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REPORT # H80003
AIRCRAFT ACCIDENT
AIRWEST AIRLINES LTD
DE HAVILLAND DHC6-200 (FLOATS)
REGISTRATION CF-AIV
VANCOUVER HARBOUR, BRITISH COLUMBIA
00:42 GMT, SEPTEMBER 3, 1978

AVIATION
SAFETY BUREAU

AVIATION SAFETY
INVESTIGATION

BUREAU DE LA SÉCURITÉ
AERONAUTIQUE

ENQUÊTE SUR LA SÉCURITÉ
DE L'AVIATION

"This accident was investigated to provide guidance toward the prevention of a recurrence. The content of this report is confined to relevant circumstances and is published for accident prevention purposes only."

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SYNOPSIS

A visual approach was being made to Vancouver Harbour. When at 175 ft above the surface and approximately 2500 ft from the intended landing area, the aircraft suddenly rolled left and plunged into the water.

This accident was investigated and this report was prepared by the Aviation Safety Investigation Division, Aviation Safety Bureau, Transport Canada.

Honourable the Minister of Transport

This Accident Investigation and this Report have been audited by the Aircraft Accident Review Board. The Board has considered all information available including that from involved parties. The Board agrees with the contents of this Report for release as public information.

Aircraft Accident Review Board

ERRATA

Index	1.16.7	Flat should read flap
Page 2	1.1	Victorial should read Victoria
Page 7	1.12.1	2nd sentence - tower should read lower
Page 21	1.16.7	Title para 1 - Flat should read Flap 1-½" orack should read crack

INDEX

1. FACTUAL INFORMATION

1.1 History of the Flight

Twin Otter CF-AIV, operating as a Scheduled VFR flight departed Victorial Harbour, B.C. at 00.18*, 3 September 1978, with Vancouver Harbour water-aerodrome as destination. The estimated time en route was 20 minutes.

The flight proceeded normally and reported by Active Pass at 2000 ft. This altitude was maintained in order to cross the Vancouver Control Zone in accordance with standard procedure; once out of the control zone, a slow descent was begun towards Vancouver Harbour.

Normal radio procedures were followed as the flight reported by standard visual reporting points. Just before joining final approach, the transmission, "AIV, Third Beach", was made and landing clearance was given to the flight by the Harbour Tower. The approach continued, and when the aircraft reached approximately 175 feet above the surface, nine ground witnesses heard a loud noise from the aircraft. Two surviving witnesses also heard a noise. Power was subsequently applied and CF-AIV yawed left, rolled in the same direction and plunged into the harbour in a left-wing and nose-down attitude, 2500 feet from the intended landing area.

* All times in this report are GMT which is Pacific Daylight Time +7 hours.

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An ELT, (Emergency Locator Transmitter), tone was heard by the tower controller 54 seconds after the radio call at Third Beach. The controller called the aircraft several times but there was no response.

The flight had been of 24 minutes duration.

1.2 Injuries to Persons

Injuries	Crew	Passengers	Others
Fatal	2	9	-
Serious	-	2 *	-
Minor/None	-	-	-

1.3 Damage to Aircraft

The aircraft was destroyed on impact with water and the wreckage was confined to a small area.

1.4 Other Damage

Nil.

* One, a pilot with 300 hours experience, employed as 2nd officer on 747 aircraft. One, a tour-guide.

1.5 Personnel Information

(a) The pilot in command was 27 years old and held a valid Airline Transport rating. He had been working for the Company since 1974 and had accumulated in excess of 6,000 hours flight time of which 3,600 were on the Twin Otter. His last medical, June 27, 1978, was assessed as Category I, with no restrictions. Transport Canada medical records indicate that his first examination for a pilot's licence was in 1967 when it was noted that he had previously experienced back pains for two years, diagnosed by an orthopaedist as epiphysitis. The records show no indication of physical problems thereafter except for a torn ligament in his left ankle in 1968.

Because of the previous diagnoses, enquiries were made of Company personnel and of various physicians who had examined him. It was found that his disabilities during previous employment had prevented him from doing any heavy work such as lifting, and he would not load aircraft. Information was also obtained that on one occasion while employed by another company, he was in so much pain following a flight, that he had to be lifted from the cockpit and out of the aircraft. Enquiries did not reveal a firm diagnosis but evidently he had been complaining of pains in the back and swelling and pain in the left and/or right foot, periodically since 1964. A diagnosis of gout had been considered at one time.

On the day of the accident he had flown 3.5 hours which consisted of two flights to Nanaimo and three flights to Victoria Harbour. The first (to Nanaimo) commenced at 1503 hours, the accident occurring on the return leg of the third flight to Victoria Harbour. His activity during the day appeared to have been normal although he was observed to have rested supine on a counter top in the Vancouver office before his last flight to Victoria.

(b) The Cabin Attendant was 23 years old and held a valid Commercial Pilot's licence. He began flying in 1974 and had acquired over 450 hours of flying time. Normally he sat in the right hand pilot seat but was not authorized by the Company or Transport Canada to handle the controls of the aircraft on revenue flights. Discussions with other Company pilots, however, indicated that this Captain allowed him to fly the aircraft on occasion and a survivor stated that he did so during the cruise segment of the accident flight. His last medical examination was in March 1 1978 and he was granted a medical Category I with no restrictions. He was employed by the Company in April 1977 as a ticket agent but later transferred to the position of Cabin Attendant on Twin Otters.

1.6 Company/Aircraft Information

The aircraft was operated as a Class II service. The Vancouver Harbour to Victoria Harbour segment of the company's operation is classed as Scheduled VFR (floats). For this type of operation the company uses a crew complement of a captain and one cabin attendant; the latter was permitted by Transport Canada to occupy the first officer's station.

The aircraft, Serial No 215, was manufactured in 1969 by de Havilland Aircraft of Canada Ltd and powered by two Pratt and Whitney PT6A-20 engines. The maximum authorized take-off weight was 11,600 lbs; the weight and centre of gravity of CF-AIV, at the time of the accident were within limits.

The maintenance facilities, records and practices were examined and no deficiencies were found. However, the maintenance procedures in effect at the time, precluded adequate inspection of the installed flap control rods.

1.7 Meteorological Information

The weather at the time of the accident was VMC*. The post weather sequence, recorded at 00.52Z, showed 1500 feet scattered, 4000 ft scattered, estimated 7500 feet broken, 9000 ft overcast. The visibility was 12 miles, temperature 16°C, dewpoint 14°C, wind 050° at 3 knots.

* Visual Meteorological Conditions

Pilots flying in the area at the time of the accident reported no significant turbulence.

1.8 Aids to Navigation

There are no aids to navigation serving the Vancouver Harbour water aerodrome.

1.9 Communications

The aircraft's communication equipment was selected to appropriate local frequencies and operated correctly. Communications with Air Traffic services and company dispatch were normal and routine.

Because of the location of the Vancouver Harbour tower it was impossible for controllers to view continuously aircraft landing from the west until on short final; because of this, the first indication of a crash was the transmission of the ELT tone. Confirmation of the accident was obtained by communication with the pilot of another aircraft who observed the impact. The controller immediately notified the Vancouver Marine Traffic Centre in accordance with Vancouver Harbour emergency procedures.

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1.10 Aerodrome

The water aerodrome is situated 49°18'N 123°07'W, is generally well sheltered and has a tidal range of 14 feet. The alighting area is 4 miles long in an east/west direction and one mile long, north/south. A note of caution on the chart which depicts the water drome states "Aircraft arriving Vancouver from the west are to approach over Beaver Lake in Stanley Park."

1.11 Recorders

There was neither a voice nor a flight data recorder on board, nor was there a requirement for either.

1.12 Wreckage and Impact Information

1.12.1 Power Plants

Left Engine-PT6A-20, Serial Number PCE 20786

The engine had separated from the wing at the firewall area. The tower area of the cowl and nacelle had received impact forces in an outboard direction. The inboard side of the engine support structure had been in tension while the outboard side was in compression. Engine control cables had failed in tension. A section of the propeller control linkage (from firewall to cambox) was found in the nose of the left float. The propeller and engine were recovered as a unit.

1.12.2 Right Engine-PT6A-20, Serial Number PCB 21873

This engine separated from the wing at approximately the same location as the left. Examination of the support structure showed failure from torsional forces opposite to the direction of rotation. This was probably a reaction from the propeller striking the heavier fuselage structure of the cockpit. Engine control cables had failed in tension. The propeller and engine were recovered as a unit.

1.12.3 Left Propeller - Hartzell Model HC-B3TN-3* Serial BU577

Two blades were bent aft at mid-span and the third had a slight forward bend near mid-span then an aft bend near its tip; there were no nicks or gouges. The latch pins of two blades were corroded in the recessed position; the pin of the third blade held the piston and the other two blades at the zero thrust angle; when the latch was depressed the propeller piston and blades travelled to the "feather" position. The spinner, which had impacted onto the piston and counterweight assemblies, was ruptured, and provided no valid information.

* See app. "A" (Description of propeller function)

Externally there was visible damage to the propeller hub/spider assembly. The blade links, attaching the blade/clamps to the piston, appeared compressed; the guide collar had been impacted by all three of the blade-link pins and/or zero thrust latch cams; also some damage to the guide rod bushing circlips was evident. It was therefore apparent that the piston at some time during impact had moved to the full reverse position.

1.12.4 Right Propeller - Hartzell HC-B3TN-3 Serial BU1492

All three blades had been bent in a forward direction, the centre of the bends being about mid-span. Two blades had sustained severe tip damage resulting in loss of metal; the third blade was intact. The piston was in the feathered position. The spinner had impacted onto the piston and counterweights but no useful information was gained from the spinner damage.

1.12.5 Engine Control (Overhead) Console

The engine control console was virtually destroyed by impact forces. Power levers, the right propeller lever, fuel levers and their respective pulleys were shattered or broken out of the console; the left propeller lever was not recovered. The right propeller mechanical stop appeared to have been out of rig by comparison with other adjacent stops. Mod 6/1223 (Prop/Power

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lever interlock) had been fitted. The power-lever microswitch, was removed, tested and found to be serviceable. No pre-impact lever positions could be determined because of extensive damage.

1.12.6 Fuel System

The fuel cells were ruptured on impact which precluded determination of the quantity or quality of the fuel. It was reported that a substantial amount of fuel was present on the water after the accident. Weight and balance calculation showed that there should have been approximately 310 pounds of fuel remaining. The fuel ejectors and motive line check-valve (with finger strainer) from the aft tank (left engine) were checked and found to be clean and serviceable.

1.12.7 Structures

During recovery attempts to lift the aircraft from the water and to remove the passengers, the fuselage suffered further damage. A cable had been attached to the empennage and the wreckage had been dragged over the harbour floor, tearing away pieces of significant value.

The fuselage was lifted from the water, taken by barge to a dock and transported by flatbed to Vancouver International Airport. It was then placed in a storage area and impact damage

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was analysed to determine the breakup sequence. Initially safety investigators considered that both right and left flaps were down at impact. Some months later, a swaged portion of a flap actuating rod, the fracture-face of which mated with the fracture-face of a portion that was recovered with the aircraft, was retrieved from the harbour. The wreckage was moved to a warehouse and re-assembled on jigs to determine the direction of initial impact forces and to re-assess the sequence of breakup.

Damage to the aircraft, supported by the injury pattern of the passengers, indicated that the left wing tip impacted the water prior to any other part of the aircraft. Damage examination further indicated that the flaps were up on the left and down on the right at the time.

(See Analysis of Sequence of Break-up Appendix "D".)

1.13 Autopsy and Biochemical Information

1.13.1 Captain

The autopsy showed no evidence of disease which might have impaired his flying performance and death was due to asphyxia due to drowning. The injury pattern suggested that at impact the pilot was thrown towards his left side and that his head contacted the instrument panel. Also it would appear he either held the control column neutral or slightly forward with his left hand, while his right hand was above his head in the area of the overhead controls. The former conclusion was supported by contact marks on the yoke assembly which indicated a near neutral position.

Injuries to his feet and an imprint on the sole of his right shoe suggested that he was applying moderate pressure on the right rudder with his right foot placed at an angle on the pedal with his toes outward. This finding is compatible with foot movement limited by pain and led to enquiries into his medical history.

Biochemical tests on the blood, liver and urine did not show the presence of any alcohol or drugs. A carbon monoxide saturation level of less than 10% was present in the blood and was considered insignificant.

1.13.2 Cabin Attendant

Autopsy of this crew member showed no evidence of disease which might have affected his duties. From his injuries it was apparent that he had been struck by the right propeller during the crash sequence and died instantly. At the time he was leaning forward and to the left. The injury pattern does not suggest that this person was handling any flight controls at the time of impact.

Biochemical analysis of blood did not show any presence of alcohol or drugs and less than 10% of carbon monoxide was found.

1.13.3 Passengers

The position of the eleven passengers in the aircraft was determined from information provided by survivors. Of the nine passengers who died in the accident one was killed by impact forces. This passenger was seated in the first row left hand side. Two other passengers probably lost consciousness and died of aspiration of water. One of these two was seated in the third row on the right hand window seat and suffered a fracture of the skull and contusion of the brain. The other, sitting in the first row right window, was found to have a small fracture at the floor of the skull and some fractured ribs. The injuries to the remaining passengers did not appear sufficient to cause any unconsciousness and their deaths were due to drowning. Of the two survivors, one was in the front row centre seat and the other occupied the rear seat. Both sustained serious injuries.

The skin on some bodies showed the effects of exposure to fuel.

1.14 Fire

There was no fire.

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1.15 Survival Aspects

The accident was survivable.

All passengers sat at window seats except for the front row survivor who sat in the middle seat; seat belts were used by all on board.

The impact forces were not severe enough to produce failure of seat attachments to the floor and wall, which are required to withstand an ultimate acceleration of 9G forward.

There appeared to be a progression to the injury patterns, being more severe to the forward passengers and diminishing toward the rear. Also most of the injuries were more severe on each person's left side.

A survivor seated on the middle seat of the front row suffered multiple injuries but, because of a break in the fuselage, was able to swim free. The survivor occupying the rear window seat injured his back but managed to escape with difficulty through the rear baggage door which had opened at impact.

The survivors' statements indicate that the standard pre-take-off briefing described in Airwest's Operations Manual was

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not given in full. The actual briefing was given in Japanese, the native tongue of all the passengers, by the tour guide accompanying them, and included only the use of seatbelts and smoking restrictions. Thus the emergency exits were not pointed out nor was reference made to the safety information cards which show the location of the emergency exits.

A survivor stated that there were several people standing when he left the aircraft. Those passengers who may have been conscious after impact but drowned, possibly did not escape because they were unaware of the position of the emergency exits. Although the main door was jammed because of distortion, the main emergency exit on the right side and one on top of the fuselage were available for use and functioned properly.

1.16 Tests and Research

Voice Identification

The voice transmissions from the aircraft to the tower were positively identified as those of the cabin attendant; they were consistent with a routine approach to the Vancouver Harbour, with no indication of concern or alarm.

The interval between the last transmissions and the ELT signal was determined to be 54 seconds. The ELT transmitted for

57 seconds. Analysis of the tape revealed no other significant information.

1.16.1 Airspeed Indicator Analysis

The pilot's airspeed indicator was severely damaged by direct impact. The instrument case and face-glass were missing. The dial, pointer and aneroid were all deformed and corroded by sea-water. The pointer was indicating 195 knots but no corresponding impact imprints were found on the dial. However, when the deformations of the dial and pointer were matched, the pointer indicated 75 knots. Examination of the mechanism also showed that it was jammed in the 75 knot position; the sector gear and pointer pinion did not reveal any marks which would indicate their disengagement during the impact sequence.

1.16.2 Emergency Locator Transmitter (ELT)

The ELT was undamaged by impact forces but there was evidence of the effects of having been immersed in sea-water. This damage was limited to corrosion deposits on most terminal posts and some staining of the circuit board and a number of components.

The inertia switch had been triggered. Functional testing of the switch to establish its trigger threshold was not

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considered practical. As received, the switch was unserviceable because of the sea-water deposits and although removal of these deposits and re-furbishing the switch would have allowed functional testing, it would probably have altered the switch characteristics. It would be equally reasonable to assume that the inertia switch was within required tolerances at the time of impact. With this assumption, the deceleration experienced by the ELT along the aircraft's longitudinal axis would have been greater than 5 G.

1.16.3 Systems Examination

Left Propeller

The propeller dome (piston), on examination showed a light impact mark at +5°, (the blades latch at +2°) and a heavier and more obvious mark at -9°. As discussed previously, the propeller was found to have been in the full reverse position so that it must have struck the water at some angle above +5° blade angle, and then the blades were progressively driven back to the reverse position. When the turbine stopped rotating the internal propeller spring drove the blades toward feather; they were, however, trapped in the zero thrust position by the engagement of the serviceable latch.

1.16.4 Right Propeller

On examination it was determined that the blades were in positive pitch (+23° at time of impact). (See App C)

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1.16.5 Propeller Operating Oil Pressures

To determine the propeller-oil operating pressures during a wide range of power/propeller selections, tests were conducted using a series 300 DHC6 powered by PT6A-27 engines fitted with an oil line routed from an oil pressure source between the propeller governor and the overspeed governor. The oil line terminated in a 0-1000-psi, (pounds per square inch), pressure gauge with 10 psi graduations.

Oil pressure indications were noted during start, taxi, reverse, take-off, climb and descent. The results showed very rapid response in all modes with maximum pressures being reached during take-off, and minimum, at coarse-pitch constant speeding.

Although these tests were carried out on a -27 engine, the results obtained are considered to apply also to the -20 engine.

1.16.6 Flap Control Tube

Failure analysis of the recovered portion of the rod was conducted to determine possible pre-impact, impact and post-impact conditions and a probable sequence of failure for the separation of the rod end (ASE Report LP 65/79, refers). Later scanning electron microscope confirmation of aspects crucial to the determined failure sequence is covered in ASE Report LP 23/80 (Appendix E).

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From these analyses, it was concluded that prior to impact, the flap control rod contained one major and other secondary stress corrosion cracks. The main fracture was 2-3/4 inches in length and extended longitudinally from the open end of the tube. This crack alone was shown to have reached a sufficiently advanced stage of growth to bring about separation of the tube from the end cap fitting at less than applied loads from the full flap setting in the approach configuration. In the case of the accident aircraft, the flaps were set at the full down position during the approach.

NOTE: *Stress corrosion cracking (SCC) is a mechanism of brittle fracture in a nominally ductile material involving a combination of electrochemical action and sustained tensile load. Failure occurs over a period of time which is determined by the severity of the conditions which are prerequisites for SCC, namely:*

i *a particular metallurgical requirement, or susceptibility in terms of the composition and structure of the alloy;*

ii *a sustained tensile stress, either applied or residual in excess of some minimum threshold level below which SCC does not occur;*

iii *a specific corrosive environment.*

For CF-AIV all three conditions for SCC were present:

i The 2024-T3 alloy of the flap control rod is known to be highly susceptible to SCC and not recommended for applications where high sustained tensile stresses are likely to act in either the long or short transverse grain directions;

ii The magneforming of the tube to the end fitting leaves residual tensile (hoop) stresses in the area of the swaging of sufficient magnitude to exceed the threshold stress level for SCC to occur in the long transverse grain structure of the tube;

iii CF-AIV's normal working environment of sea air, coupled, with sulphur-bearing gases in the ever present exhaust fumes is a more than adequate corrosive environment to support SCC.

It should be noted that operationally induced loads generally have a negligible effect on the stress corrosion cracks. Accordingly, cracking is a function of calendar, not flying time, through the sustained residual stresses in the swaged rod end.

Aircraft maintenance records show that inspections had been done in accordance with approved inspection procedures. However, detection of a crack at the swaged end of a control rod would have been quite unlikely under the inspection requirements as detailed in EMMA* checks, as they existed prior to the accident. Proper inspection would require both physical and visual access to the rod ends and this would entail removal of the paint for proper penetrant checks.

* EMMA (Equalized Maintenance Maximum Availability)

1.16.7 Flat Rod Tensile Tests

Tests were performed on sample flap push-rods to simulate the effect of a longitudinal crack in the magneform tube end on the load carrying capability of the rod. It was found that there is considerable variation in tensile separation load versus pre-crack length from sample to sample i.e.: -

1-1/2" crack	- 2400 lb.
2-1/4" crack	- 510 lb.
2-1/2" crack	- 610 lb.
2-3/4" crack	- 1680 lb.

Manufacturer's tests showed that when two rods were cracked in a manner similar to the accident case, one (2.5 inches long), failed at 825 lbs and the other, (2.75 inches long), at 500 lbs. It is considered that the difference in separation loads is because of the variation in original clamping action (i.e. variation in material strength, fit, residual stresses). The manufacturer stated that the working loads on the rod during a normal approach with flaps-down condition vary from 330 to 650 lbs, whereas the design breaking load is 8000 lb. and the nominal breaking load is 11000 lb. The tests therefore clearly showed that cracks greater than two inches (as found on the CF-AIV rod), and which have progressed to the free end, could lower the load-carrying capacity of the rod into the range of the actual loads with flaps down.

See App. "F" Metallurgical Analysis

See attached pictorial of Swaged-connection. Figure 3.

1.17 Additional Information

1.17.1 Cine Film Analysis

An 8 mm motion picture camera containing a cassette with an exposed roll of colour film was recovered from the wreckage; both were damaged by immersion in the sea-water. Development of the film showed that the camera had been operated by a passenger from a starboard rear seat to within 30 second of impact.

The results of the analysis are tabulated on the attached Appendix "G" and "G (1)" and establishes;

- the position of the aircraft at 9 (nine) stations in metres in the UNIVERSAL TRANSVERSE MERCATOR PROJECTION, Zone 10;
- the height of the aircraft at the same 9 stations;
- the wing flap angles at the first, centre and last frame stations;
- the pitch attitude of the aircraft to the true horizon at the above 9 stations;
- the rate of descent; and
- the ground speed in knots and metres per second.

1.17.2 35mm Photograph - Analysis

Approximately 10 seconds prior to impact the aircraft was photographed by a person who was standing in Stanley Park, Vancouver. After taking the photograph the cameraman lowered the camera and closed the front flap, (about 2-3 seconds). He then heard a loud "bang", looked towards the aircraft and saw it rolling to the left.

The related photographic slide was analysed and the left flap angle, altitude of the aircraft and its distance from the camera were determined. (See Appendix "H")

An Azimuth (the yaw of the fuselage to the camera negative) of 39° and Tilt (the elevation of the lens longitudinal axis) of 27° were established.

From this information further calculations indicated that the horizontal range from the camera position to the nose of the aircraft was approximately 388 feet. The aircraft nose was between 197.87 (minus 5.5%) feet above the camera. The slant range from the camera to the nose of the aircraft was approximately 433 feet. The camera angle of view (in the horizontal) was 39.8 degrees. The left main flap was extended to the equivalent of 33° cockpit-indicator setting.

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1.17.3 Witnesses

Witness information is incorporated in this report.

Persons interviewed included eye-witnesses, survivors, peers of the accident pilot and company operations personnel.

There was an abundance of witness information available from persons who were in or near a wooded park adjacent to the impact area and there was general agreement in their descriptions of the flight trajectory. Six witnesses stated that the approach was normal; five stated it was low; none said it was high.

Although not all saw the aircraft strike the water because of boathouses and other obstructions, 21 witnesses gave evidence that they had viewed the aircraft in left-wing-low and nose-down attitude to within approximately 50 feet of the surface.

Those that reported they heard a noise described it in different ways such as, "heard a crack", - "sounded like a cable breaking", - "sounded like a metal drum being struck by a hammer", - "sounded like the load shifted from one side to the other", - "a metallic noise", "like a compressor stall on a 747 engine". Two stated they heard engines sputtering. Witnesses standing laterally to the aircraft were within 500 feet of the estimated position of where the noise originated. One witness who heard the noise was standing about 300 feet away.

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There was conflicting evidence introduced by the survivors, as each placed the geographical area over which they had heard the noise approximately 12 seconds prior to the point where ground witnesses asserted the noise had occurred. Although the analysis of the 35 mm slide showed that the flaps were almost down and the engines running 10 to 12 seconds before impact, the survivors indicated that there was a time-frame of 15 to 30 seconds from the noise to the start of the left roll.

Following analysis of the witness evidence it was found that if a flap had failed at the position stated by the two surviving witnesses, then the aircraft would have been unable to continue to the impact area; or if a left propeller problem occurred in the same area, the pitching, wobbling and yaw described by the survivors would have been obvious on the cine film. Neither was the case. It was therefore, concluded that the time-frame of 15-30 secs to the critical phase of the accident sequence was unreliable evidence.

1.17.4 Twin Otter Flap Rods

In 1973, flap control tubes fabricated from a 2024 alloy in a T-81 temper were released for service by the manufacturer. This material is less susceptible to stress corrosion cracking than the 2024 T-3 tempered alloy specified previously. This type of rod was fitted to all aircraft subsequent to Serial No 420. At the time of the accident the manufacturer had neither required nor recommended the retro-fitting of the new tube to existing aircraft. The flap system installed in CF-AIV at the time of the accident was manufactured to the old specification.

Comments from the manufacturer stated that the modification to 2024-T3 control rods followed upon a study of stress corrosion problems during the period 1968-70 and that operators were advised of the problem by Technical Advisory Bulletin 6/626 in November 1970. This was followed by Modification 6-1487, December 1973, which covered the change of control rods. The modification applicability was quoted as Aircraft No. 1 and subsequent and "production cut in" as Aircraft No. 421 and subsequent. Transport Canada's Airworthiness Division approval was not mandatory.

2.0 Resumé of Events During the Approach

Following flight from Victoria Harbour the aircraft commenced descent towards Vancouver Harbour shortly after the crew had reported by Third Beach.

From the cine camera data it was calculated that 29 seconds later the aircraft was over position "1", (as indicated on Figure 1) and that the flap at that time had been extended to 20°.

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The descent toward the harbour aerodrome continued for another 8 seconds at which time the film data showed that the flap had reached a position of 28° and that the average approach speed for the 8 seconds had been approximately 76 knots.

Following the termination of the cine film a witness took a photograph of the aircraft. It was calculated that the flaps had reached a setting of 33°* and that the aircraft was approximately 200 feet above the water surface, about 2500 feet from the intended landing area.

Approximately three seconds later a loud "noise" was heard from the aircraft. According to ground witness testimony the aircraft subsequently yawed, rolled to the left and continued an uncontrolled descent into the water. These witnesses variously state that the time from noise to impact was between 6 to 10 seconds.

2.1 Analysis

On the third day following the accident a helicopter, equipped with a video camera, was used to simulate the flight path of the accident aircraft prior to impact. Witness statements were used to trace the flight path and the resulting video film

* All flap positions are indicator settings.

indicated that the aircraft had entered an unusually violent left roll followed by a high rate of descent to impact.

After examining the initial statements of the witnesses, the following possibilities as a plausible lead event in the accident sequence were considered:

1. The aircraft had stalled aerodynamically at a height too low to effect recovery.
2. An unusual amount of drag on the port side had caused a left yaw and uncontrollable roll in the same direction.
3. Loss of lift on the port side caused a roll moment that made the aircraft uncontrollable.

2.1.1 Aerodynamic Stall

This hypothesis was discarded for the following reasons:

1. The DHC6, with full flap and at a power setting of 10 to 12 pounds torque requires a significant amount of nose-up attitude, (4° to 5°) before a stall can be induced. Only two witnesses, about three quarters of a mile away, stated that the aircraft was in a climb attitude.

(An investigator who watched several aircraft approach the harbour from the position of these witnesses, (in a high-rise apartment), noticed that all aircraft appeared to be in a nose up attitude. This illusion was attributed to the slope of the terrain toward the water's edge.)

2. It is improbable that an aerodynamic stall would generate an unusual noise.

3. The airspeed indicator was impacted at 75 kts, approximately 20 kts above the stall speed of this type of aircraft when in full flap configuration.

2.1.2 Unusual Drag - Left Propeller

The latched left propeller and other physical evidence indicated that at some time the blade angle had been in full-reverse position.

The evidence of the two on-board witnesses indicated that the pilot had both hands on the control column at the time of the unusual noise. This would be incompatible with an intentionally induced "Reverse Beta" to increase the rate of descent. Additionally; from the cine and 35 mm photo-analyses, it was concluded that the approach was normal.

Experts from the propeller manufacturer stated that when this type of propeller is impacted while at low power the tendency would be for the blades to be driven toward reverse mode.

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However, there is inconsistency in this hypothesis, since in other accidents where both propellers had struck, physical evidence showed that only one of the propellers displayed reverse blade angles. In the CF-AIV accident, the right propeller dome showed that the blades were at plus 23° angle, the mark presumably made when the propeller struck the right side of the fuselage, right door sill and the First Officer's seat.

The left propeller dome showed an initial mark of plus 5° and subsequent chatter marks, including minus 9°, as the blades were driven toward full-reverse. If the left propeller had errantly travelled into reverse mode prior to impact, the 5° mark could not have been made on the dome. It follows that the left propeller was not in the latched condition at impact.

The variance of blade angles between each propeller, (left plus 5° and right plus 23°), can be explained by the physical evidence of the impact. When the left wing struck, and was folded back, the left power and propeller control cables were stretched in a manner that would disrupt the pre-set pitch position.

It is considered that as each blade travelled deeper into the water, the propeller progressed toward the full reverse position; because the aircraft cartwheeled at impact the propeller disc may have struck the water in an unusual attitude.

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Flight tests in a - 300 and a - 200 Twin Otter showed that when "reverse beta" was induced, followed by application of 10 or 12 lbs of torque, airframe "buffeting" was very significant. It was considered by the safety investigation team that if a propeller travelled uncontrolled into the reverse regime and full power was then applied, the vibration or buffeting would be severe, (manufacturers' flight tests proved this to be so). The witnesses who survived the accident did not speak of vibration during the aircraft's left roll and dive into the water.

There was no evidence to indicate that the left propeller control system had not been functioning properly prior to the accident.

2.1.3 Loss of Lift - Port Side

The possibility that a collapse of the complete left flap system would contribute to the pre-impact trajectory of the aircraft, was considered.

Manufacturer's flight tests, with one flap secured in the up position and the other operating normally, demonstrated that the aircraft would initially yaw towards the retracted flap. This was consistent with what witnesses reported.

.../32

These flight tests conclusively proved that with more than 25° asymmetric flap the aircraft would be uncontrollable at "flight idle", and that at "full thrust", control would be lost with asymmetric flap in excess of 20°.

To account for the collapse of the left flap system, an extensive investigation of the flap control tube and of the sequence of structural failure, with the results given in paras 1.12.7, 1.12.6, 1.16.7 and Appendix "D" were derived from ASE reports LP65/79 and FI 194/78. The preponderance of evidence supports the conviction that the separation of the flap rod from its end fitting occurred in flight and that the subsequent separation of the end of the flap push-pull rod occurred during impact with the water.

It was therefore deduced that the loss of control was in-flight collapse of the complete left flap system.

2.2 Flight Path - Computer Simulations

Despite the fact that the weight of evidence supported the conclusion that the accident resulted from the collapse of the left wing flap system, further efforts were made to examine the effects on the flight path in the event of a latched propeller arising from the deliberate or unintentional selection of the

.../33

"reverse beta" mode. To this end, trajectory simulations were made by the manufacturer using a computer model designated as program EN 821; and advantage was taken of the model also to simulate trajectories arising from a collapse of the left wing flap system.

The results of these simulations cannot be definitive, in that there was convincing evidence from the results of the simulations that either of the two causes of asymmetry was compatible with the known facts relating to the terminal phase of the flight. On the other hand, the simulation evidence did tend to confirm that a condition of "reverse beta" propellor latch could produce a terminal trajectory somewhat similar to that produced by a condition of asymmetric flap.

The results of these simulations therefore provided no adequate grounds to reject the deduction that the left flap system failed in flight, but did induce concern with the safety of the "reverse beta" system in operational terms.

3.0 Findings

1. The final approach to land was normal until an unusual noise occurred followed by loss of control.

.../34

2. The aircraft dived into the water with left wing down, nose down and with some sideslip. Value of roll, pitch and yaw, at impact could not be estimated with useful accuracy.

3. At impact, the complete left flap system was in the retracted position.

4. The inboard span-wise push-pull flap control rod, (inboard bell-crank to inboard rod, PT # C6CW-1029-1), was severely stress corroded and had at least three longitudinal cracks; the rod had separated from its inboard fitting.

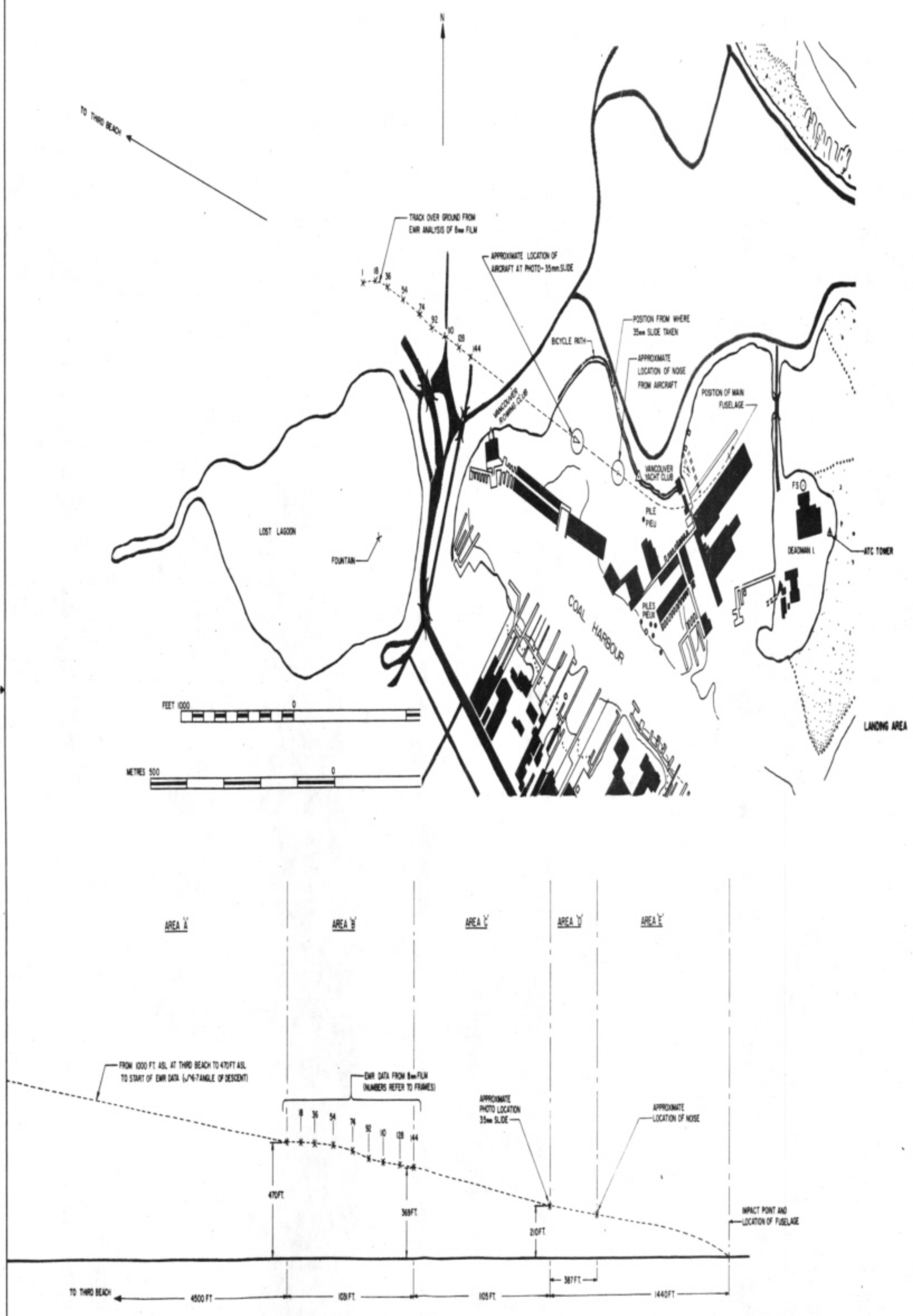
5. It was deduced that the in-flight failure of the left-hand inboard flap control rod led to sudden retraction of the complete left-hand flap system and sudden loss of control.

6. The passengers had not been briefed in evacuation procedures by either the flight-crew or the tour guide.

7. The crew was qualified for the type of operation in accordance with Transport Canada regulations After the failure of the left flap control rod, no action by the pilot could have averted the accident.

FIG-1

Transport Canada Transports Canada
Air Air



AREA 'A'

AT THIRD BEACH—LAST RADIO TRANSMISSION FROM AIRCRAFT C-FAIV ACKNOWLEDGEMENT OF LANDING CLEARANCE 54 SECONDS PRIOR TO IMPACT AND ELT SIGNAL. ESTIMATED 1000FT ASL IS CONSISTENT WITH A NORMAL APPROACH.

AREA 'B'

FROM AN 8mm MOVIE FILM RECOVERED FROM THE WRECKAGE, IT WAS EVIDENT THAT THE CAMERA WAS IN OPERATION THROUGHOUT A PORTION OF THE APPROACH.

THIS FILM EMBRACES A SEGMENT OF 7.99 SECONDS—FRAME 1 THROUGH 144.

AVERAGED RATE OF DESCENT = 758FT/MIN
AVERAGED GROUNDSPEED = 76 KNOTS

FLAPS WERE AT THE 20° SELECTOR POSITION AT FRAME 1 AND AT 28° SELECTOR POSITION AT FRAME 144. THIS IS EQUIVALENT TO 1° OF FLAP EXTENSION PER SECOND.

AREA 'C'

PRIOR TO THE CRASH–A PROFESSIONAL PHOTOGRAPHER LOCATED ON A BICYCLE PATH IN STANLEY PARK TOOK A PICTURE OF THE AIRCRAFT.

FROM THIS 35mm SLIDE IT WAS DETERMINED THAT THE AIRCRAFT WAS AT A HORIZONTAL DISTANCE OF 388FT AND 198FT ABOVE THE CAMERA AZIMUTH AND LOCATION WHERE PHOTO WAS TAKEN.

THE AIRCRAFT APPEARED NORMAL AND THE FLAPS WERE AT A SELECTOR POSITION OF 33°.
NOTE FULL FLAP IS $37\frac{1}{2}°$

TIME FROM FRAME 144 IS 8.6 SECONDS.
AVERAGE GROUNDSPEED = 76 KNOTS.

THE DISTANCE FROM THE PLOTTED POSITION OF THE 35mm SLIDE TO THE USUAL TOUCHDOWN AREA IS APPROXIMATELY 2500 FT.

AREA 'D'

A NOISE EMANATED FROM THE AIRCRAFT APPROXIMATELY 3 SECONDS AFTER THE PHOTO WAS TAKEN.

USING A GROUNDSPEED OF 76 KNOTS PLACES THE AIRCRAFT 387FT BEYOND THE LOCATION OF THE 35mm SLIDE. OBSERVATIONS OF GROUND WITNESSES SUBSTANTIATE THIS LOCATION.

FROM THIS POINT A LEFT BANKED TURN AND DESCENT BEGAN AND CONTINUED TO IMPACT.

AREA 'E'

POINT OF IMPACT, LOCATION OF MAIN FUSELAGE AFTER THE CRASH. THE AIRCRAFT ENTERED THE WATER IN A LEFT WING LOW AND NOSE DOWN ATTITUDE. THE AIRCRAFT HEADING WAS APPROXIMATELY 29° TRUE.

THIS WAS DETERMINED FROM WITNESS OBSERVATIONS, ANALYSIS OF FLIGHT INSTRUMENTS AND THE AIRCRAFT STRUCTURAL DAMAGE PATTERN.

USING AN AVERAGED GROUNDSPEED OF 76 KNOTS THE TIME FROM THE PLOTTED POSITION OF THE 35mm SLIDE IS 11.2 SECONDS.

THE TIME FROM THE PLOTTED POSITION OF THE NOISE TO IMPACT IS 8.2 SECONDS.

4-SEPTEMBER-1979

A		ORIGINAL ISSUE			

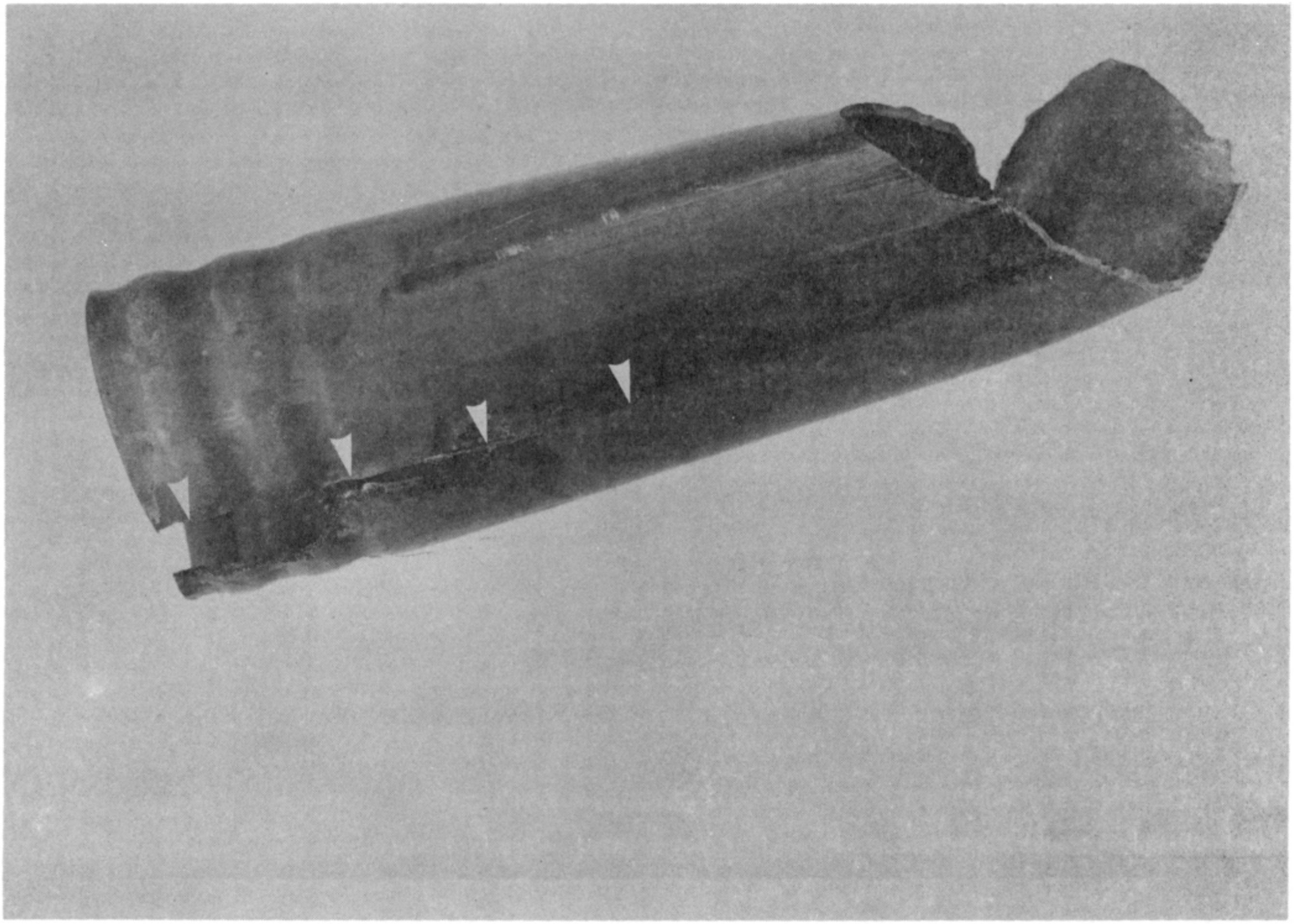
AVIATION SAFETY–INVESTIGATION

FLIGHT PATH PROFILE
DHC-6 DE HAVILLAND C-FAIV
AIRCRAFT ACCIDENT VANCOUVER BC 2 SEPT. 1978

4-SEPT-79	E.W. [illegible]	
AS SHOWN		ADS002HB003

FIG. 2

DHC 6 FLAP ROD - P/N C6CW 1029-1



CLOSE-UP PHOTOGRAPH OF THE INBOARD TUBE END OF THE BELL CRANK TO INBOARD ROD, FLAP ROD ASSEMBLY. SHOWN BY ARROWS IS THE 2-3/4 INCH STRESS CORROSION CRACK.

<u>DHC 6 Flap Rod - Magniform Connection - Tensile Tests</u>

The manufacturer carried out a series of tensile tests of the load required to break a magniform connection with cracks artificially cut to specific lengths. Three of these, samples 2, 9 and 5 are shown below with tensile-strength values of 12150 lbs., 1221 lbs. and 500 lbs. Those that were cracked to the free end, failed at low values; such was the case with the CF-AIV rod.

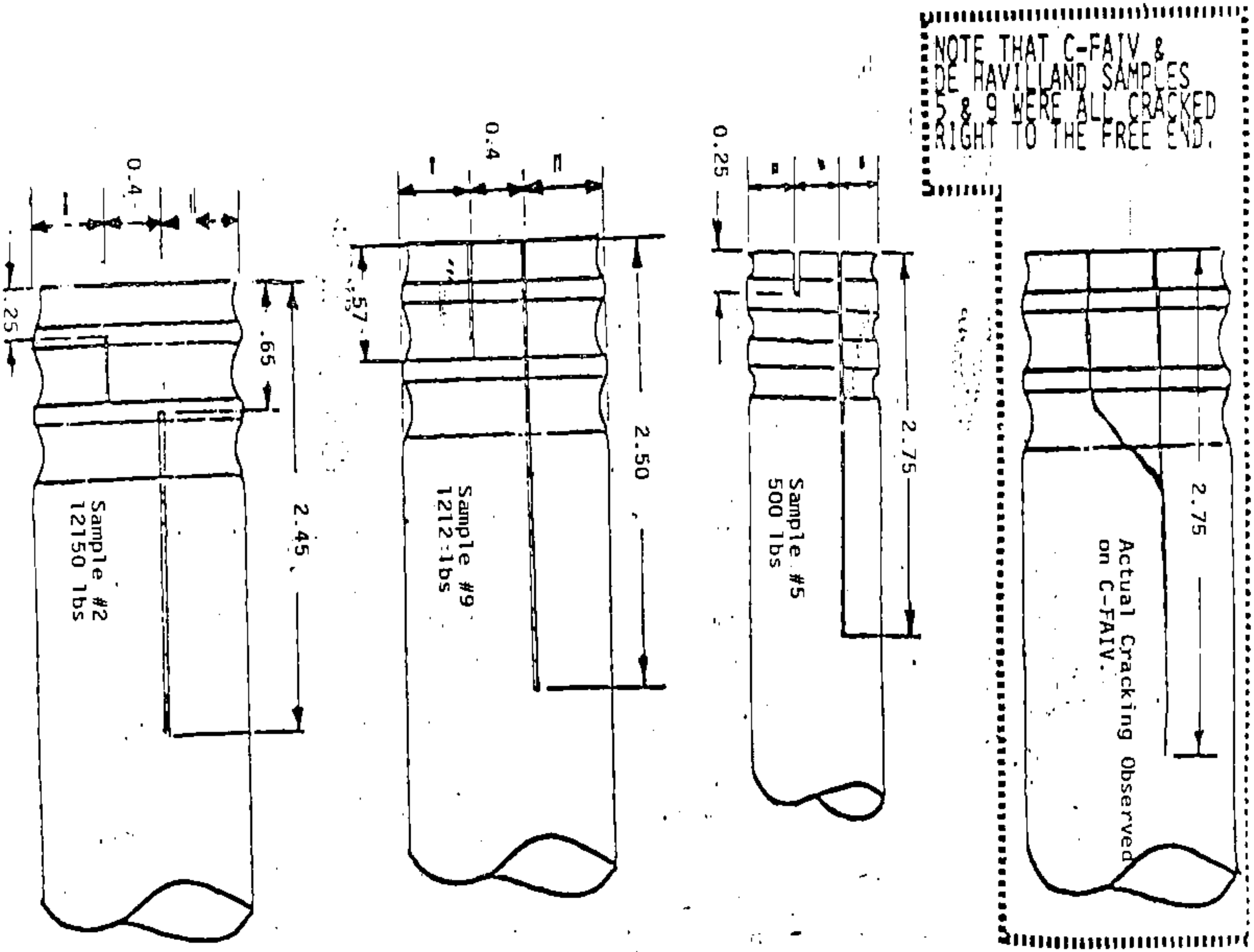


Fig. 3

SECTION 12

PROPELLER REVERSING INSTALLATION (PT6A–6B & PT6A–20)

GENERAL

1 The PT6A-6B and PT6A-20 reversing propeller installation utilizes a single-acting hydraulic reversing propeller controlled by a propeller governor. An optional propeller overspeed governor may be provided to limit power turbine speed in the event of failure of the propeller governor. A power turbine governor controls propeller speed in the Beta range and will also limit propeller speed in the remote event of the propeller overspeeding.

2 The integrated engine and propeller combination has three cockpit controls: a fuel cut-off lever, a power control lever and a propeller control lever. The fuel cut-off lever has two positions: CUT-OFF and IDLE. An additional HI-IDLE position is provided on certain installations. This lever controls the cut-off function of the fuel control unit (FCU) and resets the power control lever Idle Stop to provide 50% minimum N_g in the IDLE position and up to a minimum of 80% N_g in the HI-IDLE position, where applicable. The N_g selected by the power control lever is a maximum value which is reached only if the power turbine governor does not act to reduce N_g below the selected value. The power control lever also selects low blade angle (Beta) and resets the power turbine governor from over propeller selected speed at full forward power to under propeller selected speed in the Beta range. The propeller control lever is connected to the propeller governor through linkage. Movement of the linkage determines and limits the rpm range of the propeller governor and so governs the propeller. When the control lever is moved to the minimum rpm position, the propeller will automatically feather.

PROPELLER

3 The engine is normally equipped with a three bladed metal propeller dowelled and bolted to the front face of the propeller shaft flange. The propeller consists of a hollow steel spider hub which supports three propeller blades and also houses an internal oil pilot tube and the feather return springs (see Figure 1-12-1). Movement of the propeller blades is controlled by a hydraulic servo piston mounted on the front of the propeller spider hub. The servo piston is connected by a link to each of the blade trailing edge roots. Centrifugal counterweights on each blade and feathering springs in the servo piston tend to drive the servo piston into the feather, or high pitch position. This movement is opposed by the oil pressure generated and controlled by the propeller governor. The pressure oil is transferred to the servo piston via an oil transfer housing and the hollow center of the propeller shaft. An increase in the oil pressure moves the propeller blades toward the low pitch position (increased rpm). A decrease in the oil pressure allows the blades to move toward the high pitch position (decreased rpm) under the influence of the feathering springs and blade counterweights.

4 The servo piston is also connected by three spring-loaded sliding rods to a feedback ring mounted behind the propeller. Movement of the feedback ring is transmitted by carbon blocks through a propeller reversing lever and linkages to the propeller governor override rod. The lever functions as a link between the push-pull wire rope and the override rod in the propeller governor and also functions as a feedback link in the reverse pitch and low pitch selections.

PROPELLER GOVERNOR

5 The propeller governor performs two functions. Under normal flight conditions it acts as a constant speed governing unit (CSU) to regulate power turbine speed (N_f) by varying the propeller blade pitch to match the load torque to the engine torque in response to changing flight conditions. During low airspeed operations, the propeller governor can be used to select the required blade angle (Beta control): while in the Beta control range, engine power is adjusted by the FCU and the power turbine governor to maintain power turbine speed (N_f) at a speed slightly less than the selected rpm.

6 The propeller governor consists of a single-acting centrifugal governor mounted at the 12 o'clock position on the reduction gearbox front case (see Figure 1-12-2). The propeller governor function in forward thrust is to boost engine oil pressure to decrease propeller blade pitch, while the centrifugal force of the blade counterweights, assisted by feathering springs, tend to increase the pitch.

7 The propeller governor includes an integral gear type oil pump with a pressure relief valve, a pair of pivoted flyweights mounted on a rotating flyweight head, a spring-loaded pilot valve and the necessary cored oil passages contained in the governor housing and base. The rotating flyweights determine the position of the pilot valve while the spring load can be varied by an externally mounted propeller control lever.

8 In the event of a propeller control linkage failure, a spring attached to the propeller speed adjusting lever holds the lever in its last selected position or moves it toward the high speed setting. The high speed stop prevents the lever moving beyond the 100% position and enables the engine to be operated at or near full rated speed and develop maximum power. Moving the propeller speed adjusting lever against the low speed stop raises the pilot valve to a positive coarse pitch or feathering position

regardless of flyweight force. This enables the pilot to feather the propeller and minimize propeller drag in the event of engine failure.

9 To provide the propeller governor with a sensing element, the rotating pivoted flyweights are linked mechanically to the engine by a hollow drive gearshaft. The rotating flyweights, actuated by centrifugal force, position the pilot valve so as to cover or uncover oil ports in the drive gearshaft and regulate the oil flow to and from the propeller servo piston. The centrifugal force exerted by the flyweights is opposed by the force of the adjustable speeder spring. This determines the engine rpm required to develop sufficient centrifugal force on the flyweights to center the pilot valve, thereby preventing oil flow.

10 An override rod is located inside the hollow center of the propeller governor pilot valve and is connected through a linkage to the propeller reversing lever. The rod is ineffective when the engine is in the constant speed range as the rotating flyweights hold the pilot valve clear of the rod. However, when the propeller is running at a speed lower than that selected on the propeller control lever, the flyweights allow the pilot valve to drop, causing the blade angle to decrease. As the propeller blades approach the angle selected by the power control lever, the feedback ring repositions the propeller reversing lever to move the pilot valve, through the override rod, to an on-speed or null position and so prevent further blade movement.

PART I - Procedure

CHAPTER I - Definitions

1.1 Design Approval Representative

1.1.1 A Design Approval Representative is one who, by reason of his training and experience in aeronautics, is capable of professional responsibility for the drawings, calculations, test and engineering reports necessary to substantiate the airworthiness of a design, a repair or modification and who has been approved by the Department. Normally such approval will only be recognized during the period of the D.A.R.'s employment with the particular company, and for that company's work. The Department may also recognize the professional standing of an engineer registered as such in the province in which he is practising and may grant approval as a Design Approval Representative. Departmental approval, however, does not imply automatic qualification of membership in any recognized body of professional engineers. It is only the recognition of certain qualities which would enable the engineer to act on behalf of the Department for the approval of designs, modifications and repairs, as outlined in Para. 2.1.5 or 2.5.2 as applicable.

CF-AIV TWIN OTTER ENGINE & PROPELLER EXAMINATION

Final Examination

The engines, complete with attached propellers, were salvaged from the Vancouver Harbour accident site, flushed with fresh water and transported to a quarantined area in the South Vancouver Terminal area. There they were examined externally and photographed.

The propellers were removed from their respective engines in the approved manner. At this time oil samples were taken from the propeller shafts of each engine for analysis (LP 281/78 refers). The left propeller piston was found, along with the #2 blade, in the zero thrust position by a zero thrust latch. After documentation the blade latch was released, allowing the piston to move to the feathered position.

Engine oil filters and magnetic chip plugs were examined on both engines. Except for corrosion products they were clean. Oil and fuel was in evidence in lines and filters on both engines. A visual examination of the first stage compressor and power turbine blades revealed no apparent foreign object damage or bending. Three inlet housing webs of the right engine were cracked. Minor torsional buckling of the left and moderate torsional buckling of right engine exhaust ducts was evident. There was no evidence of engine inflight fire. Electrical wiring, mechnical linkages, pneumatic and other plumbing connections for both engines had been pulled apart as the engines separated from their respective wings. Control linkage failures were the result of overload forces.

Engine Teardown

Both engines, Serial Number PCE 20786 LH and PCE 21873 RH, were dismantled in the presence of Transport Canada and company representatives.

The compressor and power sections of both engines were rotating at the time of impact. There was no evidence of pre-impact distress or imminent failure found within the engines which would have prevented either from developing power as required.

Surge bleed valves for both engines were tested and found to function normally. Subsequently the propeller governors, and overspeed governors were either tested or dismantled to determine their operational status. No abnormalities were found.

Propeller Teardown

Both propellers, S/N BU577 and BU 1492 left and right installations respectively, were dismantled in the presence of Transport Canada and company representatives. No pre-impact failures were found. Blades of the right propeller exhibited positive (forward) bending. Two blades of the left propeller were bent aft, the third had a slight mid-span forward bend plus an aft bend about 10" from the tip.

CONCLUSION

Both engines were operating at the time of impact and both propellers were capable of normal operation at the time of impact.

CF-AIV TWIN OTTER ANALYSIS OF SEQUENCE OF BREAK-UP

The left wing tip leading edge was massively damaged by water impact rotating the wing rearward about the root attachment.

The left wing rear spar attachment failed in compression. This bowed the spar cap forward and failed it in bending and compression just outboard of the lug which remained fixed on the fuselage fitting. Buckling failure of the left wing rear spar allowed the left wing to swing back into the fuselage side, pivoting about the front spar attachment until the front fuselage lug was fractured by bending overload just inboard of the attachment bolt hole. The inboard ends of the forward and trailing flaps for the left wing were therefore pushed up against the side of the fuselage as the fuselage former cut into the wing skin panels in front of the rear spar. This caused flap cross-section outline marks to be made on the left side of the fuselage skin which were consistent with the <u>left flaps being fully up at impact</u>.

All deformation, fractures and other witness marks on the left flap hinges and actuators were consistent with <u>left flaps being fully up</u> when flaps were end-loaded by fuselage contact.

Because of the deceleration of the aircraft, the right wing, which did not immediately hit the water, was thrown forward, failing the rear spar wing attachment in tension. The right wing flaps were found to be seized in the almost fully down position. No marks were made on the right side of the fuselage by the flaps because they moved away from the fuselage during impact.

During the investigation, at the instigation of the operator, the end of the left flap push-pull tube, which was at first missing, was found by their diver in the water at the crash site. It was about 5.7 inches long with the rod end lug fitting missing and it was cracked along its length where the internal lug fitting had been.

It was determined by extensive metallurgical analysis of this cracked tube and other tubes in which cracks were found in the magneformed end attachment to the internal lug fitting, that stress corrosion cracking (SCC) had occurred in the magneformed end and this had reduced the tensile strength.

Tests performed by Transport Canada to simulate the effects on tensile separation load of a longitudinal crack through the swaging gave:

2400 lb.	for	1.5"	crack
1680 lb.	for	2.75"	crack
610 lb.	for	2.5"	crack
510 lb.	for	2.25"	crack

Variations in load versus crack length are due to variations in original clamping action (i.e. in material strength, fit, residual stresses etc.).

The rod was designed to withstand an 8000 lb. test load and to withstand 330 - 650 lb. working loads. One test of a typical rod end gave tensile strength of 9,250 lb. for an uncracked rod. The manufacturer's tests showed a normal breaking load of 11,000 lbs.

The material of the 1.50 inch O.D. x .049 inch tube was 2024-T3 aluminum alloy (Spec. WW-T0785,T3). The magneform swaging on the end left residual hoop tension stress in the circumferential direction. This, in combination with a particular corrosive environment, provided the essential ingredients which were conducive to SCC. It was considered entirely consistent with the investigation findings that the push-rod failure had occurred when the push-rod load reached the reduced strength of the cracked tube-end fitting connection as the tension load increased with increasing flap deflection.

This resulted in the rod moving back almost completely into the wing allowing flaps to come fully up. (Note: The internal plug with its end lug and a small portion of the inboard tube from inside the fuselage roof were never recovered.) The short inboard end of the larger tube, without the internal plug, separated about 5 inches from the end by bending and pinching loads as it protruded from the face of the wing root rib. It was possible to match the fractured end face with the tube end inside the wing. This position of the push rod is consistent with an intermediate flap position. As the flaps were pushed outboard by interference with the fuselage side, this rod was being pushed inwards into the fuselage and into the shroud. Overload bending failure of the tube occurred when the wing rotated aft pinching off the end which was projecting through the fuselage. A dent was caused by the tube at the front lower edge of the hole in the fuselage side. A hole was punched out of the protective shroud by the empty flap tube end when it contacted the inside of the shroud as the wing root moved forward. This shows that the fitting had already pulled out of the end of the flap tube. If the flaps had been down, the tube would have projected farther out of the wing and the pinched off piece of tube would have been about the length of the one on the right side and the metal shroud would have been split along its length.

CONCLUSION

1. The end of the left flap actuator tube failed in flight when the aircraft was on approach with flaps nearly down. When the inboard tube-end failed it allowed the complete left flap to blow up. This was followed by a loss of control and crash into the water with the nose down and with the left wing down at a high roll angle thereby giving first contact with the water on the left wing tip.

2. (a) The pre-impact flap configuration was flaps up on left wing and flaps down on right wing.

(b) The probable sequence of break-up was: first water contact was with left wing tip; left wing rotated aft; aircraft skidded to the right; left float was pushed to left, up and aft; front fuselage pushed to left, up and aft; right float pushed aft and right wing rotated forward as aircraft cartwheeled right.

DEPARTMENT OF TRANSPORT
AVIATION SAFETY ENGINEERING LABORATORY
LABORATORY TECHNICAL INVESTIGATION REPORT

MINISTÈRE DES TRANSPORTS
LABORATOIRE TECHNIQUE DE LA SÉCURITÉ AÉRONAUTIQUE
RAPPORT DE L'ANALYSE LABORATOIRE

ACCIDENT REPORT NO. / *NO. DE RAPPORT D'ACCIDENT* 5002-H80003	CASE PROJECT NO. / *NO. DE PROJET DE CASE* LP 23/80
SUBMITTED BY – *SOUMIS PAR* ASI	APPROVED – *APPROUVÉ* ASE

This report is submitted in response to a Request for Technical Services. It consists of a factual list of laboratory laboratory findings which relate to the parts received only.

Ce rapport est soumis en réponse à une demande de service technique. Il consiste en une énumération des constatations objectives du laboratoire le portant que sur les pièces reçues.

REQUIREMENT(S) ON PARTS RECEIVED – *NATURE DES DEMANDES SUR LES PIÈCES REÇUES*

To examine the longitudinal crack section through the swage in the CF-AIV push-pull flap control rod. The crack was removed, ultrasonically cleaned and mounted for scanning electron microscopic (SEM) analysis. The work was carried out by a member of the Department of Metallurgy, University of British Columbia, under the direction of ASE (DOT).

TECHNICAL FINDINGS – *CONSTATATIONS TECHNIQUES*

The longitudinal crack was flat, brittle and transverse in nature throughout its length except for a small lip at the free end of the tube.

The SEM disclosed a classic intergranular stress corrosion crack (SCC) along almost the whole length of crack examined.

At the freee end of the tube, a small zone of ductile dimpled topography intermixed with some inter granular areas was found.

The lip was a small overload crack which was approximately 0.4 mm at its maximum length.

The premature SCC was due to a susceptible material 2024-T3, a specific environment (chemically reactive marine), and the presence of high sustained residual hoop stresses.

The tiny overload zone occurred when SCC propagation left material insufficient in area to withstand the residual hoop stresses.

ASI
Ottawa, Ontario

21 January 1980

23-0041 (9-78)

CF-AIV TWIN OTTER - METTALLURGICAL ANALYSIS

Initial separation of the rod-end from the main rod assembly occurred at the time of the accident. The part, however, was not recovered from the site of the accident when the fuselage was removed but remained on the sea bed for almost five months where it was discovered during further recovery operations.

Examination of the rod-end was conducted at ASE for one day with the operator's consultant in attendance, and then independently thereafter.

As received, the flap rod-end comprised a 5.7 inch length of aluminum alloy tubing detached from its end-cap fitting and containing the magneformed swaged rod-end.

The fractures present in the tube at the end of initial examination consisted of:

i) a primary crack, extending longitudinally 2.75 inches from the end of the tube through the length of the swaged region into the body of the tube.

ii) a secondary longitudinal crack of 0.7 inches length located 0.5 inches circumferentially from the primary crack which joined up with the main fracture via an irregular fracture surface.

iii) numerous very small longitudinal cracks, mainly concentrated in the area between the two main longitudinal cracks.

In the original configuration one surface of the primary longitudinal crack overlaps the other causing the tube to appear ovalised. The small triangular shaped piece between the primary and secondary fractures was detached at the time of submission and a further fragmentation of this piece occurred during the course of the investigation.

Crack surfaces examined in detail included:

i) a one inch length of the surface from the end of the primary longitudinal crack (this section not prepared in ASE).

ii) one of the short longitudinal cracks remote from the main fracture.

The above surfaces were compared chemically and fractographically in the scanning electron microscope with a control sample from a rod of identical type and utilization containing a known stress corrosion crack.

All longitudinal cracks in the rod end exhibited the same brittle intergranular mode of failure as the known stress corrosion crack. Long term immersion in sea water was <u>not</u> sufficient to hide the salient features of the original failure.

Chemical analysis of the crack surfaces showed the same chemistry, namely a high concentration of chlorine and sulphur, as for the control sample. Both elements are present in sea water and sulphur is also in the engine exhaust gases.

The three criteria essential for the occurrence of stress corrosion cracking are satisfied in respect of the flap control rod end, being respectively:

i) the presence in the swaged end of the rod of high sustained residual tensile (hoop) stresses as the result of the magneforming process.

ii) exposure to a chemically reactive marine environment.

iii) material in a metallurgical condition of known susceptibility to stress corrosion cracking.

In the original manufacture of the various push-pull contol rods in the Twin Otter, a 2024 aluminum - copper based alloy was used in the T3 temper, which is solution heat-treated and naturally aged. The literature notes the susceptibility of the alloy in this condition to stress corrosion cracking.* In reflecting the current state of metallurgical testing and research it recommends the T81 or variant heat treatment, being a solution heat treatment, cold working and artifical aging process, which confers greater resistance to stress corrosion cracking. Such a rationale has been applied by the manufacturer through the temper change T3 to T81 in Modification 6/1487, for all replacement control rods.

* An appropriate reference to the above would be "Aluminum", Vol. 1, Properties, Physical Metallurgy and Phase Diagrams, American Society for Metals, Metals Park, Ohio 1967, pp 237-239.

The failure sequence of the rod was determined to involve:

i) the initiation and growth of one or more stress corrosion cracks in the swaged portion of the rod end.

ii) crack extension to the point where separation of the rod the end cap fitting was able to occur at normal operation flight loads.

APPENDIX G

RESULTS OF SUPPLEMENTARY ANALYSIS OF OBLIQUE SUPER-8MM COLOUR FILM

FROM CRASH SITE OF TWIN-OTTER C-FAIV, VANCOUVER HARBOUR, SEPT. 1978

13 AUGUST 1979

Frame No.	Elapsed Time (sec)	Ratio of change	"f" at camera	"f" at 20X print	% Change	Eastings (metres)	Northings (metres)	Height (ft.)
1	0.0	1:1	9.0mm	180.00mm	0.0	489988	460473	470
18	1.0	1:1	9.0	180.00	0.0	490023	460475	470
36	2.0	1:1	9.0	180.00	0.0	490056	460465	465
54	3.0	1:1.002	9.02	180.42	0.23	490097	460450	456
74	4.11	1:1.002	9.02	180.42	0.23	490142	460430	435
92	5.11	1:1.007	9.06	181.26	0.70	490179	460414	402
110	6.11	1:1.047	9.42	188.43	4.68	490212	460402	390
128	7.11	1:1.115	10.03	200.66	11.47	490251	460386	379
144	7.99	1:1.119	10.07	201.50	11.94	490282	460372	369

* Only those positions and elevations below the heavy black line have changed from the 1978 report.

APPENDIX G

VALUES CALCULATED FROM BASIC DATA IN APPENDICES "A" AND "A_1"

FRAME #	TIME SECONDS	GROUND DISTANCE TRAVELLED METRES (FEET)	GROUND SPEED OF AIRCRAFT (KNOTS)	CHANGE IN AIRCRAFT ALTITUDE (FEET)	RATE OF DESCENT FEET PER MINUTE
1 to 18	1.0	35 (115)	68	-0	0
18 to 36	1.0	35 (115)	68	-5	300
36 to 54	1.0	44 (144)	86	-9	540
54 to 74	1.11	49 (161)	86	-21	1135
74 to 92	1.0	40 (131)	78	-33	1980
92 to 110	1.0	35 (115)	68	-12	720
110 to 128	1.0	42 (138)	82	-11	660
128 to 144	0.88	34 (112)	75	-10	682
TOTAL	7.99	314 (1031)	N/A	-101	N/A
AVERAGE	N/A	39 meters/sec (129 feet/sec)	76	-12.6 feet/sec	758

APPENDIX G2

INTRODUCTION

On the 2nd September 1978 a DeHavilland DHC6-200 (CFAIV) crashed into the sea at Vancouver Harbour, B.C. while on its VFR landing approach.

Immediately prior to the crash, the aircraft was photographed by a person walking through Stanley Park, Vancouver. (see Annex A)

TASKING

A letter dated 6 Apr 79 from D.L. Button, Acting Director, Aviation Safely Bureau (DOT) addressed to the Director General Intelligence and Security requested the assistance of the Canadian Forces Photographic Interpretation Centre (CFPIU).

Specifically CFPIU were asked to attempt to recover from the photography the following data:

a. the field of view dimensions of the slide at the slant range of the aircraft;

b. the aircraft slant range from the camera and;

c. the angular displacement of the main plane control surfaces;

d. any other pertinent data that CFPIU may consider relevant.

DATA PROVIDED

a. The original 35 mm transparency taken with a Pentax SP1000 fitted with a 50 mm lens (uncalibrated). See Annex A.

b. A line drawing of the plan, front and side views of the DHC6-200 (float plane) series.

c. Projected flight path of a/c. See Annex E

METHOD OF RECTIFICATION

To ascertain the mensurational data required,it became necessary, in the first instance, to solve the three point perspective triangle, that would indicate the attitude of the aircraft to the camera. Once proved, ranges and flap angles could be calculated.

The perspective triangle

The determination of the perspective triangle proved difficult due to no visual horizon showing on the photography to which the aircraft vanishing points could be related, indicating any possible pitch and roll.

With no vanishing point clearly definable, with the possible exception of the vanishing point left (the aircraft spans), the perspective triangle could not be readily solved.

> **NOTE:** Under normal circumstances, provided that the focal length is known, and a full format print is provided, two of three vanishing points must be definable.

A technique not previously attempted was used to solve the perspective triangle using the known data - the camera focal length and the principal part of the photograph.

Determination of a single vanishing point to the left (spans) and hence the perspective triangle was determined by what can only be described as a mathematically calculated hit and miss method.

While the mean direction of the vanishing line was projected graphically, an arbitrary point was selected on or near the mean line and mathematically related to the Principle Point and focal length.

Trigonometrical solutions were used to determine the two other vanishing points (VP Right and VP Up) and with all the points mathematically located, the aircraft attitude to the camera was calculated.

Using the calculated attitude,the photograph of the aircraft was rectified using a Zeiss Stereocord and DIREZ unit coupled to a preprogrammed Hewlett Packard 9825A.

Several machine runs were made using the arbitary vanishing point left, each run producing a refinement of tilt, azimuth and swing until a final selection was made for the vanishing point that produced a correct or near correct length/span/height ratio for the aircraft. The mathematical solution to the perspective triangle is shown at Annex B.

Shown at Annex C are the computer solutions using the arbitary VP_L position and the calculated VP_{Up} position with the programmed focal length and principle point of the photograph.

An Azimuth (the yaw of the fuselage to the camera negative), of 39° and Tilt (the elevation of the lens longitudinal axis) of 27° are shown at the bottom of column two.

DIMENSIONS

Using the azimuth of 39° and tilt of 27° the following dimensions were obtained:

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	ACTUAL	CALCULATED	% ERROR
Length-nose to base of fin trail edge	49.45ft	49.45ft (factor)	
O/B-O/B of control surfaces	62.25ft	62.78ft	1%
Fin height (vertically above base of fin trail edge)	12.6 ft	13.3 ft	5.5 %
Lead edge vertical fin	28.5°	29.7°	
Trail edge vertical fin	10°	12° (approx)	
O/B flap length	16.9ft	16.9ft	0%

NOTE: These readings can be found at Annex D.

From these figures the azimuth bearing (yaw) of 39° is considered correct and a tilt angle (camera elevation along the lens longitudinal axis) of 27° may be in error by approximately 2°. The tilt angle variance will have little effect on the flap angles calculated, certainly less than one degree.

Flap angles

O/B port underside 11.8°
O/B stbd underside 13.6°

Mean angle of O/B flaps 12.7° ± 1°

I/B port underside 52.9° ± 2°
I/B stbd underside - not readable

NOTE: The flap angle between upper and undersides is 17.7°

CAMERA - A/C ORIENTATION (See ANNEX D page 1)

The camera was located 244.82 feet ahead of the nose of the aircraft along its ℄ and 301.52 ft to the left giving a horizontal range from camera position to nose of the aircraft of 388 ft. The aircraft nose was between 197.87' and 187' (197.87 - 5.5%) above the camera azimuth.

The slant range from camera to nose of aircraft was between 435.5 ft and 430.7 ft.

The camera angle of view (in the horizontal is 39.8°).

ADDITIONAL DATA

Assuming the aircraft to be on level flight, all calculations indicate a 2° roll (port wing down)

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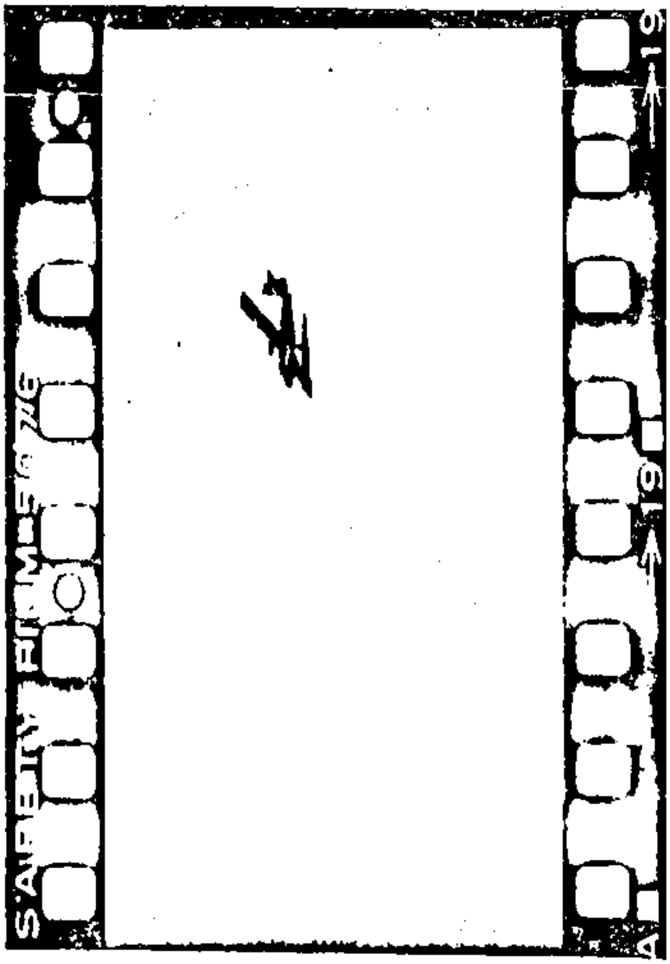
PLOTTED POSITION OF A/C (See ANNEX E)

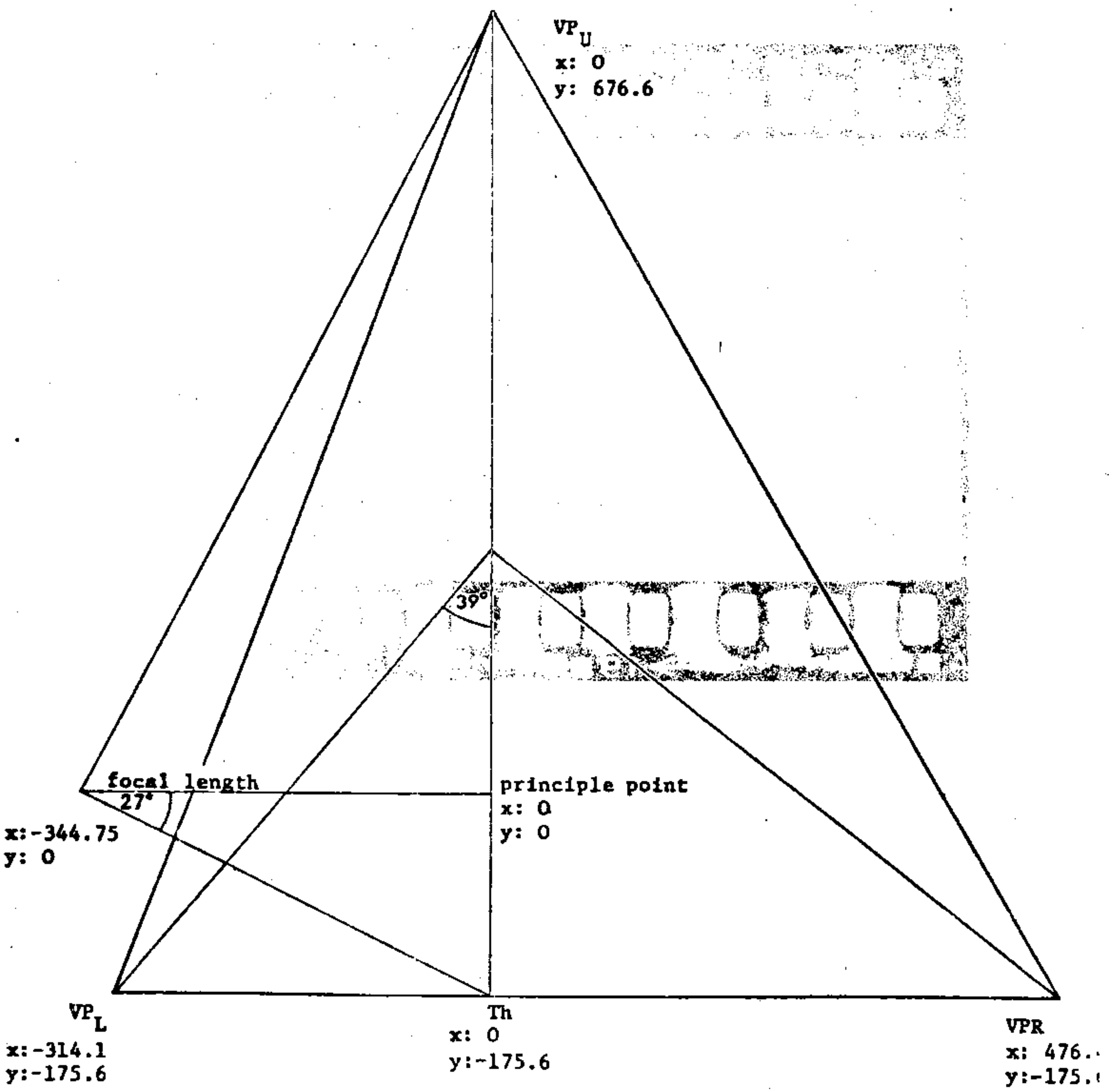
From data already supplied by Energy Mines and Resources, a projected flight path was provided. The probable camera position was also sited.

With the given camera position an arc of radius 388 ft representative of the aircraft horizontal range at the instant of photography added to the plan. CFPIU postulated optional positions of the a/c flight path are added.

H5



ANNEX B to:
CFPIU Rept 2075-6/030/79



INTRODUCTION

The report is issued as a supplement to CFPIU Report #2075-6/030/79 and should be read in conjunction with that report.

TASKING

Subsequent to the issuing of Report #2075-6/030/79 the Directorate of Aviation Safety Bureau (DOT) requested that an attempt be made to determine the perspective triangle geometry and hence range from camera to aircraft if negative pitch could be introduced into the calculation.

DISCUSSION - THE PERSPECTIVE FORM

For any given camera position and orientation of an object to the camera film plane (ie pitch, roll, azimuth to the camera horizontal and vertical plane) to which a range can be applied, a predetermined image in perspective can be derived.

Figure 1 shows these circumstances of a camera position and object ABCD in top view with 0^{o} pitch, 0^{o} roll and 30^{o} azimuth. The perspective view A_p B_p C_p D_p can be developed as shown.

The converse is equally true. The perspective image can only produce one orthographic plane form - in this instance the square ABCD.

Located to the right of ABCD is an identical square designated A_1 B_1 C_1 D_1 with the pitch and roll both at 0^{o} and an azimuth of 30^{o}. The unmoved camera position is now to the left of A_1 B_1 C_1 D_1.

Under such circumstances an entirely different perspective form develops at A_{1p} B_{1p} C_{1p} D_{1p}, for the identical square.

Studying the two perspective views it can be seen that:

(1) A_{1p} B_{1p} appears much longer than A_p B_p

(2) A_{1p} C_{1p} appears shorter than A_p C_p

(3) the three angles at A & A_1 are different.

.../3

In summary then, by moving the object, a different perspective image is produced, both in dimensional and angular form. Put another way, no two people can see an identical image of an object at any given time as they are viewing from different positions.

RESULTS ACHIEVED

As explained above, as soon as any of the parameters are changed a different perspective form develops, or as a corallary, if any of the perspective triangle parameters are changed WITHOUT changing the perspective form, then proportionality of the object must change (see Figure 2), and ABCD is no longer a square.

If a AB represents the length of an aircraft and AD a half span, it becomes apparent that while perspective triangles can be developed, to produce identical perspective images, each will have evolved from different plan views - in the case of an aircraft, different length/span ratios.

These findings in respect to the aircraft length/span ratio were somewhat unexpectedly derived when different pitch angles were applied, until the above rationales were thought out and applied to Figures 1 & 2.

STATISTICAL RESULTS

In tabular form the results obtained are listed below. The first set of figures are those quoted in CFPIU Report # 2075-6/030/79. Reworking those figures indicated a -2° pitch. This study applied two additional pitch angles of -8° and -11°. Two pitches were used solely for comparative purposes and to enable the figures derived to be plotted in graph form for visual comparison (Figure 3).

The graphic showed that the changes in azimuth, tilt, swing, range and altitude, produced straight line graphs. Consequently azimuth, tilt, range, and altitude can be derived for any pitch angle.

AIRCRAFT PITCH	AIRCRAFT AZIMUTH°	CAMERA TILT°	AIRCRAFT RANGE'	AIRCRAFT ALTITUDE'	AIRCRAFT SPAN'
-2°	39	27	388	197.87	62.78
-8°	36.55	30.55	355.67	220.03	54.94
-11°	35.18	32.18	338.75	225.93	53.34

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CONCLUSIONS

The above table shows that azimuth, tilt, range, and altitude for any given pitch angle of the aircraft. Within those factors, and an unchanged perspective image, the length/span ratio of the hypothetical aircraft will change.

The only one to satisfy the DHC6 CFA IV aircraft dimensions are those quoted in the original Report #2075-6/030/79. The aircraft cannot be located in any other position.

FIGURE ONE

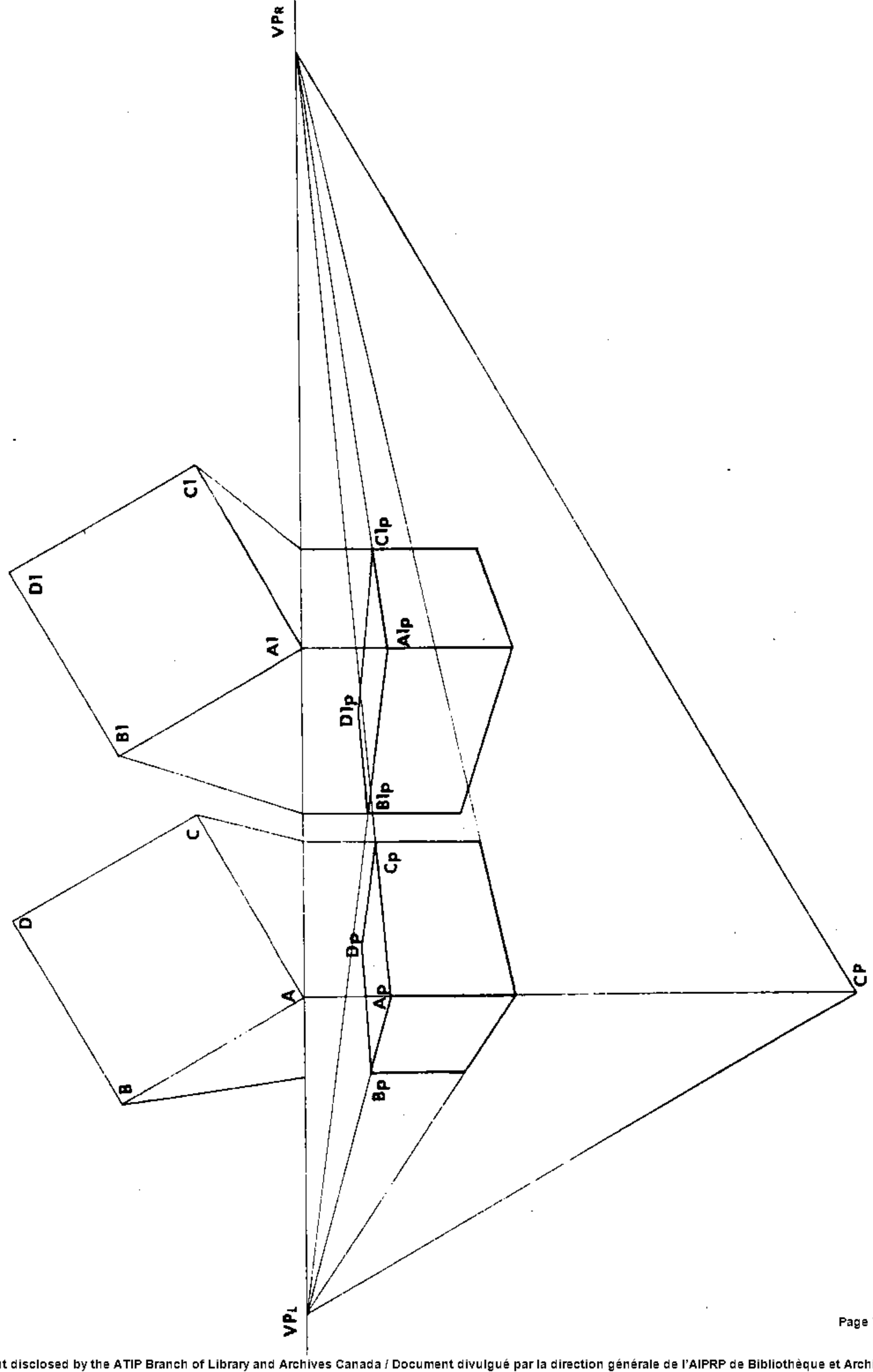


FIGURE TWO

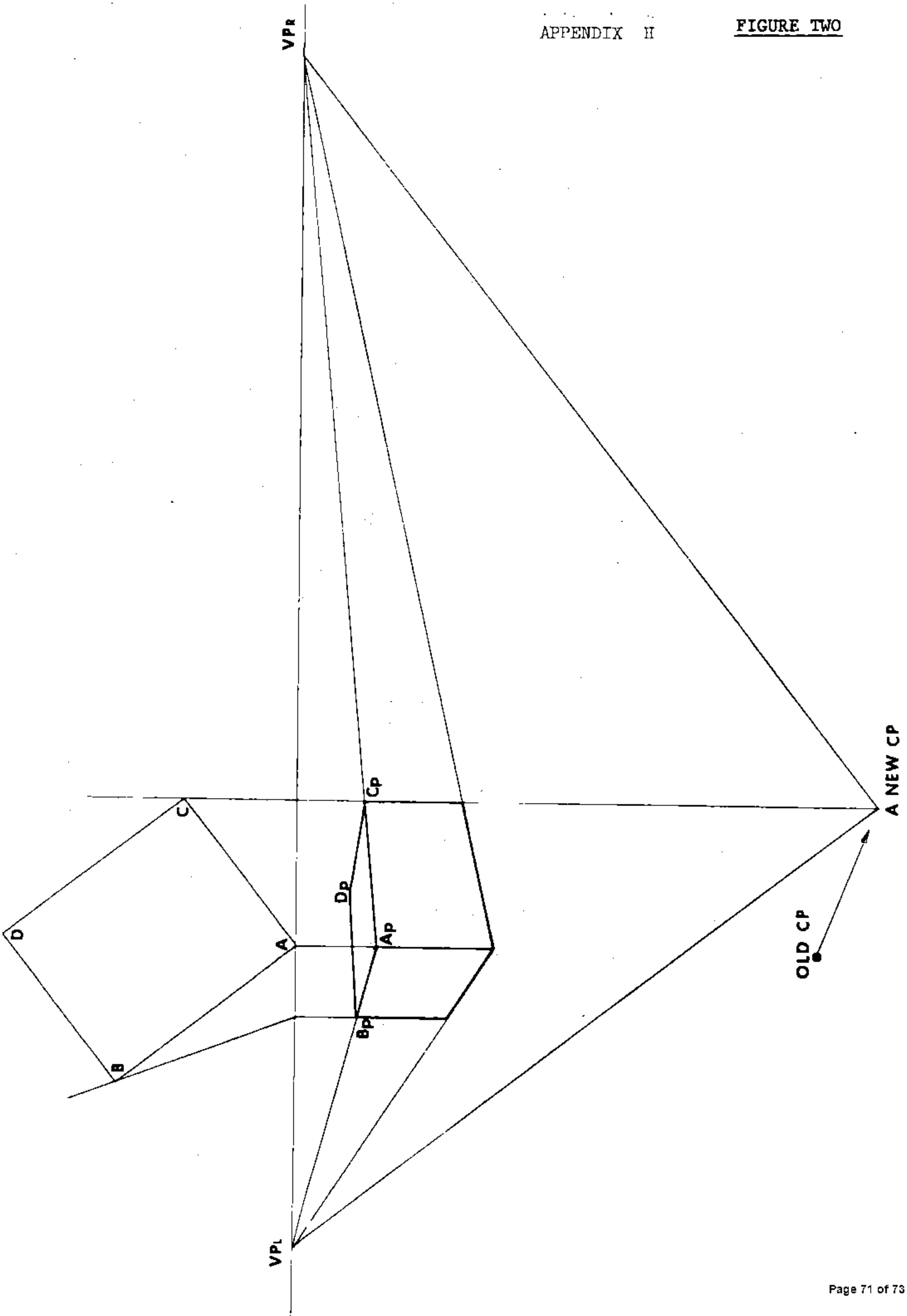
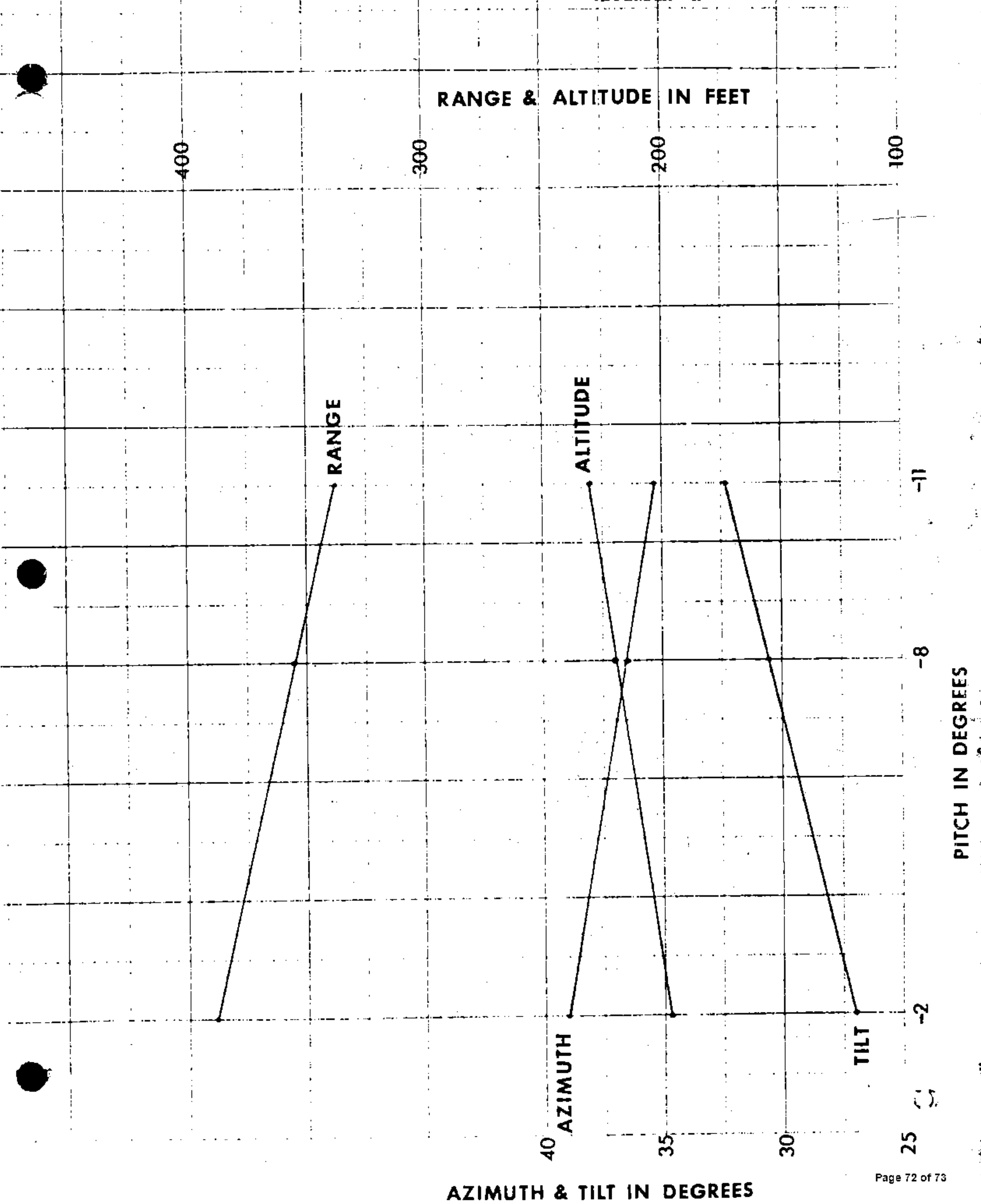
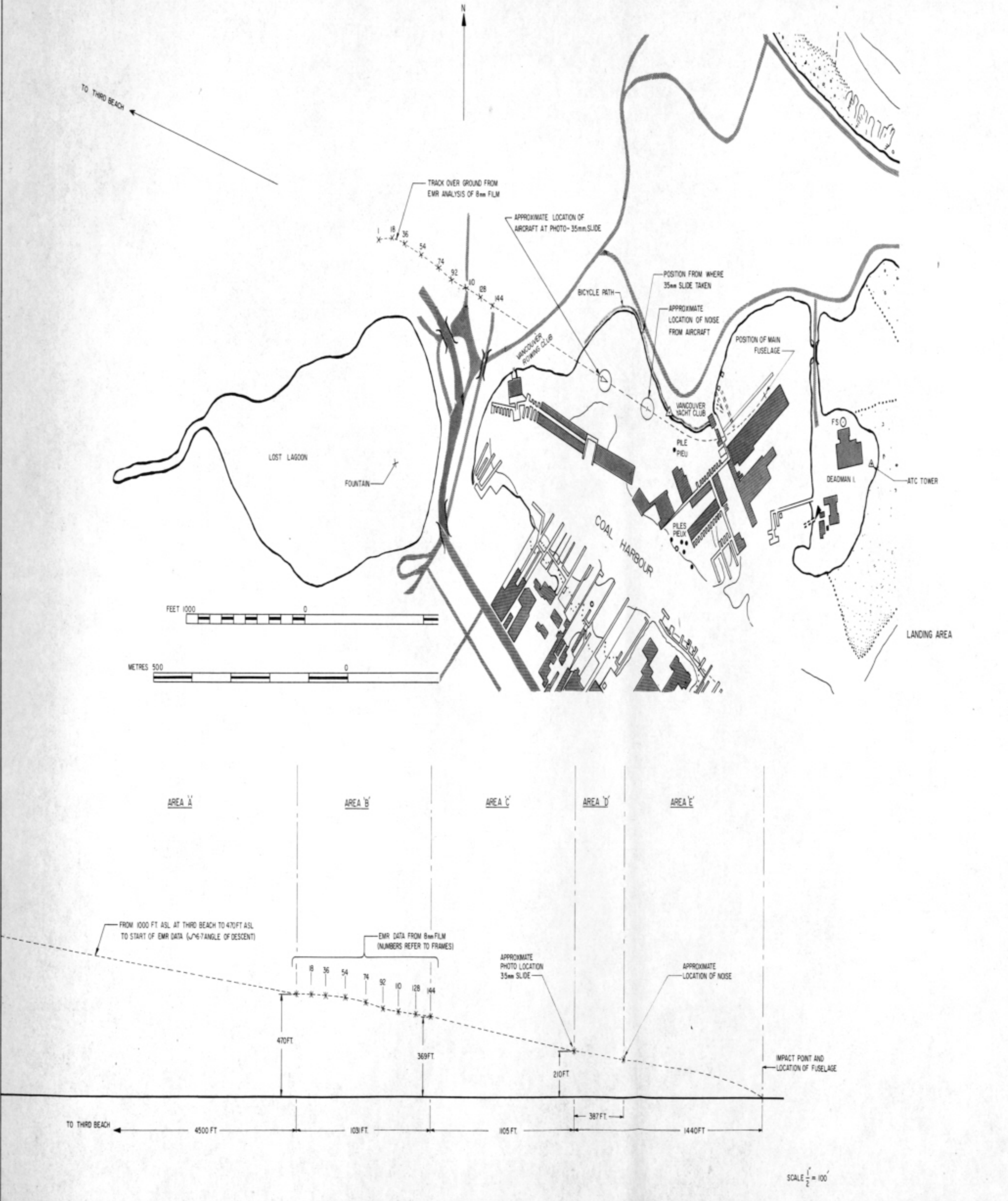


FIGURE THREE





SCALE $\frac{1}{2}$ = 100'

Transport Canada Air
Transports Canada Air

LIST OF MATERIALS
LISTE DU MATERIEL

qty quant.	item article	description désignation	part no. no pièce	reference référence

AREA 'A'

AT THIRD BEACH—LAST RADIO TRANSMISSION FROM AIRCRAFT C-FAIV. ACKNOWLEDGEMENT OF LANDING CLEARANCE 54 SECONDS PRIOR TO IMPACT AND ELT SIGNAL. ESTIMATED 1000FT. ASL IS CONSISTENT WITH A NORMAL APPROACH.

AREA 'B'

FROM AN 8mm MOVIE FILM RECOVERED FROM THE WRECKAGE, IT WAS EVIDENT THAT THE CAMERA WAS IN OPERATION THROUGHOUT A PORTION OF THE APPROACH.

THIS FILM EMBRACES A SEGMENT OF 7·99 SECONDS–FRAME 1 THROUGH 144.

AVERAGED RATE OF DESCENT = 758FT/MIN
AVERAGED GROUNDSPEED = 76 KNOTS

FLAPS WERE AT THE 20° SELECTOR POSITION AT FRAME 1 AND AT 28° SELECTOR POSITION AT FRAME 144. THIS IS EQUIVALENT TO 1° OF FLAP EXTENSION PER SECOND.

AREA 'C'

PRIOR TO THE CRASH–A PROFESSIONAL PHOTOGRAPHER LOCATED ON A BICYCLE PATH IN STANLEY PARK TOOK A PICTURE OF THE AIRCRAFT.

FROM THIS 35mm SLIDE IT WAS DETERMINED THAT THE AIRCRAFT WAS AT A HORIZONTAL DISTANCE OF 388FT AND 198FT ABOVE THE CAMERA AZIMUTH AND LOCATION WHERE PHOTO WAS TAKEN.

THE AIRCRAFT APPEARED NORMAL AND THE FLAPS WERE AT A SELECTOR POSITION OF 33°.
NOTE FULL FLAP IS $37\frac{1}{2}°$

TIME FROM FRAME 144 IS 8·6 SECONDS.
AVERAGE GROUNDSPEED = 76KNOTS.

THE DISTANCE FROM THE PLOTTED POSITION OF THE 35mm SLIDE TO THE USUAL TOUCHDOWN AREA IS APPROXIMATELY 2500FT.

AREA 'D'

A NOISE EMANATED FROM THE AIRCRAFT APPROXIMATELY 3 SECONDS AFTER THE PHOTO WAS TAKEN.

USING A GROUNDSPEED OF 76 KNOTS PLACES THE AIRCRAFT 387FT BEYOND THE LOCATION OF THE 35mm SLIDE. OBSERVATIONS OF GROUND WITNESSES SUBSTANTIATE THIS LOCATION.

FROM THIS POINT A LEFT BANKED TURN AND DESCENT BEGAN AND CONTINUED TO IMPACT.

AREA 'E'

POINT OF IMPACT, LOCATION OF MAIN FUSELAGE AFTER THE CRASH. THE AIRCRAFT ENTERED THE WATER IN A LEFT WING LOW AND NOSE DOWN ATTITUDE. THE AIRCRAFT HEADING WAS APPROXIMATELY 25° TRUE.

THIS WAS DETERMINED FROM WITNESS OBSERVATIONS, ANALYSIS OF FLIGHT INSTRUMENTS AND THE AIRCRAFT STRUCTURAL DAMAGE PATTERN.

USING AN AVERAGED GROUNDSPEED OF 76 KNOTS THE TIME FROM THE PLOTTED POSITION OF THE 35mm SLIDE IS 11·2 SECONDS.

THE TIME FROM THE PLOTTED POSITION OF THE NOISE TO IMPACT IS 8·2 SECONDS.

4-SEPTEMBER-1979

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A		ORIGINAL ISSUE			

AVIATION SAFETY-INVESTIGATION

FLIGHT PATH PROFILE
DHC-6 dE HAVILLAND C-FAIV
AIRCRAFT ACCIDENT VANCOUVER BC 2 SEPT. 1978

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