also aware that the lever could be baulked and had an operational procedure requiring that the lever be deselected and then reselected in an unhurried manner.

The lever system adjustment procedure had not been called up either by the operator's elected maintenance schedules or in the manufacturer's recommended maintenance schedules even though the correct adjustment of the system is necessary to ensure that there is no hindrance to rapid selection. Periodic checking of the adjustment of the landing gear selector system is therefore a critical practice that was not addressed in the systems of scheduled maintenance.

The CAA is not required to assess the operator's system of maintenance for this class of aircraft and therefore the opportunity to note the non-inclusion of this and possibly other critical practices was missed.

2.5.2 Manuals and procedures

2.5.2.1 Operations manual and checklists

The emergency procedures section of the operator's operations manual contained deficiencies which fall into the category of latent failures.

The first deficiency was the checklist that could have been interpreted as requiring the pilot to delay feathering the propeller of a failed engine until 400 ft AGL. The manufacturer's publication *Crew's Notes for the Dove Aircraft Series 5, 5A and 6, 6A, Second Edition*, specifies the procedures described in para. 1.18.3 and does not imply that the aircraft has any capability to climb with a windmilling propeller. Since the checklist had been proposed by TAA, there had been at least three opportunities for this deficiency to be detected and corrected, but all were missed. Thus the system defence of having the regulatory authority of the time conduct independent checks had failed.

The second deficiency was a contradiction between the manufacturer's information on OEI performance and that contained in the operator's checklist. Information published by the manufacturer included a warning that the OEI performance with flap extended was 'limited' plus the advice that 'it is essential, therefore, when early engine failure has occurred (or is simulated for practice or crew training purposes) that the flaps are raised immediately after alighting gear retraction'. However, the operator's emergency checklist included the check 'Flaps—Up if airspeed above 96 kts, 20° if airspeed below 96 kts'. As the right engine failed at 84 kts, these procedures required the flaps to remain at 20°.

Following the engine failure and the reduction in airspeed, the pilot retained 20° of flap. By the time he reached the checklist point of raising the flaps, the airspeed was very low (probably reducing through 76 kts) and he feared that raising the flaps at this speed would increase the aircraft's rate of descent. Both the operator's operating instructions and the manufacturer's notes required propeller feathering to follow selection, at least, of flaps up. As the airspeed had reduced to well below 84 kts the pilot kept the flap at 20°. He did not reach the subsequent 'Pitch lever—Feather' check before the aircraft crashed.

The organisational factor involved in this deficiency is that the contradiction between the manufacturer's published information on emergency procedures and the procedures in the operator's operations manual and checklists had not been detected and corrected, in line with the manufacturer's published information, during any of the independent checking processes carried out over the years by the regulatory authority of the time.

2.5.2.2 Flap configuration for takeoff

The operator's pre-takeoff checklist included the check 'Flaps—As required'. The take-off and climb checklist included the check 'Flaps—Up at 90 kts'. 'Section E: Ground and Flight Procedures' included the statement 'When V_2 (84 knots flapped, 96 knots unflapped) speed is attained, slight backward pressure on the control column will result in the aircraft becoming airborne'. The next paragraph nominated flaps up at 90 kts.

The emergency procedures section nominated V_2 speeds for zero flap and 20° of flap. 'Section B: Loading' of the operating instructions referred to maximum permissible take-off weight charts for both configurations. It noted that 20° of flap permits a shorter takeoff and that zero flap provides a higher level of safety when operating OEI.

However, the Australian CAA-approved flight manual for VH-DHD did not include a take-off performance chart for zero flap. The organisational factor involved is that, although an Australian CAA-approved zero flap chart was available for the DH-104 aircraft, it had not been included by the operator in its operations manual or the aircraft flight manual.

Moreover, its omission had not been detected by the regulatory authority during its checks of the flight manual.

Had the pilot considered that he was permitted to use zero flap for takeoff, this option would have been beneficial for takeoff from Essendon. Although the take-off run would have been slightly longer than that with 20° of flap, having the flap already retracted at the time of the engine failure would have placed the aircraft in a better configuration for a climb OEI.

3. CONCLUSIONS

3.1 Findings

General

1. The pilot was correctly licensed and qualified to undertake the flight.
2. There was no evidence that the pilot was suffering from any illness or incapacity during the flight.
3. The pilot was adequately rested prior to the flight.
4. The pilot gave the passengers an emergency briefing before the flight.
5. There was adequate fuel in both aircraft fuel tanks to conduct the flight.
6. The fuel in the aircraft fuel tanks was of the correct specification.
7. No pre-existing airframe fault which may have contributed to the accident was found.
8. No defect that may have prevented or restricted normal control of the aircraft was found.
9. The aircraft may have experienced a 3-kt tailwind component during takeoff.
10. All but one of the passenger seats detached from the aircraft in the impact sequence.

Aircraft performance and handling

11. The right engine failed after takeoff, at a point between liftoff and 50 ft AGL, due to fuel starvation. This occurred at a critical phase of flight where airworthiness requirements for the DH-104 performance do not take an engine failure into consideration during takeoff or initial climb.
12. The certificate of airworthiness for VH-DHD specified normal category which included charter operations but excluded regular public transport operations. No guarantees of OEI performance can be given or are required for normal category certification until landing gear and flaps have been retracted, the propeller feathered, and the aircraft has achieved optimum climb speed. The accident flight was a charter flight.
13. During controlled test flights in a DH-104 configured and flown to approximate conditions experienced during the Essendon accident, it was discovered that a DH-104 aircraft, at a gross weight of 3,680 kg with a 20° flap setting and landing gear retracted, descended at about 150 ft/min at 89 KEAS with the left engine at maximum power and the propeller of the failed right engine windmilling. The same aircraft climbed at 60 ft/min at 91 KEAS with the landing gear up, the flaps at 20° and the right propeller feathered. With the flaps up as well, the test aircraft climbed at 230 ft/min at 93 KEAS.
14. The use of 0° flap would have reduced, but not eliminated, the period of time during which the aircraft was vulnerable to a high-risk situation during the takeoff and initial climb.
15. The operator’s Australian CAA-approved flight manual for VH-DHD did not include an approved zero-flap takeoff chart.
16. Based principally upon instrument indications, the pilot initially diagnosed the engine failure as partial.
17. Engine instrument indications could have misled the pilot into assessing the engine failure as partial.
18. In the judgement of the pilot, the risk of an attempted forced landing on the remaining runway was greater than that of attempting to climb on one engine.

19. The landing gear did not retract when the pilot first selected it up but retracted following deselecting and reselecting. Vital time and airspeed were lost by the pilot having to deselect then reselect the landing gear up.
20. By the time the pilot had retracted the landing gear, the aircraft's performance had deteriorated to the extent that he then concentrated on maintaining control of the aircraft to the exclusion of continuing the engine failure emergency checks.
21. The aircraft was unable to maintain height or airspeed with the right propeller windmilling.
22. As the indicated airspeed reduced through V_{MCA}, the pilot reduced the power output of the operating engine to prevent uncontrollable yaw.

Fuel control unit malfunction

23. Extensive wear had occurred on the thrust face of the right engine fuel control unit fuel pump drive spindle and on the thrust face of the bush fitted to the right engine fuel control unit fuel pump endplate. The wear had progressed to the point where the radial lubrication grooves had been worn away. The reason for the excessive wear on the bush was not determined.
24. Without the lubrication grooves in the bush, fuel was less able to hydrodynamically lubricate the thrust surfaces between the spindle and the bush.
25. Metallographic examination of the worn bush confirmed that the bush material was typical of leaded bronze. Frictional heating, generated between the thrust surfaces of the bush and the spindle, caused the low melting point lead to exude from the bush, resulting in seizure of the drive spindle in the bush.
26. Seizure of the drive spindle resulted in a torsional overload failure of the vertical drive shaft to the fuel control unit.
27. The failure of the vertical drive shaft resulted in loss of drive to the right engine fuel control unit fuel pump. Loss of drive to the right engine fuel control unit fuel pump resulted in rapid reduction of fuel pressure. Loss of fuel pressure resulted in the right engine being starved of fuel and totally losing power.
28. The recorded total time in service of the right engine fuel control unit fuel pump was 747 hours, leaving 753 hours to the next overhaul.
29. No evidence of abrasive particles was found in the right engine fuel control unit or in the fuel filters.

Landing gear retraction problem

30. The landing gear retraction problem following the engine failure probably resulted from incorrect rigging tolerances within the selector lever system.
31. The importance of periodically checking the rigging tolerances of the selector lever system was not clearly emphasised in the maintenance data available.
32. An opportunity for detecting the deficiency in the maintenance system was missed because the CAA was not required to assess the system of maintenance for this class of aircraft.

3.2 Significant factors

1. The right engine fuel control unit fuel pump failed causing the engine to fail at a critical phase of flight.
2. Maintenance inspections did not detect the abnormal wear on the thrust face of the right engine fuel control unit fuel pump.

3. The landing gear did not retract on the first attempt and aircraft performance decayed while the pilot resolved this problem.
4. The pilot was probably forced to abandon the emergency procedures to concentrate on maintaining control of the aircraft.
5. The aircraft was unable to maintain altitude and airspeed with the right propeller windmilling and 20° of flap.
6. The investigation identified organisational factors concerning deficiencies in the manuals and procedures available to, and used by, the operator for the operation and maintenance of the accident aircraft.

4. SAFETY ACTIONS

4.1 Interim recommendation

During the course of the investigation BASI made the following interim recommendation (IR). The IR document included a 'Summary of Deficiency' which is not reproduced in this report. The recommendations and the CAA response are quoted in full.

The Bureau classifies responses to IRs as:

CLOSED–ACCEPTED.	The Bureau accepts the response without qualification.
CLOSED–NOT ACCEPTED.	The Bureau regards the response as unsatisfactory but considers that further correspondence is not warranted.
OPEN.	The Bureau regards the response as unsatisfactory and that the safety significance of the IR warrants further correspondence

IR940078 **The Bureau of Air Safety Investigation recommends that the Civil Aviation Authority initiate inspections of all installed or uninstalled Hobson MK 103/B type fuel control unit assemblies to ensure:**

(a) integrity of operation of the fuel pump;

(b) freedom from abnormal wear of the bush flange face;

(c) adequacy of the observed radial lubrication grooves across the bush face; and

(d) compliance with DH Modification 2239 (Hobson Mod. 853) and DH Modification 2185 (Hobson Mod. 833) regardless of component dataplate entries.

CAA response

Whilst this Authority agrees that the FCU from the right engine did exhibit abnormal wear and overheating of the fuel pump driving spindle bush, we do not consider that the cause of the wear and overheating has been adequately determined. A review of the defect reports held by this Authority and the CAA (UK) did not disclose a report related to the failure of the FCU since modifications were introduced in 1954 to address overheating of the FCU Fuel pump spindle bushes. The lack of similar reports indicates that the failure may well have been an isolated occurrence.

Advice received by this Authority indicates compliance with the requirements of the Recommendation cannot be carried out in Australia. The data and test equipment necessary to carry out the inspection, assembly and testing of the FCU is available only overseas, therefore imposing both an operational and financial burden on Australian DH-104 operators.

In consideration of the Authority's view that the cause of the FCU failure has not been adequately determined, the lack of a reported similar failure in forty years of operation and the burden of compliance, the subject report recommendation to initiate inspections of certain Hobson Fuel Control Unit FCU assemblies cannot be supported by the Authority at this time.

This Authority intends to continue investigations into the matter in consultation with the Australian DH-104 operators. The Bureau will be advised of the Authority's final decision on the Interim Recommendation at the outcome of the investigation.

Response status: CLOSED–ACCEPTED

4.2 Interim safety action by BASI

Upon receipt of the National Test Pilot School test flying report, two copies were forwarded to the CAA in Canberra, one copy to Airworthiness and Operations and one to the CAA test pilot.

The contrast between the engine failure emergency actions listed in the operator's operating instructions and the DH-104 aircraft manufacturer's recommended actions has been brought to the attention of CAA Safety Regulation and Standards, Moorabbin District Office, Flying Operations Section.

The CAA Safety Regulation and Standards, Moorabbin District Office was made aware of the existence of the zero-flap take-off chart for takeoff from Essendon.

Furthermore, the NTPS report made the following recommendations:

1. The Flight Manual should be changed to include the position error corrections found during the course of this flight test.
2. The Flight Manual should be changed to tell the pilot to climb at 88 KIAS with the flaps up to maximise OEI rate of climb.

The Bureau of Air Safety Investigation requested that the Civil Aviation Safety Authority (CASA) provide details of proposed action to address the NTPS recommendations.

CASA response

CAR 3.779 required that information concerning emergency speeds and procedures be published in the Flight Manual. CASA is about to issue a General Amendment to the Flight Manual for DH-104 Series 5 and 6 aircraft to achieve the following:

In Section 3 HANDLING two statements will be added to paragraph 3.3. The first will advise that flap may be retracted from 20 degrees to the UP position at 86 kts IAS or any faster speed. (The 20 degree flap take-off charts show a safety speed of 84 kts IAS.) The second will advise that, at MTOW and flap in the UP position, the airspeed for maximum rate of climb with one engine inoperative is 88 kts IAS.

At a weight of 8,800 lb (3991 kg) and with flap in the UP position, the airspeed for maximum rate of climb with one engine inoperative was found to be 93 kts EAS. See page 11 of Reference A [NTPS report]. Page 11 also shows that at 93 kts EAS the airspeed error is 5 kts so that the indicated airspeed for maximum rate of climb is 88 kts.

There are presently four DH-104 aircraft on the Australian register with a MTOW of 8,800 lbs (3991 kg). The Flight Manuals for these four aircraft will be amended promptly as shown above. There are also four DH-104 aircraft with MTOW different to 3991 kg. These are Series 1 (3855 kg) or Series 8 (4159 kg) aircraft. Reference A provides no information about these aircraft. However, they have the following statement in their Flight Manuals:

WARNING: WITH FLAPS EXTENDED, CLIMB PERFORMANCE ON ONE ENGINE LIMITED. IT IS THEREFORE ESSENTIAL WHEN EARLY ENGINE FAILURE HAS OCCURRED (OR SIMULATED FOR PRACTICE OR CREW TRAINING) THAT THE FLAPS ARE RAISED IMMEDIATELY AFTER ALIGHTING GEAR RETRACTION.

In addition, speeds V_x and V_y are specified in the manuals for the Series 1 and Series 8 aircraft. At this stage it is not intended to alter these Flight Manuals.

4.3 Safety advisory notice

The following safety advisory notice is issued:

SAN960007 **The Bureau of Air Safety Investigation suggests that the Civil Aviation Safety Authority include, in all systems of maintenance for Australian DH-104 aircraft, reference to the need to periodically check the adjustment of the landing gear selector lever.**

4.4 Safety action by the operator

1. Since the accident, the operator has elected to carry out most takeoffs in the zero flap configuration.
2. Since the accident, the operator has instigated a series of non-destructive testing (X-ray) examinations of all of his existing fuel pumps to ensure grooves and holes are still present for internal lubrication. No defects have been found.