ARDU-TI-953

DTIC FILE COPY

AR-003-256

4

AD-A205 303

DEPARTMENT OF DEFENCE

ROYAL AUSTRALIAN AIR FORCE

AIRCRAFT RESEARCH AND DEVELOPMENT UNIT

EDINBURGH, SOUTH AUSTRALIA

TECHNICAL INVESTIGATION NO 953

FLIGHT TESTING OF THE SOUTHERN CROSS REPLICA AIRCRAFT

APPROVED

FOR PUBLIC RELEASE

COPY NO

SEPTEMBER 1988

ACTION AUTHORITY

This report is for the communication of information and is not an authority. Any action required within the RAAF will be authorised by Headquarters Support Command.

Security classification of this page UNCLASSIFIED

DOCUMENT CONTROL DATA SHEET

1 DOCUMENT NUMBERS	
AR Number:	AR-003-256
Report Number:	ARDU-TI-953
Other Numbers:	

2 SECURITY CLASSIFICATION	
a. Complete Document:	UNCLASSIFIED
b. Title in Isolation:	UNCLASSIFIED
c. Summary in Isolation:	UNCLASSIFIED
d. Privacy Marking:	

3 TITLE

FLIGHT TESTING OF THE SOUTHERN CROSS REPLICA AIRCRAFT

4 PERSONAL AUTHOR(S):

SQNLDR N.G. COULSON

5 RECLASSIFICATION AND CONTROL REVIEW AUTHORITY

CO ARDU

6 DOCUMENT DATE:

SEPTEMBER 1988

7 CORPORATE AUTHOR(S):

Aircraft Research and
Development Unit

8 Total Number of Pages: 92

Number of References: 6

9 COMPUTER PROGRAM(S):
(Title(s) and language(s))

10. RELEASE LIMITATIONS:

Approved for public release

Security classification of this page UNCLASSIFIED

Security classification of this page UNCLASSIFIED

11. **ANNOUNCEMENT LIMITATIONS** (of the information on these pages):

Announcement of this report is unlimited

12. **DESCRIPTORS:**

a. Thesaurus Terms: Flight testing

b. Non-Thesaurus Terms: Southern Cross
Fokker

13. DRDA Cat: ØØ51C

14. **SUMMARY OR ABSTRACT:**

(if this is security classified, the announcement of this report will be similarly classified)

Aircraft Research and Development Unit was tasked to carry out the test flying of a replica of the 1926 Fokker Tri-Motor as flown by Australian aviation pioneer, Sir Charles Kingsford-Smith. The purpose of the test programme was, firstly, to ensure safe operation of the aircraft throughout its proposed flight envelope and, secondly, to provide data to allow the issue of a Certificate of Airworthiness or Permit to Fly. The trial included a cockpit and systems assessment as well as an evaluation of the aircraft's flight and ground handling characteristics. Airborne assessments covered stability and control characteristics, stall characteristics, general aircraft performance, asymmetric power characteristics and an evaluation of the aircraft's take-off and landing performance and handling.

The flight characteristics of the test aircraft were found to be similar to those expected from an original Fokker VIIb-3M. Consequently, the aircraft could not meet some modern certification requirements. Notwithstanding this the aircraft was found to be generally safe and airworthy provided it was operated by experienced pilots in daylight Visual Meteorological Conditions and that the main recommendations of this report are adopted.

The major recommendations from the trial include the application of a five knot crosswind limit when the aircraft is to be operated from short, narrow runways; the requirement for cockpit occupants to wear helmets when operating the aircraft; a requirement for two pilots to crew the aircraft; a limitation on the aircraft's centre of gravity range; and the requirement for the incorporation of some form of mechanical stop to limit the amount of available tailplane adjustment. A further twelve recommendations were made.

Security classification of this page UNCLASSIFIED

AIRCRAFT RESEARCH AND DEVELOPMENT UNIT

FLIGHT TESTING OF THE SOUTHERN CROSS REPLICA AIRCRAFT

Project Officer:

(N.G. COULSON)
Squadron Leader

Approved By:

(R.A. HOWARD)
Wing Commander
OIC Flight Test Squadron

(K.J. BRICKNELL)
Group Captain
Commanding Officer

ARDU File Reference: 2654/953/Tech

HQSC File Reference: AIR4/4082/03/953 Pt1 (6)

DEPARTMENT OF DEFENCE

ROYAL AUSTRALIAN AIR FORCE

AIRCRAFT RESEARCH AND DEVELOPMENT UNIT

TECHNICAL INVESTIGATION NO 953

FLIGHT TESTING OF THE SOUTHERN CROSS REPLICA AIRCRAFT

SUMMARY

Aircraft Research and Development Unit was tasked to carry out the test flying of a replica of the 1926 Fokker Tri-Motor as flown by Australian aviation pioneer, Sir Charles Kingsford-Smith. The purpose of the test programme was, firstly, to ensure safe operation of the aircraft throughout its proposed flight envelope and, secondly, to provide data to allow the issue of a Certificate of Airworthiness or Permit to Fly. The trial included a cockpit and systems assessment as well as an evaluation of the aircraft's flight and ground handling characteristics. Airborne assessments covered stability and control characteristics, stall characteristics, general aircraft performance, asymmetric power characteristics and an evaluation of the aircraft's take-off and landing performance and handling.

The flight characteristics of the test aircraft were found to be similar to those expected from an original Fokker VIIb-3M. Consequently, the aircraft could not meet some modern certification requirements. Notwithstanding this the aircraft was found to be generally safe and airworthy provided it was operated by experienced pilots in daylight Visual Meteorological Conditions and that the main recommendations of this report are adopted.

The major recommendations from the trial include the application of a five knot crosswind limit when the aircraft is to be operated from short, narrow runways; the requirement for cockpit occupants to wear helmets when operating the aircraft; a requirement for two pilots to crew the aircraft; a limitation on the aircraft's centre of gravity range; and the requirement for the incorporation of some form of mechanical stop to limit the amount of available tailplane adjustment. A further twelve recommendations were made.

CONTENTS

TABLES

ANNEXES

FLIGHT TESTING OF THE SOUTHERN CROSS REPLICA AIRCRAFT

1. INTRODUCTION

1.1 Background. A group known as the Southern Cross Museum Trust sponsored the development of a flying replica of the 1926 Fokker Tri-Motor as flown by Australian aviation pioneer Sir Charles Kingsford-Smith. The new 'Southern Cross' is a full-size reproduction of the original aircraft and is presently the largest flying replica in the world. The aircraft will be used for the promotion of Australian aviation history and in fund-raising campaigns for charity. The project has had a long, troubled development and is now Commonwealth Government funded and owned. Subsequently the RAAF, and specifically Aircraft Research and Development Unit (ARDU), was tasked to carry out the test flying of the aircraft.

1.2 Task. Reference A tasked ARDU to provide assistance to the Southern Cross Museum Trust by carrying out the ground evaluation and flight testing required at Reference B to allow the Southern Cross Replica aircraft to be awarded a Department of Aviation (DoA) Special Category Certificate of Airworthiness. The evaluation was to include assessments of:

a. the cockpit and systems;

b. the aircraft's ground handling characteristics;

c. longitudinal, lateral and directional stability, control, handling and trimmability;

d. aircraft performance;

e. the aircraft's low-speed and stall characteristics; and

f. the aircraft's asymmetric power performance and handling characteristics.

2. DESCRIPTION OF TEST AIRCRAFT

2.1 Fokker Tri-Motor Replica. The aircraft was a full-scale replica of a 1926 Fokker Tri-Motor (designation F.VIIb-3M). It was built by Famous Australian Aircraft Pty Ltd at Parafield in South Australia. The basic dimensions and form of the original aircraft were followed although the engineering was to the standard required by Reference B. The aircraft was a high-wing light-transport aeroplane powered by three Jacobs R-755A1 seven cylinder radial engines; one mounted in the nose and the other two under the inboard wing sections. The maximum all-up-weight (AUW) of the aircraft was 5700 kg. The aircraft alighted on a fixed, tailwheel configuration landing gear and had conventional mechanical flight controls comprising ailerons, rudder and elevator. An adjustable tailplane was provided to allow variation

in longitudinal trim. There were no wing flaps. The open cockpit had a standard dual control layout for side-by-side pilot and co-pilot. The aircraft was registered VH-USU (as was the original 'Southern Cross') and had not been flown before the commencement of the trial. A detailed description of the aircraft is provided at Annex A.

3. CONDITIONS RELEVANT TO THE TRIAL

3.1 Purpose. The purpose of the evaluation was to define the aircraft's performance and handling characteristics in order to:

a. ensure safe operation throughout the proposed flight envelope;

b. provide data to allow the issue of a Certificate of Airworthiness or Permit to Fly; and

c. provide basic flight, performance and weight and balance information for future operation of the aircraft.

3.2 Evaluation Principles. Although modern stuctural standards were applied in the construction of the aircraft, in basic design it remained true to the form of an original Fokker VIIb-3M. The size and section of the wooden wing, the fuselage design and the size and arrangement of the tail surfaces were all the same as the original. The only anomalies between the two basic designs were a slight lengthening and downward canting of the nose section of the replica, and the inclusion of a tailwheel rather than a tail skid on the replica. The flight control system of the replica was also very similar to that of the original aircraft although some concessions, such as the inclusion of control surface mass balancing, had been incorporated. The flying qualities of the test aircraft were therefore found to be similar to those which could be expected from an original Fokker VIIb-3M. Consequently, the test aircraft could not meet many of the requirements of Reference B. Some of the aircraft's flight characteristics were found to be unsatisfactory by modern standards; however to attempt to rectify these deficiencies would probably detract from the presentation of the replica as such. During the evaluation the principle has been used whereby unsatisfactory features, arising from the basic aircraft design, have been called acceptable as long as there is no detriment to flight safety. Also, the test aircraft was a valuable, one-off replica so high-risk areas of the flight envelope were only investigated to the extent that safe flight characteristics could be ensured, assuming the aircraft was to be operated by experienced pilots in daylight Visual Meteorological Conditions (VMC).

3.3 Trial Location and Conditions. Initial ground tests and assessments were carried out at Parafield Airport in conjunction with final manufacture of the aircraft. Engine running, aircraft ground handling and high-speed taxiing were also carried out at Parafield while flight testing was conducted from both Parafield and RAAF Edinburgh airfields. All flights were conducted in daylight VMC. Stalling and low-speed handling tests were carried out above 4000 ft with

the only exception being the evaluation of the aircraft's asymmetric power minimum-control speed which was completed at 1500 ft. A total of 18.3 hours were flown in completing the evaluation. A summary of the flight test programme is presented at Annex B.

3.4 Limitations. As the aircraft had not been flown before the trial, the designer's calculated limits were applied initially until these could either be validated or modified. The engine manufacturer's limitations, as presented at Reference C, were applied throughout. Operating limits for other aircraft systems and equipment were based on the applicable manufacturer's specifications. The replica was a civilian aircraft and was therefore subject to Department of Aviation regulations. For the purpose of the trial, operations were conducted under a Permit to Fly (Annex C) which imposed some additional restrictions.

3.5 Aircraft Configuration and Test Loadings. The fixed landing gear and lack of flap or other high lift devices meant that the aircraft had only one flight configuration. Aircraft all-up-weight (AUW) was varied throughout the trial from a minimum of 4500 kg to the maximum of 5706 kg. The designer's centre of gravity (CG) range was used as a basis for the initial loading of the aircraft during the trial, however the lack of sufficient nose-down tailplane trim led to a recommended modification of the rearward limit of this range (see Paragraph 6.5.4.c and Annex D). Lead ballast was used to vary the aircraft's weight and CG position. The take-off weight and CG position for each sortie is presented at Annex B.

3.6 Instrumentation.

3.6.1 Flight Parameters. Flight parameters were measured or estimated in the cockpit using basic instrumentation. The aircraft flight and engine instruments were used to measure performance and attitude information. The pressure instruments were calibrated prior to installation; the aircraft pressure error corrections were then defined by having a RAAF CT-4A, with calibrated pressure instruments, formate on the test aircraft (refer to Paragraph 6.4). A portable temperature gauge and probe were used to measure ambient outside air temperature. Aircraft AUW was estimated by subtracting the calculated fuel used from the known aircraft take-off weight.

3.6.2 Control Positions. The positions of the cockpit flight controls were measured using scales and tape measures. Control position measurements were taken as follows:

a. Longitudinal Control Position - measured using a light-weight expandable tape measure set mid-way up the right-hand control column.

b. Tailplane Position - estimated using the cockpit gauge.

c. Lateral Control Position - measured using a fixed scale set up on the right-hand control yoke.

d. Rudder - measured using a fixed scale marked on the floor under the pilot's right-hand rudder pedal.

3.6.3 Control Forces. The control forces were measured or estimated at the cockpit. A hand-held force gauge was used initially to measure elevator and aileron forces, however the large amount of mechanical friction in the control circuits made the the use of this gauge impractical and so the control forces were then estimated. The rudder control forces were also estimated.

3.6.4 Carbon Monoxide. Carbon monoxide samples were taken from the cockpit and cabin using a Drager Hand Pump.

3.6.5 Take-Off and Landing Distances. Take-off and landing distances were measured using an Askania kinetheodolite.

3.6.6 Meteorological Conditions. Airfield surface meteorological conditions as measured at the control tower were considered accurate enough for the trial.

3.7 Chase Aircraft. Photographic and safety chase was carried out using either a CT-4A and/or UH-1H aircraft. The CT-4A was also used to enable the definition of the test aircraft's pressure error corrections (see Paragraph 6.4).

3.8 Trial Personnel. In all, a total of eight qualified test pilots flew the aircraft at some stage during the trial. A flight-test engineer was carried on all but the initial flight.

3.9 Safety Equipment. Standard flying clothing, which included flying boots, flying suit, jacket and gloves, was worn by all occupants. The pilots also used HGU-26/P Helmets and MBEU-5/P Oxygen Masks. The masks had the oxygen hoses removed and were used to reduce the noise levels at the pilots' microphones. All occupants were provided with a MC3 Parachute Assembly.

3.10 Resources. A total of 18.3 hours were flown in the test aircraft during the trial, while another 8.1 hours were flown converting the operating pilots onto the type. These hours were all paid for by the Southern Cross Museum Trust. A total of 7.0 hours CT-4A and 4.0 hours UH-1H were expended by the chase aircraft. The project team worked approximately 400 manhours completing the trial.

4. TEST PROCEDURES

4.1 The test flying techniques detailed in References D and E were used throughout the trial. The requirements and techniques outlined at Reference B were used as a basis for the evaluation. The general progression of the trial was as follows:

a. Cockpit Assessment - carried out during the final stages of aircraft construction.

b. Flight Control Evaluation - carried out in conjunction with the cockpit assessment then airborne during the first three flights.

c. Ground Handling - consisted of engine running, low speed-taxiing then high-speed taxi runs which included raising and lowering the aircraft's tail.

d. Initial Flight - included general appraisal of aircraft handling and a check of the aircraft's performance and handling following a simulated engine failure.

e. Pressure Error Corrections - definition of the pressure error corrections applicable to the aircraft's airspeed indicators and altimeters.

f. Stability and Control - evaluation of the aircraft's stability and control characteristics; initially with the aircraft CG in its mid range, then with it at the forward and aft limits.

g. Low-Speed Handling and Stalling - evaluation of stall characteristics with varying aircraft CG positions as at 4.1.f above.

h. Asymmetric Power Performance and Handling - evaluation of flight characteristics with only two of the aircraft's three engines developing power.

i. Aircraft Performance - definition of aircraft performance parameters including take-off and landing performance.

5. TESTS MADE

5.1 The following tests were made during the evaluation:

a. Cockpit and Systems Assessment.

b. Control Characteristics.

c. Ground Handling.

d. Pressure Error Corrections.

e. Stability and Control Characteristics.

f. Stall Characteristics.

g. Take-Off and Landing Performance and Handling.

h. Aircraft Performance.

i. Asymmetric Power Performance and Handling.

6. RESULTS AND DISCUSSION

6.1 Cockpit and Systems Assessment

6.1.1 Cockpit Entry and Exit. The cockpit was entered via the cabin through a 75 cm by 100 cm hatchway. The hatchway was located 50 cm above the cabin floor and a small inset step was provided. Once through the hatchway the pilot had to manoeuvre himself between the two seats and the centre console. The available manoeuvring area was small, restricting larger pilots from getting quickly and comfortably into their seats. Exit was achieved using the reverse procedure, i.e. climbing out backwards. Attempting to exit front first was difficult and dangerous since the pilot's back was arched uncomfortably and he could easily hit his head on the rear cockpit bulkhead or slip off the step. Emergency exit through the hatchway would be difficult and time consuming. If the aircraft was on the ground with the propellers stopped, emergency exit from the cockpit could probably be achieved more easily by climbing out the open windows. Although cockpit entry and exit procedures were difficult and uncomfortable to accomplish they were considered acceptable since the cockpit design was necessarily in keeping with that of the original aircraft.

6.1.2 Pilot Seating. The two pilot seats were fixed in position with no form of adjustment available. In addition, there was no provision for adjusting the position of the cockpit flight controls and therefore pilots of varying anthropometric percentiles had to adapt to less than ideal sitting positions. Some improvization was possible in that eye height or the inclination of the back could be varied by using cushions. A variety of test pilots, some very tall and others quite small, flew the aircraft during the trial and were generally comfortable in the seat. Each pilot seat was provided with an inertia-reel four point harness. The pilot seating was acceptable.

6.1.3 Field of View. The pilot's field-of-view from the cockpit was assessed qualitatively throughout the trial. The high wing, tailwheel configuration of the aircraft and the 1920's cockpit design produced large obstructed areas in the field-of-view outside the cockpit. The field-of-view from either seat was restricted in the opposite forward and rear quarters by the instrument coaming and rear cockpit bulkhead respectively. These restrictions were especially noticeable when the aircraft was in the tail-down attitude. When the aircraft was in a level attitude, either on the ground during take-off and landing or when airborne, the forward view from the cockpit improved markedly. Problems associated with the cockpit field-of-view will be discussed separately for the ground and airborne cases.

a. Ground. Each pilot could see only the propeller area for the engine on his respective side of the aircraft and neither pilot could see the lower propeller area of the centre engine. The high wing restricted the field-of-view above and behind the aircraft. Due to the restricted field-of-view during taxi the

pilots had to depend on each other for advice on the presence of external obstacles. The aircraft will generally be the centre of public attention, and when on the ground, may attract undisciplined spectators. There is clear potential for an accident if spectators approach the aircraft unobserved, while the engines are operating.

b. Airborne. The high wing restricted the field-of-view above and behind the aircraft. During turning flight the area into the turn behind the lowered wing could not be seen and therefore good lookout procedures were essential, especially when operating in the circuit or in congested airspace. The wing also restricted the pilots' view of the runway during the base turn.

The cockpit field-of-view was unsatisfactory, but acceptable provided that two pilots are used. A single pilot and a competent observer with extensive aviation experience may be satisfactory for some short duration sorties. It is recommended that during engine ground runs either both pilot seats be occupied, or the aircraft be attended by a ground observer to enable the continuous observation of the propeller arcs.

6.1.4 Instruments and Controls. The cockpit controls and instruments were evaluated initially on the ground and then airborne throughout the trial. In general the controls and indicators were well presented and easy to operate so only points of note are reported. The fuel controls and indicators warrant detailed comment and this is presented at Paragraph 6.1.5. The primary and all other important controls were within reach of either or both pilots when strapped in, although a single pilot, sitting in either seat, would be unable to reach all of the switches. The engine controls and instruments were arranged logically although care and positive identification techniques were required during airborne engine shutdowns. The mixture levers were not fitted with a positive detent to prevent inadvertent selection of the full lean position, but the possibility of this selection was considered remote. All flight and ancillary instruments, with the exception of the fuel gauges, were easy to see and read. The engine instruments were appropriately marked. The airspeed indicators (ASIs) were not marked at the time of testing but this was due to be completed at the end of the trial (see Paragraph 6.5). Controls and switches were labelled clearly and unambiguously although in some instances the decals used for this purpose were fading. It is recommended that these decals be treated to prevent deterioration or that they be replaced by more durable items. The cockpit controls and indicators were satisfactory. Due to the inability of a single pilot to reach all of the required switches, aircraft crewing by two pilots or a single pilot plus an observer with extensive aviation experience is recommended.

6.1.5 Fuel Controls and Indicators. The fuel control panel and fuel gauges were designed for similarity with the original aircraft and were located on the rear cockpit bulkhead immediately behind, between and above the pilots' heads. The following points are worthy of note:

a. Fuel Control Panel. The fuel control panel incorporated five mechanical fuel cock handles - three for the main flow lines and two for crossfeed. The handles were approximately 80mm long and protruded from the panel by up to 35mm. The panel included a

schematic outline of the fuel system which made correct fuel selections relatively simple. For normal operations all five handles were vertically oriented and the selections were easy to see. The handles were logically presented with each of the three main flow lines corresponding to the appropriate engine. Being behind and above the pilot, the handles, especially those toward the further side of the cockpit, were somewhat difficult to reach and operate. Their major drawback however, was that they would be a considerable hazard to the pilots in a crash situation. The large solid handles could easily damage an unprotected pilot's head during significant aircraft deceleration. To alleviate this hazard the pilots must wear suitable head protection. The fuel control panel was unsatisfactory but acceptable provided the cockpit occupants wear protective helmets when operating the aircraft.

b. Fuel Gauges. Fuel tank contents were indicated by three sight gauges located above the fuel control panel on the rear cockpit bulkhead. Each sight gauge gave a direct reading of the fuel level at the front of the associated tank and indicated the amount of fuel remaining in the tank. The fuel levels in these gauges were difficult to see because of their location in a dimly lit area behind and above the pilots. The difficulty in seeing the fuel contents was further aggravated when the fuel quantity was high. Accurate readings were also difficult to obtain due to the sensitivity of the indicated fuel levels to aircraft attitude. Changes of five degrees in pitch attitude could lead to incorrect fuel readings of up to ± 50 litres (when the aircraft was on the ground with the tail down the gauges under-read by up to 55 litres). Unbalanced bank angles also affected the indicated fuel levels with the outboard gauges being the most sensitive. During flight, accurate fuel quantity information was therefore only obtained by monitoring power settings and flight time; the fuel gauges could only be used to confirm the calculated fuel usage. The fuel gauges were unsatisfactory but acceptable. Accurate flight planning will be essential when operating the aircraft. Conservative fuel reserves should be carried and their calculation should be based on an accurate pre-flight fuel dip.

6.1.6 Cockpit Environment. The cockpit had open side windows and the pilots were partially exposed to the elements. Operating with an outside air temperature (OAT) of 5 degC or less was uncomfortable and it was necessary to wear a helmet and warm flying clothing, including thick woollen socks, boots and gloves. There was no cockpit heating provided. For long duration sorties at low temperatures two pilots would be needed to allow each in turn to warm his hands or take a hot drink etc., while the other flies the aircraft. The engines, located abeam the open cockpit windows, produced very high cockpit noise levels, especially at high power settings. An intercom system was provided, however it was important to ensure the helmet headphone system gave a good seal over the pilot's ears. A mask system was used to reduce noise levels at the microphones although a boom-mike was also tried and found to be acceptable. The carbon monoxide levels present within the cockpit were measured during flight at both high and low power settings. The highest

in-flight measurement was 10 parts per million (ppm) which occurred during a low power descent. During engine start 50 ppm (which is the maximum level allowable at Reference B) was recorded however this reduced to less than 20 ppm after five minutes. In general, although uncomfortable at times, the cockpit environment was satisfactory.

6.1.7 Cabin. The cabin was large enough to provide seating for up to eight passengers although its operational configuration will include four passengers seated along the left side with baggage and servicing equipment strapped down on the right side. Entry and exit was gained through a lockable door at the left rear of the cabin and there was also an emergency exit door at the right rear. Seating was reasonably comfortable although it would become trying during a long trip. There were no in-built toilet facilities. The cabin was warmer than the cockpit although it could still become cold in low OATs. The average carbon monoxide level in the cabin was 6 ppm, which was well below the acceptable maximum level of 50 ppm. The noise levels in the cabin were high and hearing protection for the passengers was required. Two intercom leads with headsets were provided. The cabin facilities were satisfactory although it is recommended that hearing protection be provided for passengers.

6.2 Control Characteristics

6.2.1 General. The primary flight controls were evaluated throughout the ground and flight tests. There were no significant differences between the control mechanical characteristics measured on the ground and those measured airborne. The design of the flight control systems remained essentially true to the original Fokker VIIb-3M systems, employing internally and externally rigged control runs through pulleys and fairleads. Consequently all flight control circuits had large amounts of inherent mechanical friction. Breakout forces were also high but well matched with the friction. The controls were generally heavy to operate although not to the point where it was difficult to fly the aircraft or tiresome for the pilot. The exposed control runs both inside the cockpit and outside the fuselage were susceptible to jamming by foreign objects. Reasonable precautions had been taken to minimize this risk but care will still need to be exercised in respect to loose objects in the cockpit.

6.2.2 Elevator Controls. The one-piece elevator was horn and mass balanced and was connected to the cockpit control columns by dual externally rigged cables attached to horns and torque tubes. The elevator control mechanical characteristics, as felt at the cockpit, were measured both on the ground and airborne and are presented in Table 6.1.

TABLE 6.1 - ELEVATOR CONTROL MECHANICAL CHARACTERISTICS

Ser. (a)	Characteristic (b)	Measurement (c)	Remarks (d)
1.	Breakout	12.5 lbf Pull 10.0 lbf Push	Measured at the centre of the control wheel.
2.	Friction	± 5-8 lbf	Estimated.
3.	Freeplay	± 1 mm	Control wheel centre.
4.	Oscillations	Nil	Friction sufficient to ensure control remained at any displaced position.

The control column could be moved longitudinally through a total range of 43 cm as measured from the centre of the control wheel. Force versus displacement curves were not plotted since the control would remain at any displaced position. The large amount of mechanical breakout and friction gave the elevator circuit a characteristically heavy feel and the control friction significantly masked the longitudinal trimmability of the aircraft. As the breakout and friction were well matched and of similar magnitude, the heavy feel did not create problems in controlling the aircraft in pitch. The elevator control system was satisfactory.

6.2.3 Tailplane Control. The adjustable tailplane was operated using a wheel veticaly mounted on the cockpit floor to the right rear of the left-hand pilot's seat. The wheel was easy to reach and to operate needing approximately 5 lbf to initiate movement. At least five half-turns of the trim wheel were required to move the tailplane through five percent of the available range. This was considered excessive but, due to the expense and difficulty of rectification, was accepted in this condition.

6.2.4 Aileron Control. The large plain ailerons were mass balanced and had no other aerodynamic features. The control mechanical characteristics were measured and are presented in Table 6.2.

TABLE 6.2 - AILERON CONTROL MECHANICAL CHARACTERISTICS

Ser. (a)	Characteristic (b)	Measurement (c)	Remarks (d)
1.	Breakout	± 6.5 lbf	Measured at the inside of the control wheel.
2.	Friction	± 5 lbf	Estimated.
3.	Freeplay	Zero	—
4.	Oscillations	Nil	Friction sufficient to ensure control remained at any displaced position.

The maximum control wheel deflection was ± 160 degrees from neutral. Force versus displacement curves were not plotted since the control would remain at any displaced position. A large amount of breakout/friction was evident and had the effect of increasing the lateral control heaviness, although not to an unsatisfactory level. The aileron control system was satisfactory.

6.2.5 Rudder Control. The rudder had aerodynamic balancing and a degree of feedback provided by the integral spring bias trim system. The rudder control circuit characteristics are presented in Table 6.3.

TABLE 6.3 - RUDDER CONTROL MECHANICAL CHARACTERISTICS

Ser. (a)	Characteristic (b)	Measurement (c)	Remarks (d)
1.	Breakout	± 15 lbf	Estimated
2.	Friction	± 5 lbf	Estimated.
3.	Freeplay	Negligible	—
4.	Oscillations	Deadbeat	—

Full deflection of the rudder pedals was ± 9 cm from neutral. The rudder control circuit was heavy and positive but commensurate with this type of aircraft. There were no problems associated with the magnitude of the breakout force, and rudder control was sufficient for both symmetrically and asymmetrically powered flight. The rudder control mechanical characteristics were satisfactory.

6.3 Ground Handling

6.3.1 The low-speed ground handling characteristics of the aircraft were evaluated during taxi over sealed and unsealed surfaces and in wind conditions of up to 20 knots. Ground handling during take-off and landing is reported in Paragraph 6.7. The aircraft was fitted with a tailwheel assembly which incorporated a castoring lock system. The original small diameter solid-tyred tailwheel failed during early taxi tests. The tailwheel assembly was redesigned to include a much larger wheel with a pneumatic tyre and this improved ground handling characteristics markedly. The power required to move the aircraft depended on aircraft weight and the surface on which the aircraft was parked but was typically between 1000 and 1200 RPM per engine. Once moving, the taxi speed was easy to control using power or brakes. Directional control was also easy using either, or a combination of, rudder, differential power and differential brake. With the tailwheel castoring lock engaged, heading changes were limited to approximately ±10 deg., but once the tailwheel was unlocked turns through larger angles could be carried out easily. The aircraft was sensitive directionally with the tailwheel unlocked and the bes

taxiing technique was to leave the tailwheel locked unless large heading changes were required. When turning crosswind or downwind the aircraft was stable and controllable directionally although caution was required in winds stronger than 15 knots. Also, in strong winds the wing started to generate lift and the aircraft became sensitive laterally. For these reasons a slow taxi speed should be maintained in all but light wind conditions. The low speed ground handling characteristics of the aircraft were satisfactory. The field-of-view restrictions described earlier necessitate the use of caution when taxiing and the requirement for two competent crew members to be occupying the pilot stations.

6.4 Pressure Error Corrections

6.4.1 Airspeed Indicating System. The pilot and co-pilot airspeed indicating systems were independent and used separate identical pitot-static tubes located under the leading edges of the left and right wing at the same distance out from the fuselage. There was provision for selection of alternate static sources but all pressure error correction (PEC) determination was carried out using the primary system.

6.4.2 Test Method. The PECs for the pilot and co-pilot airspeed indicating systems were determined by formating a RAAF CT4-A Airtrainer with known instrument errors on the test aircraft during steady level flight over a speed range of 67 to 103 KCAS. After stabilising at each test point, the airpeed indicator (ASI) readings for both aircraft were taken at 15 second intervals over a period of approximately two minutes.

6.4.3 Magnitude of PEC. As expected, the PEC for the pilot and co-pilot systems were almost identical and so all flight test data was grouped together to determine a PEC applicable to both systems. The flight test data is presented at Annex E together with the ASI calibration and definition of the various airpeeds referred to (ie IAS, RAS, CAS, EAS and TAS). The PEC was very large varying from +14 knots at the designer's calculated stall speed (Vs) of 57 KIAS to +2 knots at the designer's never exceed speed (Vne) of 113 KIAS. The PEC was always positive ie. the ASIs read low. The large PEC at the lower airspeeds provided an inherent buffer between the actual and calculated indicated stall speeds, while at the high-speed end of the flight envelope the PEC reduced to acceptable levels, limiting the possibility of an inadvertent aircraft overspeed. With the aircraft in the cruise configuration and trimmed for 80 KIAS then, accounting for PEC and instrument error, the calibrated airspeed was 88 KCAS. The large airspeed indicating system PEC was unsatisfactory but acceptable. It is recommended that pilots operate the aircraft with respect to indicated airspeed (IAS), altnough calibrated airspeed (CAS) should be used for navigational flight planning. An ASI Pressure Error Correction Chart is included at Annex E to aid in flight planning.

6.4.4 Airspeed Indicator Marking. Reference B (at FAR 23.1545) requires that the ASI be marked to indicate never exceed speed (Vne), manoeuvre speed (Va) and stall speed (Vs) to the pilots. The reference requires these markings to be in terms of CAS; however, because of the large PEC for the test aircraft it is recommended that they be in terms of IAS to make them of immediate use to the pilot. The derivation of these markings is presented at Annex F. The recommended markings for both the pilot and co-pilot ASIs are:

Vne (113 KCAS) marked as 109 KIAS
Va (90 KCAS) marked as 83 KIAS
Vs (57 KCAS) marked as 45 KIAS

6.4.5 Altimeter PEC. Assuming that all the pressure error for the ASI system resulted from errors in the sensing of static pressure, the corresponding PEC for the altimeters at sea level would be -54 ft at Vs and -52 ft at Vne (ie. the altimeter would indicate a lower altitude than actual and would therefore be conservative). During the PEC test flights the altimeter PEC was checked and found to be negligible.

6.5 Stability and Control Characteristics

6.5.1 General. The longitudinal, lateral and directional stability and control characteristics of the aircraft were investigated throughout the trial. Measurements were taken for an aircraft cruise configuration at 80 KIAS and power for level flight (PLF); a climb configuration at 70 KIAS with climb power set; and an approach configuration at 70 KIAS with a minimum power setting. Static and dynamic stability characteristics were evaluated with the CG varying from a full forward position, to an aft position of about 60% of the design range. The aircraft displayed neutral to weak positive static stability about all three axes. Controllability was sufficient to compensate for the low static stability and the aircraft was generally easy to fly, although it required constant monitoring. There were no unsatisfactory dynamic characteristics.

6.5.2 Longitudinal Static Stability. The stick-fixed and stick-free longitudinal static stabilities of the aircraft were evaluated in the cockpit by measuring the changes of fore-aft control column displacement and force with changes in airspeed. The aircraft configuration, weight and CG information for these tests is summarized in Table 6.4.

TABLE 6.4 - STABILITY TEST CONFIGURATIONS

Ser. (a)	Configuration (b)	Trim Airspeed (KIAS) (c)	Power/ Engine (in Hg/RPM) (d)	Take-Off Weight (kg) (e)	Take-Off CG (%MAC) (f)
1.	Cruise (CR)	80	18-20/1950	4503	23.2
2.	"	"	"	5093	28.9
3.	Climb (CL)	70	24/2050	4503	23.2
4.	"	"	"	5093	28.9
5.	Powered Approach (PA)	70	12-13/ approx 1700	4503	23.2
6.	"	"	"	5093	28.9

The tests were carried out at pressure altitudes between 1000 and 3000 ft. The aircraft was stabilised and trimmed in each of the CL, CR or PA configurations and then, using the elevator control alone, the airspeed was at first reduced by 20-25 KIAS (in 5 KIAS increments), returned to trim airspeed, then increased by 20-25 KIAS. At each stabilized point the longitudinal control position and force were measured or estimated. The test results are presented at Annex G.

a. Stick-Free Stability. The inordinate amount of longitudinal control friction made it very difficult to assess the weak underlying control forces accurately. The friction was also sufficient to mask any discernible differences in the stick-free stability of the aircraft when in the different configurations or with the CG in either the forward or aft positions. A variation in airspeed of ±15 KIAS from trim was necessary before a change in the required control force was noted. Even then these forces were only approximately 3 lbf per further 10 KIAS of airspeed variation and were accompanied by fluctuations around this mean of an estimated ±1-2 lbf.

b. Stick-Fixed Stability. The weak stability characteristic was also reflected in the measurement of longitudinal control column position with variation in airspeed. Measurements were taken midway between the top of the control column and the cockpit floor and were only a few millimetres per 10 KIAS variation in airspeed. Once again there was virtually no apparent differences with change in aircraft configuration or CG position.

The longitudinal control power was well matched to the breakout/friction forces and was more than sufficient to compensate for the lack of stability. There were no problems with controllability and the aircraft was reasonably easy and pleasant to fly. The weak longitudinal stability characteristics did require the pilot to devote some attention to airspeed control, and consequently fly the aircraft 'hands on' almost constantly. On long duration flights, the constant monitoring task would become tiring, necessitating the employment of two pilots. The longitudinal static stability characteristics of the aircraft were acceptable. It is recommended that a minimum of two pilots be employed for sortie durations longer than two hours.

6.5.3 Longitudinal Dynamic Stability and Control. A qualitative assessment of the short period pitch oscillation (SPPO) and phugoid characteristics of the aircraft was carried out at the configurations presented at Table 6.4. The SPPO exhibited a low frequency with moderate to heavy damping, giving the aircraft its characteristic heavy response to pitch control inputs. The pitch response was positive and there were no longitudinal control problems. Elevator control power was sufficient during all stages of flight and was well matched with the aircraft's SPPO. The phugoid was difficult to demonstrate or observe because of the aircraft's large trim speed band and, if present, was not intrusive during any phase of flight. The aircraft was generally easy to control in pitch and should not provide any problems during normal operations. The longitudinal dynamic stability and control characteristics were commensurate with this type of aircraft and were satisfactory.

6.5.4 Longitudinal Trim. The longitudinal trim characteristics of the aircraft were assessed throughout the trial with special attention being given to changes in trim with changes in the position of the CG, changes of power and changes of airspeed. Trimmability was also assessed; trimming the aircraft in pitch was accomplished by adjusting the position of the horizontal tailplane. Four aspects of the aircraft's longitudinal trim characteristics deserve comment.

a. Trim Speed Band. The neutral to weakly positive longitudinal static stability of the aircraft, in combination with the large amount of control friction, produced a relatively large trim speed band (approximately ±15 KIAS). Given that the speed range of the aircraft was only from a stall speed at 45 KIAS to a Vne at 109 KIAS, the 30 KIAS trim speed band is quite substantial. The large trim speed band made the aircraft difficult to trim precisely, and contributed to the requirement for the pilot to constantly fly the aircraft 'hands on'.

b. Effect of Power and Airspeed. Large changes of power or airspeed affected the longitudinal trim of the aircraft in the conventional sense (ie. the application of power or an increase in airspeed caused a pitch-up and vice-versa). The magnitude of these trim changes was small and easily controlled using the elevator. With the aircraft trimmed for 70 KIAS and the power at idle, a rapid application of full power would cause the nose to pitch up slowly at less than one degree per second. The pitch up was easily controlled using less than 10 lbf at the control column. If the aircraft pitch attitude was maintained during the power application, the airspeed would increase at approximately 2 KIAS per second. Changes in the longitudinal trim of the aircraft with changes in power and airspeed were satisfactory.

c. Centre of Gravity Range. The aircraft required large amounts of nose-down trim to be selected at all times (between 50% and 100% of the available nose-down trim range). Examination of photographs of the original Southern Cross in flight indicate that aircraft also required large amounts of nose-down trim since the tailplane is generally seen in the full leading edge up position. With the aircraft CG at 29.0% MAC the amount of forward trim required was at the maximum available and so this CG position should be considered as the aft limit. (Note: 29.0% MAC equates to 60% of the design CG range. Weight and CG information is summarised at Annex D). Conceivable practical loading of the aircraft will put the CG forward of 29.0% MAC and so an aft limitation of the CG at this position will not affect the general operation of the aircraft. It is recommended that, due to the lack of available nose-down tailplane trim, the CG range be limited to between 22.6% MAC (which equates to the forward design limit) and 29.0% MAC. Loading of the aircraft with the CG either aft of 29.0% MAC or forward of 22.6% MAC would be possible and care should be taken to ensure that the proposed limits are not exceeded. A load sheet for incorporation in the Flight Manual is presented at Annex D.

d. Tailplane Trim Setting. The adjustable horizontal tailplane was quite a powerful means of varying the longitudinal trim of the aircraft. With tailplane trim settings of less than 50% of the available nose-down range the control column push force increased significantly (30 lbf or greater) and the aircraft became difficult to control in pitch. The trim power with the tailplane set at between 50% and 100% of the available nose-down range was sufficient for all normal operations, and it is recommended that mechanical stops be incorporated into the system to limit tailplane movement to this range.

6.5.5 Lateral/Directional Static Stability. The lateral and directional static stability characteristics of the aircraft were assessed in the configurations laid out at Table 6.4 and at pressure altitudes between 1000 and 3000 ft. Steady heading side-slips (SHSS) and turns on one control were used to obtain an an estimation of the aircraft's stick-fixed and stick-free lateral/directional stabilities. SHSS results are presented at Annex H. Once again the large amount of inherent control friction was dominant and made the precise measurement of the weak underlying control forces difficult. This was especially true of aileron control force. Differences with change in configuration or CG position were, as for the longitudinal case, negligible. The aircraft displayed weak but positive static stabilities in both the lateral and directional planes. When considering the layout of the aircraft (ie. the large thick wing and the relatively small amounts of keel, fin and rudder surface) a lack of directional stability could be expected, however it was found that the lateral and directional characteristics were well matched. The large wing coupled with the plain ailerons did lead to the aircraft developing noticeable adverse yaw on lateral control application but this could be countered easily with the use of rudder to balance all turns. The lateral and directional static stabilities were satisfactory, given the design of the aircraft.

6.5.6 Lateral/Directional Dynamic Stability and Control. The dynamic characteristics of the aircraft in the lateral and directional planes were evaluated in the configurations at Table 6.4, and qualitatively throughout the trial. The stick-fixed and stick-free Dutch roll and spiral stabilities were excited and observed. The Dutch roll displayed a yaw to roll ratio of approximately 2:1 and was well damped with only one or two overshoots. It was difficult to excite unintentionally and was not noticed during normal operations. Spirally the aircraft was slightly divergent at the rate of approximately 0.5 degree per second. This did not create any problems as it was readily apparent and easily controlled. Lateral and directional controllability was sufficient for all phases of flight. Aircraft roll rates were checked in the CR configuration. With rapid application of full aileron and the rudder held neutral, the aircraft took approximately five seconds to roll through a bank angle change of 45 degrees. There was no significant difference between the roll rate to the left and that to the right. Rudder control power was sufficient and was needed during normal operations to balance the large amount of adverse yaw induced by the application of aileron. The lateral and directional dynamic stability and control characteristics were commensurate with the aircraft design and were acceptable.

6.5.7 Lateral/Directional Trim. Large changes of power resulted in large changes in the directional trim of the aircraft. At high power settings and low airspeed (eg. during take-off) approximately 50% right rudder trim was required. The spring bias rudder trim system was sufficiently powerful to counter these changes, however, precise directional trimming was difficult due to a lack of sensitivity inherent in the trim system. This was not normally a problem since the balance of the aircraft was constantly changing with gust response and aileron application. The lack of a lateral trim facility went unnoticed since the lateral trim of the aircraft did not undergo any large changes. The lateral and directional trim characteristics of the aircraft were acceptable.

6.5.8 Gust Response. The large wing made the aircraft sensitive to atmospheric turbulence and gusts, especially in the lateral and directional planes. At lower airspeeds (70 KIAS or less) the susceptibility of the aircraft to gusts increased and was most noticeable at lower power settings when the slipstream effect over the tail surfaces was reduced. The aircraft's gust response at low airspeeds and low power settings made flying an accurate landing approach quite difficult in gusty conditions. The aircraft was manageable in turbulence but it is recommended that intentional flight in strong or gusty wind conditions be avoided.

6.5.9 High Speed Characteristics. The flight characteristics of the aircraft at high speed were investigated for both the forward CG and aft CG cases. The aircraft was accelerated to 110 KIAS (equivalent to the Vne of 113 KCAS) using maximum continuous power (24 in Hg/2050 RPM) and a slight dive (approximately 3 deg.ND). There were no uncommanded changes in aircraft attitude at this speed and control forces were only marginally higher than for normal cruising flight. The increase in airframe and engine vibration and buffet levels was small with the greater wind-noise being the only noticeable effect. The aircraft's high-speed characteristics were satisfactory.

6.6 Stall Characteristics

6.6.1 General. An evaluation of the aircraft's low-speed handling and stall characteristics was conducted during the trial. Stalls were approached in an incremental manner and all were conducted at an altitude greater than 4000 ft above ground level. Stalls were carried out at forward and aft CGs and with power settings which varied from idle to 20 in Hg/2050 RPM. Actual observed stall speeds were generally up to 5 KCAS less than those calculated using design note predictions. The large PEC factor at low airspeeds (especially given that extrapolation was being used below 65 KCAS) would account for this discrepancy and it was assumed that the design maximum AUW stall speed of 57 KCAS was accurate. The evaluation was somewhat limited in that dynamic turning stalls or stalls with the aircraft under full power were not covered on the chance that the aircraft could depart from controlled flight. Given the docile nature of the aircraft's stall this was considered unlikely but because the aircraft held valuable and unique status it was decided not to proceed further than necessary with this phase of the evaluation. The general stall characteristics of the aircraft, to the level tested, were satisfactory.

6.6.2 Forward CG, Power Off Stall. Full forward CG stalling characteristics were assessed with an aircraft take-off weight of 4528 kg and the CG at 22.7% MAC. The aircraft was trimmed at 60 KIAS with power at idle (less than 10

in Hg/1500 RPM). The airspeed was reduced at a rate of approximately 1 KIAS per second. A constant rate of airspeed reduction was difficult to achieve due to unsteady ASI PEC effects below 50 KIAS. Controllability about all three aircraft axes was checked throughout the deceleration and found to be sufficient to allow recovery at any stage. At 45 KIAS the aircraft had a 5-10 degree nose-up attitude and a 300 ft per minute rate of descent (ROD). The control column position was at 60% of its aft range and an aft stick force of approximately 10 lbf was required. The aircraft felt less stable about all three axes in that its sensitivity to atmospheric gusts was more marked. At 33-35 KIAS the aircraft was considered to be in the stall since full back stick was being applied and a ROD of 900-1000 ft/min had been established. The aircraft attitude was generally between 10-15 degrees nose-up with the wings level although an occasional tendency to pitch up a further 2-3 degrees was noted. All controls were still effective. Although no airframe buffet was noted prior to the stall, the nose-up attitude, rearward stick position and the increased aircraft gust response were considered as adequate stall warning. Recovery could be effected almost immediately by either releasing the applied back stick, applying power or a combination of both. Typical height loss was about 150 ft, taken from the time recovery action was initiated until a rate of climb had been established. The stall under these conditions was mild and, even though there was no tactile stall warning in the form of airframe buffet, the stall was considered unlikely to be encountered inadvertently. Within the scope of this test, the power off, forward CG aircraft stall characteristics were satisfactory.

6.6.3 Rear CG, Power Off Stall. The power off stall characteristics were also observed with the aircraft at a higher AUW and with the CG in a rearward position. The take-off weight was 5093 kg and the CG was at 28.9% MAC. The stall was approached using the same method quoted in Paragraph 6.6.2, however in this instance it was noted that the control column position required for deceleration to less than 45 KIAS was only approximately 40% of the available rearward movement. Nose high attitudes and rates of descent similar to those in the forward CG case were encountered, however a very slight airframe buffet could be discerned as the airspeed reduced below 37 KIAS. At 35 KIAS, with the stick at about 60% to the rear, there was a definite, although mild 'g'-break and the nose pitched down through approximately 5 degrees at a rate of approximately 2 degrees per second. The nose-down pitch was occasionally accompanied by a gentle wing drop through 5-10 degrees angle-of-bank at about the same rate as the nose drop. Apart from one occassion, all of the wing drops were to the left. The recovery was the same as for the light-weight forward CG case with height losses of the same magnitude. The stall in this configuration was quite docile and should not provide any problems for the operational pilots. The power off, aft CG aircraft stall characteristics were satisfactory.

6.6.4 Power On Stalls. The aircraft stalling characteristics with increasing levels of power selected were investigated at both the forward and aft CG. The maximum power selected during the approach to the stall was 20 in Hg/2050 RPM. With the introduction of power, the approach to the stall was similar to the power off situation except that the nose-high attitude increased to between 15 and 20 degrees above the horizon immediately prior to the stall. Elevator control was more effective due to the increased slipstream over the tailplane and at 45 KIAS the control column position was only 30% to 40% to the rear. A very weak airframe buffet could be detected from 2-3 KIAS above the stall, however this was of insufficient intensity to be considered as a

reliable stall warning. As for the power off cases, the high nose attitudes and low indicated airspeeds were considered adequate warning especially since only experienced pilots should be flying the aircraft. At the power settings tested the stall speeds were 2-3 KIAS lower than those observed with the power at idle and the aircraft at an equivalent weight. The stall itself consisted of a mild 'g'-break, pitch down and left wing drop. The maximum change in angle-of-bank was 20 degrees. The aircraft was completely controllable throughout the approach, stall and recovery with maximum height losses of 150-200 ft. The chances of an inadvertent stall occuring during normal operation of the aircraft are considered small. Taking this and the excellent stall recovery potential into account, the power on stalling characteristics of the aircraft were satisfactory.

6.7 Take-Off and Landing Performance and Handling

6.7.1 General. The take-off and landing phases of flight were areas where particular caution was required when operating the aircraft. The large thick wing, relatively small fin and rudder area and the general susceptibility of the aircraft to gusts meant that during the take-off and landing ground rolls there was always a possibility of an uncommanded deviation from runway centreline. Controllability about all three axes was sufficient, however, and as long as care and anticipation were used the aircraft was safe to operate, even from short and narrow runways. Extra care was needed when operating in gusty conditions or with a crosswind. It is interesting to note that these problems only take on significance in the modern environment with the requirement to operate this type of aircraft to and from a runway. When the original aircraft was operating, the standard landing area consisted of an all-over grass field and crosswinds as such did not exist. Wind conditions encountered at take-off and landing during the evaluation were up to 20 knots with a right crosswind component of 14 knots; however, the aircraft was operating on a runway which was 8400 ft long and 200 ft wide when the most demanding conditions were experienced.

6.7.2 Take-Off Technique and Handling Characteristics. High-speed ground handling characteristics and the take-off technique were initially evaluated using a series of high-speed taxi runs before the aircraft's first flight. During the line-up, it was important to ensure that the tailwheel castoring lock was engaged as any attempt to take-off or land with the tailwheel in the unlocked position could lead to increased directional control problems and the possibility of a ground loop. The take-off trim settings were normally 50% of the right rudder range and 80% of the nose-down range. The take-off was accomplished from a rolling start coincident with the smooth application of full power (approximately 28 in Hg/2200 RPM under ISA sea level conditions), and the aircraft accelerated quickly. All controls were effective at a very low airspeed (approximately 10-15 KIAS) and the aircraft was easy to control both directionally using rudder (with possibly an early touch of differential brake) and laterally using aileron. At between 20 KIAS and 25 KIAS a control column push force of approximately 15 lbf and forward movement from neutral through approximately 30% of the available forward stick displacement, was used to lift the tail off the ground. Lifting the tail into the slipstream increased both the available rudder and elevator control power. Once the tail was off the ground it was important to keep the aircraft in a level or slightly nose-down attitude, as too high an angle-of-attack on the wing at this stage made the aircraft light on its wheels and subsequently more difficult to control directionally. At 55 KIAS the aircraft was rotated about the main

wheels and flown clear of the ground. The rotation required a pull force of approximately 10-15 lbf and rearward movement of the control column through 20% of its range of movement. Aborted take-off characteristics were also evaluated and were similar to those experienced during the later section of the landing roll (described in Paragraph 6.7.3). The take-off was generally easy to accomplish although anticipation and early correction of any deviation from desired flight path was essential. Deviations were more likely to occur in crosswind or gusty conditions. The take-off handling characteristics were generally satisfactory although the aircraft should only be operated by experienced pilots and within the conditions set out at Paragraph 6.7.3.

6.7.3 Landing Technique and Handling. Landing the aircraft required care and anticipation especially in gusting conditions or crosswinds. The aircraft's lack of flap meant that the safest and most comfortable means of conducting the approach to land was through a shallow glide path with some power applied. Typical power settings on mid to late finals were between 11 and 15 in.Hg with the RPM at approximately 1500. Approaches were flown at 70 KIAS, aiming for threshold speeds of 60 KIAS to 65 KIAS. Given the large PEC, the aircraft was actually crossing the threshold at 70 KCAS or greater. Although this threshold speed was a little high, the aircraft's susceptibility to gust effects was reduced and a more accurate approach could be flown. Flare and touchdown techniques were standard with only 'wheeler' type landings being carried out. Smooth touchdowns could be achieved using a coordinated flare and power reduction to idle. There was sufficient controllability throughout the approach and landing although, in crosswinds of 10 knots or more, full aileron control input could be required during the latter stages of the landing roll. After the main wheels were on the ground it was necessary to hold the aircraft in the level or slightly nose-down attitude until below 45 KIAS, when the tail could be lowered carefully. If the tail was lowered too quickly or at too high an airspeed the wing would regain lift and the aircraft could easily become airborne again. A situation where the tail was lowered without the wings being level could lead to one wheel lifting and the aircraft deviating directionally toward the down-going wing. In order to ensure continued directional control it was essential to regain wings level with both wheels on the ground. The possibility of one wing lifting was liable to arise most often in crosswind conditions, and if the aircraft was landing on a narrow runway there was a danger of running off the edge of the flight strip. For this reason it is recommended that the maximum allowable crosswind component for take-off or landing on runways of less than 90 ft in width should be 5 knots. For wider runways, a crosswind limit of up to 14 knots could be applied. It is preferable to have the crosswind from the right rather than the left. The nomination of these runway widths and crosswind limits is based on the runways at the cities and towns where the aircraft is likely to be operated. The landing characteristics of the aircraft were acceptable, however particular caution should be exercised in all instances where strong or gusting winds are present during take off or landing.

6.7.4 Take-Off and Landing Performance. The take-off and landing performance of the aircraft was measured at RAAF Edinburgh using kinetheodolite tracking. The tests were carried out under the conditions outlined at Annex I. The raw test data was reduced to standard conditions (using methods developed in Reference F) then expanded to include operations at a pressure altitude of 3000 ft and a maximum ambient temperature of 50 deg C. Standard corrections for runway surface, slope and wind direction have been addressed in the development of the performance graphs presented at Annex I. Take-off and

landing techniques used during the tests were quite conservative. Safety factors have not been incorporated in the presented data.

a. Take-Off Ground Roll. The take-off ground roll was defined as that distance from where the aircraft was stationary to the point where the main wheels broke ground. There were five take-off ground rolls completed for which the results are included at Annex I. The longest ground roll of 402 m was with an aircraft weight of 5650 kg and with no headwind. Conversely, the shortest ground roll of 219 m resulted from operating at 4510 kg with an eight-knot headwind. Figure 1 of Annex I has been developed in accordance with the recommendations of Reference F and accurately reflects the test data. The data is unfactored and indicates a ground roll of 390 m at sea level ISA conditions and maximum AUW.

b. Take-Off Air Distance. The take-off air distance was defined as that distance from the point where the wheels broke ground to the point where the main wheels passed through 15.3 metres above the ground. The maximum take-off air distance of 364 m occurred with the aircraft at 5680 kg AUW in still conditions. The data was reduced and expanded in accordance with Reference F and indicates rapidly decreasing take-off climb out performance with increasing altitude. The unfactored take-off air distance at sea level, ISA conditions and maximum AUW is 350 m.

c. Landing Air Distance. The landing air distance was defined as the distance from where the main wheels first passed through 15.3 m above the ground to the point where they first touched the ground. The landing air distances detailed at Annex I reduce with increasing weight. This effect is the result of flying the final approaches at a constant power setting. The higher weight means higher rates of descent at a constant speed and hence a steeper flight path. The weight factoring equation in Reference F could not cater for this effect so the maximum runway distance was used as a constant distance regardless of AUW. The landing air distance graph presented as Figure 3 to Annex I is unfactored and shows a sea level ISA landing air distance of 690 m.

d. Landing Ground Roll. The ground roll was defined as that distance from where the main wheels initially touched the ground to that point where the aircraft became stationary. Although the aircraft was fitted with brakes on the mainwheels, only light to moderate braking was applied after the tailwheel had been lowered to the ground. The maximum of the four landing ground rolls was 822 m at 5665 kg in still conditions while the shortest ground roll of 553 m was achieved at 4490 kg with an eight knot headwind. The data has been expanded in accordance with Reference F and is presented graphically in Figure 4 of Annex I. The unfactored landing ground roll at sea level ISA conditions and maximum AUW is 840 m.

e. Operational Runway Lengths. The aircraft was tested with the intention of presenting take-off and landing distances that do not require exceptional pilot skills to repeat. However, the combination of the flapless take-off and landing techniques used,

the large PEC factor and the use of only light braking tend to portray the aircraft as requiring runway lengths which were greater than actually necessary. As long as favourable approach paths are available and the aircraft is landed by a current, experienced pilot using the correct technique, threshold speed and touchdown point, a minimum runway length of 1080 metres can be safely accepted for landing at sea level ISA conditions and an AUW of 5000 kg. These figures are based on the operational landing technique whereby, for runways of 1500 metres or less, the pilot nominates a last point of touchdown no further than 300 metres beyond the runway threshold. If the aircraft is not on the ground by this point then an overshoot must be executed and the approach re-flown. Using this technique the aircraft crosses the threshold at a height of 10 ft instead of 50 ft and a realistic runway length for landing can be taken as a fixed air distance of 300 metres plus the required ground roll distance presented in Figure 4 at Annex I. This method is recommended for determining operational runway lengths.

6.8 Aircraft Performance

6.8.1 Climb Performance. The climb performance of the aircraft was determined using the specific excess power technique described at Annex J. The climb performance was monitored throughout the flight test series. Specific sawtooth climb data was recorded during dedicated aircraft performance sorties at both light and heavy weight (4532 and 5706 kg take-off weight respectively). The sawtooth climbs were conducted over the 55 to 85 KIAS speed range and from 1500 to 4500 ft pressure altitude. The climbs were carried out with climb power (24 in Hg and 2050 RPM) set on all engines. The climb performance was essentially independent of pressure altitude below 4000 ft because the aircraft was capable of developing the climb power of 226 BHP per engine up to that altitude (ie. the throttles were progressively opened throughout each climb to maintain the manifold pressure at 24 in Hg). The climb performance test results are included in Annex J. The optimum climb speed of 70 KCAS (60 KRAS) was obtained from the specific excess power curve presented at Figure 1 of Annex J. The graph presented at Figure 2 of Annex J represents climb performance at 70 KCAS as a function of ambient temperature and aircraft gross weight. The figures indicate a rate of climb (ROC) of 385 ft/min at sea level ISA conditions. The ceiling altitude of the aircraft was not investigated. The highest pressure altitude achieved during the trial was 6500 ft at an AUW of approximately 4600 kg. Overall, the aircraft climb performance was satisfactory.

6.8.2 Level Flight Cruise Performance. The level flight cruise performance was determined using the speed-power technique described in Annex K. Data was recorded during the two dedicated (lightweight and heavyweight) performance sorties and also during the pressure error correction sorties when the aircraft AUW was approximately 4900 kg. During the testing, the power per engine varied from a maximum of 24 in Hg/2050 RPM to a minimum of 14 in Hg/1650 RPM. The flight test data is included at Annex K. The cruise performance graphs (Figures 6 and 7 of Annex K) include fuel flow, endurance and range information, and are suitable for use during flight planning. Some interpolation between the two different sets of ambient conditions presented (ISA at sea level; ISA+10 at 5000 ft) may be required.

a. Fuel Flow. Due to the inaccuracies in the fuel indication system fuel flow information could not be recorded during the flight testing. Fuel usage between start up and shutdown was monitored accurately. The fuel flow figures used in the performance calculations were taken from the engine manufacturer's data (Reference C) for a cruise RPM of 1950. The presented fuel flow data has been factored by five percent to account for engine degradation in service. Figures 6 and 7 of Annex K present fuel flow as a function of AUW and airspeed. Figure 6 indicates a normal cruise (90 KCAS) fuel flow of 152 litres/hour for an aircraft AUW of 5000 kg at sea level ISA conditions.

b. Endurance. The best endurance speed under sea level ISA conditions varied from 50 KCAS at 4300 kg to 57 KCAS at 5700 kg. These optimum airspeeds are impractical in operation because they are close to the stall and on the 'bucket' of the drag curve, thereby requiring power manipulation to fly accurately. A best endurance airspeed of 65 KCAS (53 KRAS) would only result in a 13% degradation of optimum fuel flow and is recommended for use in operation.

c. Range. The airspeeds for best range are a function of ambient temperature, pressure altitude and AUW. Figures 6 and 7 of Annex K show variation in specific air range (SAR) with airspeed. The best range speed under sea level ISA conditions varies between 63 KCAS at 4300 kg and 75 KCAS at the maximum AUW of 5700 kg; while at 5000 ft and ISA+10 deg.C the best range speed varies from 65 KCAS to 75 KCAS at equivalent weights. A selected optimum range airspeed of 75 KCAS (65 KRAS) would result in a maximum of 5% degradation in SAR for all weights and temperatures below 5000 ft and is recommended for use in operation.

6.8.3 Engine Performance. A detailed quantitative evaluation of engine performance was not attempted. The validation of design aircraft performance predictions was taken as confirmation that the engine manufacturer's data was accurate. Engine parameters were monitored throughout the trial with no major anomalies being noted. The normal engine operating limits (manifold pressures and RPMs) were well matched to aircraft performance. Cylinder head temperatures (CHT) under high power conditions, and during prolonged climbs, stayed well within the published limits. However, if the engine was run at low power (typically less than 12 in.Hg) for periods longer than 20 - 30 seconds, the CHT dropped rapidly and care was needed to ensure that high power was re-introduced gradually (see Paragraph 6.9.2). Oil pressure generally ran high and, for brief periods after start, was occasionally above the 90 psi maximum limit. The measurement of oil temperature was made at the oil tank outlet, and consequently long periods after start were spent warming the engine and waiting for the oil temperature to climb above the 33 deg.C minimum required for run-up. The carburettor heat facility was checked airborne and found to provide an increase of approximately 15 - 20 deg C in carburettor air temperature per notch of control movement. Overall, the engines operated well and gave the impression of reliability. The engine performance was satisfactory.

6.9 Asymmetric Power Performance and Handling

6.9.1 General. The performance and handling qualities of the aircraft with a single engine inoperative were evaluated during the trial. With only two engines operating the aircraft performance suffered, although the asymmetric handling characteristics were quite benign. Flight with two engines inoperative was not investigated but it can be assumed that the reduction in performance with the aircraft in this condition would probably lead to a forced landing.

6.9.2 Engine Shutdown and Restart. Each engine was shut down and then restarted while airborne at least once during the trial. There were no noticeable differences between the engines in their actual shutdown or windmilling characteristics. In all instances engine shutdown was carried out by moving the mixture control to the full lean position. The throttle positions at shutdown varied from idle to 20 in Hg. The position of the propeller lever was also varied from either the full fine position, the full coarse position or the simultaneous reduction from full fine to full coarse as the mixture lever was moved. This variation was carried out in order to ascertain the most advantageous position of the propeller lever for subsequent windmilling RPM. Very little difference in these RPM was noticed and it was concluded that the position of the propeller lever prior to engine shutdown was unimportant. Once the engine was shut down the propeller windmilling RPM depended only on airspeed, varying between 850 RPM at 70 KIAS and 500 RPM at 50 KIAS. After engine shutdown the cylinder head temperatures fell quickly from their normal cruise power values of approximately 190 deg C and could easily stabilize at less than the minimum value for flight (120 deg C). The effects of an actual engine shutdown on the aircraft performance and handling were almost identical to those with the equivalent throttle set to idle (10 in.Hg/1500 RPM) and therefore for pilot training or demonstration purposes, moving the throttle to idle could be used to simulate an engine failure. The CHT decay with the throttle at idle was also quite rapid. Airborne restart was accomplished by selecting the propeller lever to the full fine position, ensuring the throttle was closed then advancing the mixture lever to full rich. Engine response on opening the throttle was immediate, however care was required to ensure the engine was only run at a low power setting until the CHT had risen sufficiently. Airborne engine shutdown and restart procedures and characteristics were satisfactory.

6.9.3 Critical Engine. The critical engine, in respect to the effect on the aircraft following engine failure, was the port engine. Propeller rotation, when viewed from the rear, was clockwise and consequently the slipstream and propeller asymmetric blade effects combined to give the aircraft a natural tendency to yaw to the left under high power settings and at high angles of attack. In the event of a starboard engine failure this effect was advantageous; however, if the port engine had failed, the natural left yaw aggravated the left yaw created because of the inoperative engine. This effect was confirmed during the trial when it was found that the right rudder deflection and force required following shutdown of the port engine was approximately ten percent greater than the amount of left rudder required when the starboard engine was shut down. These effects were most noticeable at low airspeeds and high power settings.

6.9.4 Minimum Control Speed. The static engine out minimum control speed (Vmca) tests were carried out with an aircraft AUW of 4900 kg and the CG at 28.7% MAC. The aircraft was trimmed at 60 KIAS and 1500 ft with power for level flight. The rudder trim was set to neutral. Full power (approx. 26 in Hg/2200 RPM) was set on the centre and starboard engines while the port engine was set to idle. The airspeed was then steadily reduced and the rudder and aileron control inputs required to maintain the aircraft on a steady heading were noted. A minimum airspeed of 33 KIAS was reached with a right rudder force of approximately 140 lbf and 90-95% of available rudder deflection applied. At this speed the aircraft was in a 20 degree nose-up attitude and approaching the stall. Some directional control was still available but further deceleration was considered unwise. Dynamic effects during simulated engine failures with the aircraft at full power and at 50 KIAS were also checked with lateral and directional control being sufficient in all cases. While the actual minimum control speeds were not observed, it was demonstrated that under asymmetric power conditions very low indicated airspeeds could be sustained without the onset of directional control loss.

6.9.5 Engine-Out Climb Performance. The climb performance of the aircraft when only two engines were operating was checked a number of times during the trial. A comparison of performance with each engine in turn being shut down, and while the other two were at full power under the same ambient and aircraft conditions, revealed negligible differences in rates of climb. With the centre engine inoperative and the outboard engines at full power (approximately 25 in Hg/2200 RPM) the aircraft developed an average ROC of 120 ft/min at 3000 ft pressure altitude and at an ambient temperature of 7 deg.C. The airspeed was 65 KIAS and the AUW was approximately 4930 kg. The graph at Figure 3 of Annex J details climb performance as a function of pressure altitude, ambient temperature and aircraft gross weight but for an airspeed of 60 KIAS which was found to be more suitable operationally. The graph indicates a ROC of 140 ft/min for a twin-engine climb with the aircraft at maximum AUW under ISA sea level conditions. The full power performance of the operating engines, and consequently the engine out rate of climb, decreases with increasing pressure altitude. Although minimal, the engine-out rate of climb at 60 KIAS below 4000 ft was satisfactory. Careful pre-flight planning to account for low engine-out rates of climb will be needed when operating the aircraft at high density altitudes.

6.9.6 Engine-Out Circuit and Landing. The circuit, landing and overshoot characteristics with two engines operating normally and the third at idle were evaluated with the aircraft at an AUW of approximately 4900 kg. For the purposes of the evaluation an engine at idle was considered to be representative of a failed engine. The aircraft's performance with an engine at idle reduced accordingly but it displayed no significant adverse handling characteristics. Simulated engine failures after take-off were carried out from 70 KIAS at 400 ft AGL by closing a throttle from full power to idle in approximately one second. The aircraft was easy to control about all three axes and required a maximum of 50% right rudder deflection with a rudder force of approximately 80 lbf when the port throttle was closed. The rate of climb reduced from 750 ft/min to 180-200 ft/min. The remainder of the circuit was easy to fly and the performance was adequate, although power settings up to 24 in Hg/2050 RPM were required to maintain 80 KIAS on the downwind leg. A twin engine overshoot, using full power on the centre and starboard engines, was initiated at 300 ft AGL on finals. Height loss was approximately 40 ft before

the aircraft established a positive rate of climb at 65 KIAS. Landing with the critical engine at idle provided no handling difficulties although, during the trial, this was only carried out on a long wide runway under favourable wind conditions. Extreme care would be required if an asymmetric landing was to be attempted on a narrow runway with a gusting crosswind from the same side as the inoperative engine. The engine-out circuit and landing characteristics were satisfactory although asymmetric landings should preferably be attempted within the most favourable wind/runway conditions available at the time.

7. CONCLUSION

7.1 Technical Investigation 953 was carried out to evaluate the performance and handling qualities of the replica of the 1926 Fokker Tri-Motor, 'Southern Cross'. As a replica of a Fokker VIIb-3M, the aircraft was professionally constructed and well presented. It was a valuable and unique aircraft which meant that the flight envelope was only investigated to the extent where safe flight characteristics could be ensured with the aircraft being operated in daylight VMC conditions. The fundamental design of the original aircraft was followed, and consequently the replica displayed basic performance and handling characteristics which were somewhat akin to, but probably better than those which would have been expected from the original. The aircraft was generally easy to fly although it required the use of a 'hands on' technique. Aircraft handling during take-off and landing, especially in gusty or crosswind conditions, displayed some potentially dangerous characteristics. However, if the aircraft is flown within the recommended limits by suitably experienced pilots these problems should be averted. While unable to completely satisfy the certification conditions of Reference B the test aircraft was airworthy for daylight VFR operations under the general requirements for normal category aircraft. Some form of limited Certificate of Airworthiness may be appropriate.

8. RECOMMENDATIONS

8.1 The following recommendations should be taken into account when considering the aircraft for certification.

8.2 Highly Desirable.

8.2.1 The maximum allowable crosswind component for take-off or landing on runways of less than 90 ft in width should be 5 knots. For wider runways a crosswind limit of up to 14 knots could be applied. Particular caution should be exercised in all instances where strong or gusting winds are present during take off or landing (Paragraph 6.7.3).

8.2.2 Cockpit occupants should wear helmets when operating the aircraft (Paragraphs 6.1.5.a and 6.1.6).

8.2.3 The aircraft should be operated by two pilots, although a single pilot and a competent observer with extensive aviation experience may be satisfactory for short duration sorties (Paragraphs 6.1.3, 6.1.4, 6.1.6, 6.3.1 and 6.5.2).

8.2.4 Due to the lack of available nose-down tailplane trim the centre of gravity range should be limited to between 22.6% MAC (which equates to the forward design limit) and 29.0% MAC (which equates to 60% of the design range) (Paragraph 6.5.4.c).

8.2.5 Mechanical stops should be incorporated into the tailplane adjustment system to limit tailplane movement to between 50% and 100% of the available nose-down range (Paragraph 6.5.4.d).

8.3 Desirable.

8.3.1 Take-off and landing distances should be based on the data presented at Annex I. When landing on runways of 1500 metres or less a last point of touchdown no further than 300 metres beyond the runway threshold must be adopted and thus a realistic runway length for landing should be taken as a fixed air distance of 300 metres plus the required ground roll distance presented in Figure 4 at Annex I (Paragraph 6.7.4).

8.3.2 Intentional flight in strong or gusty wind conditions should be avoided (Paragraphs 6.5.8 and 6.7.3).

8.3.3 The performance data presented at Annexes J and K should be used for flight planning and aircraft operation (Paragraphs 6.8.1 and 6.8.2).

8.3.4 Pilots should operate the aircraft with respect to indicated airspeed (IAS), although calibrated airspeed (CAS) should be used for navigational flight planning (Paragraph 6.4.3).

8.3.5 Conservative fuel reserves should be carried and the calculation of these reserves should be based on an accurate pre-flight fuel dip (Paragraph 6.1.5.b).

8.3.6 The operational airspeeds indicated at Paragraph 6.4.4 should be marked on the airspeed indicators (Paragraph 6.4.4).

8.3.7 The operational manoeuvre speed (Va) should be changed from 102 KEAS to 90 KEAS (Paragraph 6.4.4, Appendix 1 to Annex F).

8.3.8 Careful preflight planning to account for low engine-out rates of climb should be carried out prior to operations at high density altitudes (Paragraph 6.9.5).

8.3.9 During engine ground runs either both pilot seats should be occupied or the aircraft should be attended by a ground observer to enable the continuous observation of the propeller arcs (Paragraph 6.1.3).

8.3.10 Asymmetric landings should preferably be attempted within the most favourable wind/runway conditions available at the time (Paragraph 6.9.6).

8.3.11 Hearing protection should be provided for passengers (Paragraph 6.1.7).

8.3.12 Cockpit decals should be treated, or replaced, in order to prevent deterioration (Paragraph 6.1.4).

9. REFERENCES

A. HQSC AIR4/4082/03/953 Pt 1 (6), 'Flight Testing of the Southern Cross Replica', Technical Investigation No 953, 6 March 1987.

B. United States Federal Aviation Administration, FAR 23, 'Airworthiness Standards: Normal, Utility and Acrobatic Category Aircraft', June 1975.

C. Page Industries of Oklahoma, Operators Manual, Jacobs Aircraft Engine Models R-755A, R-755B and R-755S, 1974.

D. Royal Air Force, Empire Test Pilots' School, Fixed-Wing Test Pilot Course Lecture Notes, January 1984.

E. United States Navy Test Pilots' School, Flight Test Manual Nos 103 & 104, January 1975.

F. Australian Government, Department of Transport Report, AF-47, August 1979.

10. PROJECT PERSONNEL

Project Officer: SQNLDR N.G.Coulson

Project Pilot: FLTLT N.G.Lindorff

Project Engineer: Mr G.Readett

Project Engineer: FLGOFF J.W.Blagg

DESCRIPTION OF THE TEST AIRCRAFT

1. Fokker Tri-Motor Replica. The aircraft is a full-size replica of a 1926 Fokker Tri-Motor (F.VII/3m) which was manufactured at Parafield Airport by Famous Australian Aircraft Ltd. The basic dimensions and form of the original aircraft have been followed although the engineering is to the standard required by Reference A. The aircraft (see Figure 1) is a high-wing light transport aeroplane powered by three Jacobs 755-A1 seven-cylinder radial engines; one mounted in the nose and the other two under the inboard wing sections. The maximum all-up-weight of the aircraft is 5700 kg. The aircraft alights on fixed tailwheel configuration landing gear and has conventional mechanical flight controls comprising ailerons, rudder and elevator, which is itself mounted on an adjustable tailplane. There are no wing flaps. The cockpit has a standard dual control layout for side-by-side pilot and co-pilot.

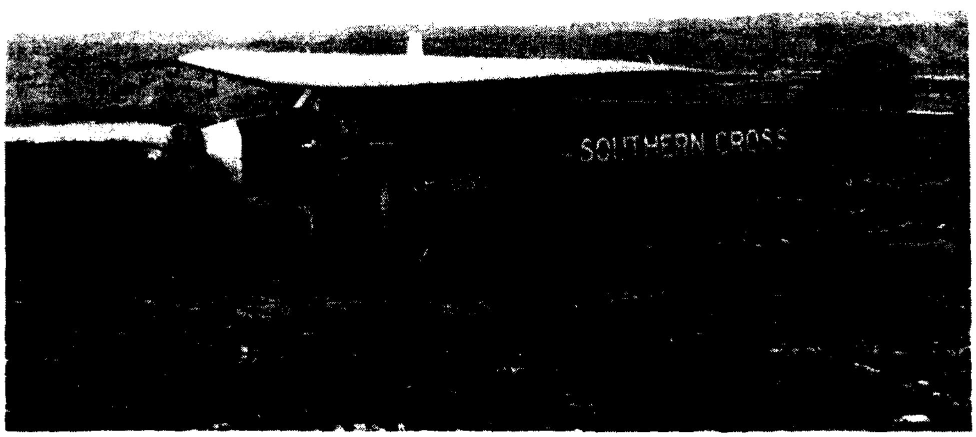


Figure 1

2. Engines. The aircraft is powered by three Jacobs Model R-755A1 engines. The R-755A1 is an unsupercharged seven-cylinder air-cooled radial engine with a piston displacement of 757 cubic inches. The engines are numbered one to three from left to right. A complete description of the engine and its operating instructions is contained in Reference C.

a. General Assembly. The engine crankcase consists of five aluminium castings which carry the cam gear and pinion assembly, tappet assemblies and the integrally-cast intake manifold and oil sump. The chrome molybdenum steel forged cylinder barrels contain aluminium alloy pistons mounted on a two-piece crankshaft.

b. Carburettors. Mounted at the bottom rear half of the engine power section is an updraft Stromberg carburettor (Type NA-R7A) which is of the single venturi type fed from a single float chamber. The carburettor is equipped with a needle-type manual mixture control operated via a cockpit pedestal-mounted lever (Figure 2). Engine power is adjusted by a manual throttle which is also operated by a cockpit lever. A carburettor heating facility is provided and this is operated by a three-position lever mounted on the front face of the cockpit pedestal.

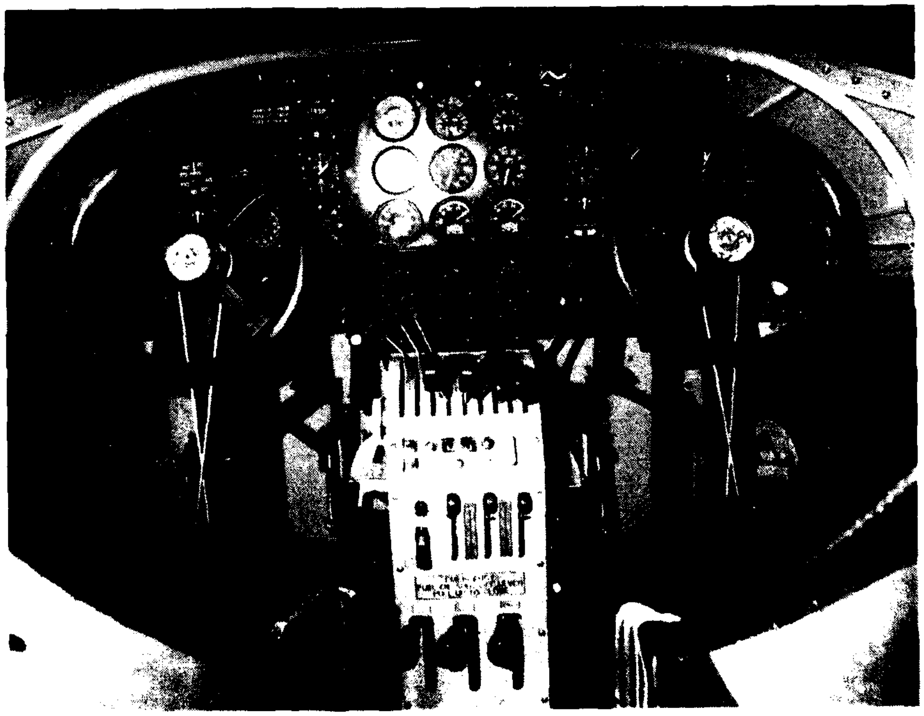


Figure 2

c. Ignition System. The engine is fitted with a magneto-battery ignition system. The rear bank of spark plugs is powered by a magneto while the front bank receives its energy from the aircraft battery and the engine-driven alternator through a distributor and coil. The battery ignition provides improved starting and idling by providing a high voltage fully retarded spark at low engine speeds. A coil ignition master switch with an associated red light is mounted on the cockpit centre overhead panel (Figure 3). Also on this panel each engine has a four position standard ignition switch which allows the selection of either or both ignition systems. These switches have the following selections:

(i) Both - both ignition systems are active.

(ii) Left - only the magneto ignition system is active.

(iii) Right - only the battery/coil ignition system active.

(iv) Off - both systems are grounded.

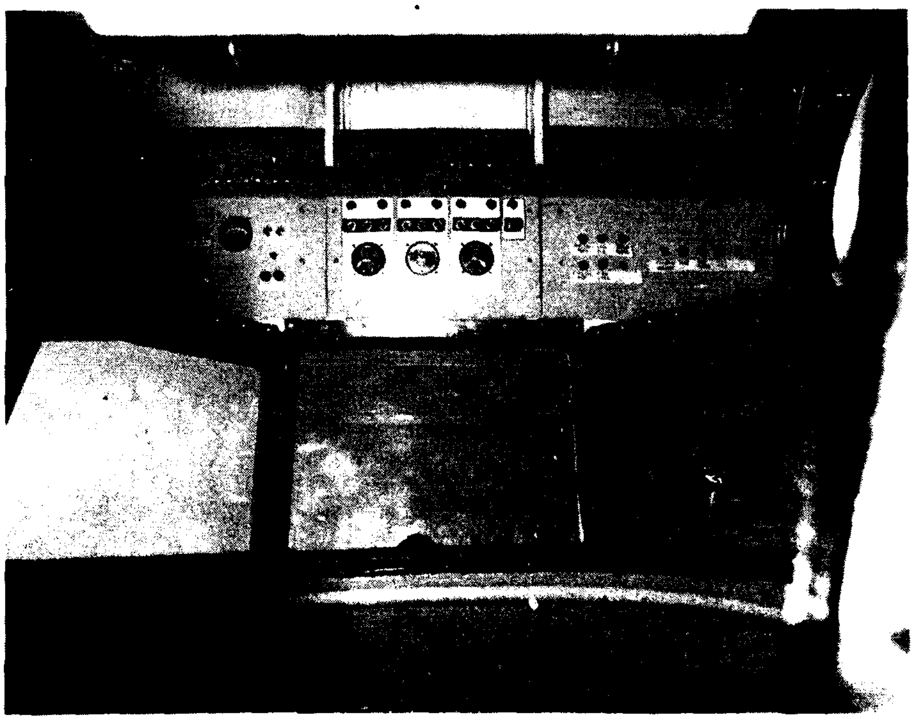


Figure 3

Above each engine ignition switch panel there is a green coil ignition available light which illuminates whenever the coil ignition for that engine is active. Next to these lights are three guarded alternate ignition switches which allow connection of battery power directly to the engine ignition.

d. Lubrication. The lubrication system consists of a tank, pressure/scavenge oil pump, filter screens and check valves. It provides pressure or splash lubrication to all engine moving parts. High pressure oil is also provided for propeller governing.

e. Accessories. Engine accessories are mounted on the rear crankcase. These include the starter, fuel pump, alternator, oil pump, magneto, distributor, vacuum pump and propeller constant speed unit. The starter is activated using a start engage switch mounted on the associated engine overhead panel. A green light above each switch indicates when the starter is engaged.

f. Firewall Shut-Offs. Each engine is provided with a firewall shut-off lever mounted on the front face of the cockpit centre pedestal. When this lever is pulled up into its detent all fuel and oil is isolated from the engine at the firewall.

3. Propellers. Each engine is fitted with a Hamilton Standard 2B/20 constant speed propeller controlled by cockpit pedestal mounted levers. The 93 inch diameter propellers are governed to fine pitch by engine oil pressure while a counter-weight system pulls the blades to course pitch in the event of a loss of oil pressure. When an engine is shut down or fails while airborne, the propeller will move toward the coarse pitch position and stabilise such that it windmills at approximately 850 RPM at an airspeed of 70 KIAS.

4. Fuel System. The aircraft runs on Aviation Gasoline. Grade 100LL is preferred but grades 100/115 or 100/130 may also be used.

a. Fuel Tanks. The aircraft has four 90 gallon fuel tanks mounted in the wing. Tank 1 feeds directly to the left engine while Tank 4 feeds the right engine. Tanks 2 and 3 are interconnected and feed the centre engine. Tank 2 is the primary tank being connected directly to the engine while Tank 3 is considered an auxilliary and is connected to Tank 2 via a non-return valve. It is normally only filled when a long-range flight requirement exists. Each tank has two vertical and three horizontal baffles to minimize fuel sloshing and to help provide tank rigidity. Refuelling is carried out to each tank individually through its own overwing gravity feed filler.

b. Fuel Flow. Fuel flows from a primary tank (ie. Tanks 1, 2 and 4) through a non-return valve to the fuel shut-off valve, thence via the firewall shut-off to a boost pump and then to the engine driven fuel pump. Fuel filtering takes place through a mesh

filter at the tank outlets and through the primary fuel filter downstream of the firewall shut-off valve. Cross-feed is available between the three main flow paths with two cross-feed valves present. The three main fuel shut-off valves and the two cross-feed valves are mechanically operated by knobs on the fuel control panel which is on the rear cockpit bulkhead above, between and behind the pilots. Normal fuel selections are as follows:

(i) Take-off and landing - main shut-off valves open and cross-feed valves closed. This ensures that any fuel contamination from an individual tank affects its associated engine only.

(ii) Normal cruise - as above or the cross-feed valves may be opened to allow fuel levels to balance.

(iii) Engine-out cruise - associated cross-feed valve closed. If fuel from the affected tank is required then the firewall shut-off of the affected engine should be closed before the cross-feed is re-opened.

c. Boost Pumps. An electric boost pump is situated downstream from each primary fuel filter. Each boost pump is operated by a cockpit switch on its associated overhead engine panel. A blue light above each switch indicates when the boost pump is selected on. Each engine has a prime switch which is operated in conjunction with the boost pump prior to engine start and allows fuel to be fed to the top cylinders for the start.

d. Fuel Quantity Indication. Fuel quantity is indicated by three sight glasses mounted on the rear cockpit bulkhead above the fuel control panel. Each sight glass gives a direct reading of the amount of fuel remaining in its associated tank with unusable fuel being indicated by the first five millimetres from the bottom of the glass. A shut-off valve is included at both the top and bottom of each glass. The bottom valve also incorporates a ball valve designed to shut off fuel flow to the sight glass in the event of breakage.

5. Flight Controls. The aircraft is fitted with conventional mechanical flight controls consisting of an adjustable horizontal tailplane, elevator, ailerons and rudder. These controls are operated from either pilot seat by dual interconnected control wheels and rudder bars (Figure 4). There is no provision for the adjustment of cockpit seat or control positions.

a. Elevator. The one-piece elevator is horn and mass balanced. It is connected to the tailplane and fuselage by a single bolt but is also supported by tailplane struts and rigging wires. The elevator is connected to the pilot's controls by dual externally rigged cables attached to horns and torque tubes. The system has a large amount of inherent mechanical friction.

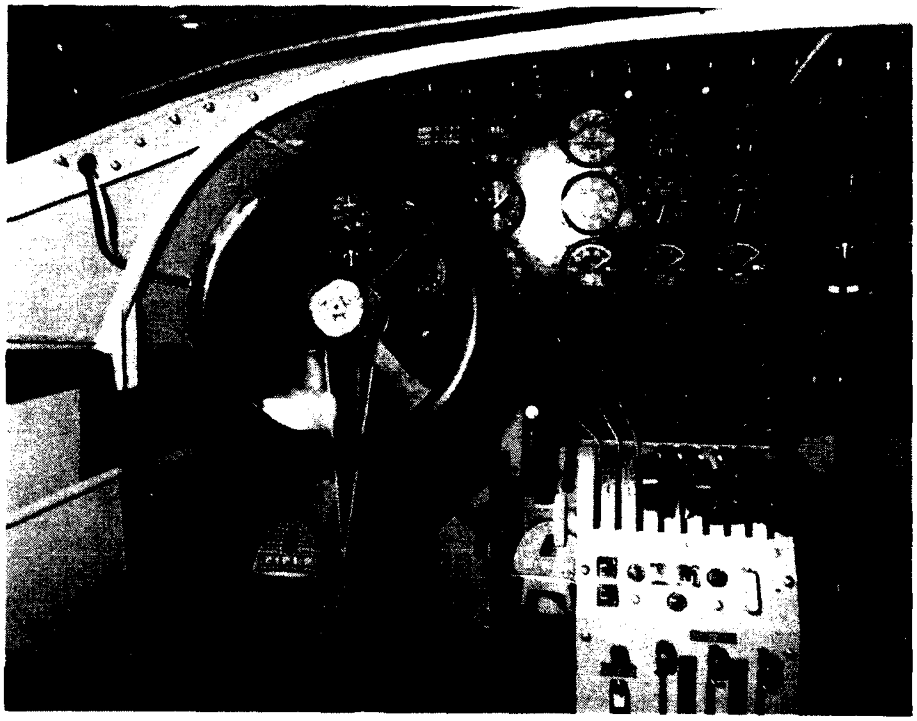


Figure 4

b. Tailplane. The aircraft is equipped with an adjustable horizontal tailplane composed of two halves mounted on a torque tube and attached to the fuselage and elevator by a single bolt. The tailplane is also supported by struts and rigging wires. It is adjustable through a range of two degrees leading edge down to four degrees leading edge up. The one degree leading edge up position is considered neutral. The pilot operates the tailplane by rotating a trim wheel mounted on the cockpit floor beside the left-hand pilot seat. This wheel operates in the natural sense; rotating the wheel forward trims the aircraft nose-down and vice versa. The trim wheel is connected to a chain and layshaft and thence to a torque tube which operates a gear box at the tailplane. This gear box contains a screw jack which adjusts the tailplane leading edge. To prevent the tailplane moving under air loads the screw jack is locked. When the pilot operates the trim wheel, micro-switches sense changes in the trim wheel chain tension and unlock the tailplane screw jack. An orange light on the cockpit pedestal illuminates when the screw jack is unlocked. The system is fail-safe in that the tailplane will remain locked if there is a loss of electrical power. There is also a manual override switch, mounted on the pedestal, which enables opening of the screw jack lock in the event of micro-switch failure. A scaled indicator on the pedestal shows the position of the trim system.

c. Ailerons. The aircraft has conventional mass balanced ailerons mounted on the outboard section of the wing trailing edges. They are operated through a cable system interconnected to the cockpit control wheels. This also creates a large amount of mechanical friction. There is no aileron trim system.

d. Rudder. A horn and mass balanced rudder is mounted on the vertical fin. The rudder is connected to the cockpit rudder bars via a single cable which also incorporates a spring bias trim system. The rudder bars have pedals mounted on a paralleling sytem such that the pedals remain parallel as the bar pivots. Toe operated brake sections are also fitted to the pedals. The rudder trim consists of an electrically-operated arm which changes the tension in a spring system thereby changing the loads felt at the pilot's pedals. The trim system is operated using a double-pole switch mounted on the cockpit pedestal below the mixture levers. A scaled indicator, also on the pedestal, shows the position of the trim.

6. Landing Gear. The aircraft has a fixed tailwheel type landing gear system. The two large main wheels are mounted on an oleo strut slung beneath the outboard engines. The wheels themselves have a degree of inward cant and therefore the tyres are susceptible to wear when the aircraft is operated on hard rough surfaces. The fully castoring tailwheel is provided with a locking system which allows it to be locked in the fore/aft position for take-off, landing and prolonged straight-line taxiing. The tailwheel lock is operated by a lever mounted on the cockpit floor behind the left-hand pilot seat. Pulling the lever up will unlock the tailwheel. Pushing the lever down will allow the spring-loaded locking pin to fall into position once the tailwheel has been aligned with the fore/aft axis of the aircraft. The locking lever itself can be locked in the castoring position by turning it through 90 degrees. It is essential that the aircraft be taxied straight forward several metres each time the tailwheel is selected to the locked position to ensure that the locking pin is in position. This is especially important prior to take-off as the aircraft will be overly sensitive to ground-looping during take-off or landing with the tailwheel unlocked.

7. Brakes. The two main wheels are fitted with hydraulic disc brakes which are operated using toe pedals on the rudder bars at each pilot station. The brakes can be applied together or differentially. A park brake function is also provided and this is applied using a handle below the left-hand instrument panel.

8. Electrical System. The aircraft has a 28 volt DC negative earth electrical system which provides power for the coil ignition and other aircraft accessory systems.

a. Batteries and Main Bus System. Four lead-acid batteries, grouped in two pairs, feed the main electric bus (which, in reality, consists of two battery busses joined in series). A battery master switch located on the left-hand cockpit overhead panel allows the selection of either the batteries or an external power unit to the main bus. An associated green light illuminates when power, from any source, is connected to the bus. A voltmeter

indicates main bus voltage. Two battery - isolation switches, guarded such that batteries are normally connected to the bus, are also located on the left-hand overhead panel and allow the isolation of either battery pair in the event of an internal fault.

b. Ground Power. An external power unit of suitable voltage and polarity can be connected to a receptacle on the left engine. A blue ground power available light on the overhead panel illuminates if external power of the correct polarity is connected (but not necessarily switched on). When ground power is selected at the battery master switch the external power unit is connected to the main bus.

c. Alternators. Two alternators, driven by each of the outboard engines, are provided to power the main bus when the engines are operating at airborne RPM settings. The alternators are excited by their respective batteries and internally rectified. They will normally come on line at approximately 1800 RPM but will not drop off line until the engine reduces to below 1400 RPM. Two switches located on the overhead panel and labelled GEN 1 and GEN 3 allow selection of the alternators by providing or interrupting the excitation supply between the regulator and the alternator. Two ammeters indicate the amperage provided by the alternators to the bus. With both alternators selected the bus load requirements will determine whether either or both are on line simultaneously. Two alternate excitation switches, guarded in the off position, are provided to allow for an alternate alternator excitation source should one battery pair be isolated.

d. Circuit Breakers. Circuit protection for electrical system components and services are located on the cockpit overhead panel and are all appropriately labelled. Additionally, there are two 50 amp alternator output circuit breakers on the lower instrument panel beneath the clock and two heavy duty bus protection circuit breakers located in the aft electrical bay.

9. Flight Instruments. Each pilot station is provided with a full set of flight instruments.

a. Pitot Static System. Two independent pitot static systems are provided. A pitot static head is mounted under each outboard wing section; the port head supplies the left-hand pilot station instruments and vice-versa. A single pitot heat switch, located on the overhead panel, allows selection of 28 volt heat to the pitot heads. Two orange lights will indicate when the pitot heat has been selected on and power is available to the associated head. Alternate static source switches are located on the lower instrument panels and allow the selection of local (cabin) pressure as an alternate static source to the selected system. The pitot static system supplies an altimeter, an airspeed indicator (ASI) and a vertical speed indicator (VSI) on each instrument panel.

b. Vacuum System. There are two independent vacuum systems used to power the gyro-driven instruments. The left-hand system is supplied by two vacuum pumps located on the left and centre engines. The right-hand system is supplied by a pump on the right engine and by a venturi source located on a right-hand engine mounting strut. Two vacuum instruments, located on the lower right instrument panel, indicate manifold pressure and also have pressure sensitive warning flags which indicate the failure of the associated vacuum source. The gyro-driven instruments are two artificial horizons (AH) and two turn and slip gauges mounted on both instrument panels and a single directional gyro (DG) mounted only on the right-hand side.

c. Compasses. A remote indicating magnetic compass (with its transmitter mounted in the left wing) is provided for the left instrument panel. The right instrument panel has a vacuum-driven directional gyro. A standby magnetic compass is mounted on the floor under the centre pedestal.

d. Clock. An electric clock is mounted on the lower right of the centre instrument panel.

10. Engine Instruments. A full set of engine instruments are provided. They are mounted vertically in rows of three (for each engine) on the centre instrument panel.

a. Triple Gauge. Engine oil temperature, oil pressure and fuel pressure are all indicated on a single gauge. Oil temperature is sensed electrically at the oil tank outlet. Oil and fuel pressures are sensed directly at the rear of the instrument via lines fitted with restrictor orifices.

b. Tachometer. Engine RPM is indicated on a tachometer driven by an engine-mounted tacho-generator.

c. Manifold Pressure. Manifold pressure (MAP) is indicated by a direct reading pressure gauge.

d. Cylinder Head Temperature. Cylinder head temperature (CHT) is measured by a thermocouple system.

e. Carburettor Air Temperature. Carburettor air temperature (CAT) is measured by a 28 volt thermistor located in the carburettor throat.

11. Radio and Intercom. The aircraft is fitted with a single VHF radio and an intercom system for cockpit and crew communications.

a. VHF Radio. The VHF-20A transceiver is located in the lower equipment bay with the controller mounted on the left-hand instrument panel. The antenna is mounted on the mid-wing upper surface. The range of selectable frequencies is from 117 - 135.95 MHz.

b. Intercom. The intercom panel is mounted on the right-hand instrument panel. It has a row of switches which allow reception of incoming signals; the No 1 switch is for the VHF and the INT switch is for the intercom, the remainder are inoperative. There is a large rotary selector used for transmit functions; when set at INT the intercom is available and when set at No 1 the VHF is available but the intercom is disabled. The remaining positions are inoperative. A volume knob allows the adjustment of incoming signal strength. Each pilot station has an intercom jack and a press-to-transmit (PTT) button and there are also two intercom leads available in the cabin. A hot mike switch is located on the lower left instrument panel and allows all microphones to be selected constantly open (for intercom only). There is also a microphone select switch which allows the selection of pilot only, co-pilot only or all microphones.

12. Lights. External aircraft lighting includes standard navigation lights and a pair of wingtip strobe lights. These are individually selected using switches on the left overhead panel. Internal lighting is available in both the cockpit and cabin from a series of individual white flood-lights.

FLIGHT TEST SUMMARY

Flight No	Date	Weight (kg)	CG %MAC	Test Description	Remarks	Elapsed Time (hrs)
1	14AUG87	4791	28.6	Control, Handling & Performance	Inaugural Flight	0.7
2	17AUG87	4660	28.4	Ferry - PEC		0.4
3	24AUG87	5029	27.9	PEC		0.6
4	25AUG87	4923	27.8	PEC		0.8
5	27AUG87	4879	27.7	Stability & Control	Mid/Rear CG	1.0
6	29AUG87	4692	25.5	Stability & Control	Mid/FWD CG	0.9
7	29AUG87	4500	23.2	Stability & Control	FWD CG	0.4
8	01SEP87	4804	26.5	Stalls	MID CG	0.7
9	01SEP87	4731	26.4	Stalls - Asymmetric Handling		1.2
10	02SEP87	4528	22.7	Stalls	FWD CG	0.6
11	03SEP87	4718	26.9	Performance		0.7
12	04SEP87	4528	22.7	Stability & Control - Stalls	FWD CG	1.0
13	10SEP87	5093	28.9	Stability & Control - Stalls	Rear CG	1.5
14	10SEP87	4950	28.7	Asymmetric Performance & Handling	Rear CG	1.4
15	11SEP87	4532	23.3	Performance	Light Weight	1.8
16	14SEP87	4543	23.3	D of Transport Famil		0.6
17	15SEP87	-	-	D of Transport Famil	2 Flights FWD/ REAR CG	1.7
18	16SEP87	5706	28.3	Performance	Heavy Weight	2.3
					Total Flight Test Time	18.3

PERMIT TO FLY

XEROX TELECOPIER 495-1; 7- 7-87; 9:08AM ; 0221142414 062485239;# 4

DEPARTMENT OF TRANSPORT

PERMIT TO FLY

...suant to regulation 108A of the Air Navigation Regulations, permission is ...by granted for the flight of FOKKER VIIB/3M REPLICA aircraft ...Marking or Serial No. VH-USU on or before 10/9/87 ...the purpose of CERTIFICATION FLIGHT TESTING PURSUANT TO ANR ...8A (1)(b).

...accordance with the following directions:—

ALL FLIGHTS TO BE CONDUCTED IN VMC AND IN ACCORDANCE WITH ATC PROCEDURES.
PILOT IN COMMAND TO BE SQN. LDR. N. COULSON, OTHER ...LOTS TO BE ACCEPTABLE TO FLYING OPERATIONS BRANCH SA/NT ...GN. CREW TO WEAR PARACHUTES ON ALL FLIGHTS.
FLIGHT TESTING TO BE CONDUCTED IN ACCORDANCE WITH ...FLIGHT TEST SCHEDULE AGREED WITH D of A.
FLIGHTS ARE TO BE CONDUCTED WITHIN THE CONFINES ...F THE PARAFIELD TRAINING AREA AND THE EDINBURGH ...ONTROL ZONE.
A MAINTENANCE RELEASE SHALL BE ISSUED PURSUANT TO ...R 48(6)(b) AND EXPRESSED TO REMAIN IN FORCE FOR 100 ...OURS OR THE DURATION OF THIS PERMIT, WHICHEVER IS LESSER.
FLIGHT OVER BUILT UP AREAS IS PROHIBITED.
SMOKING IS PROHIBITED.
AEROBATIC MANOEUVRES ARE PROHIBITED.
THE AIRCRAFT SHALL BE MAINTAINED IN ACCORDANCE WITH

...ereby direct that the following provisions of regulations 108 and 113 of ...Air Navigation Regulations are not applicable to such flight:—
...08 (1)(b), (c).

... 30/7/87 Signed [signature]
*Delegate of the Secretary ~~*Authorised Person~~

DEPARTMENT OF TRANSPORT

PERMIT TO FLY

Pursuant to regulation 108A of the Air Navigation Regulations, permission is hereby granted for the flight of FOKKER VIIB/3M REPLICA aircraft Reg. Marking or Serial No. VH-USU on or before 10/9/87 for the purpose of CERTIFICATION FLIGHT TESTING PURSUANT TO ANR 108 A (1)(b).

In accordance with the following directions:—

9 (cont) ANO's 100.5.0 AND 100.5.1.
10. THE FOLLOWING OPERATIONAL LIMITATIONS SHALL APPLY
NEVER EXCEED SPEED VNE – 113 KIAS
MANOEUVRE SPEED VA – 102 KIAS
MAXIMUM TAKE OFF WEIGHT – 5700 KG.
CENTRE OF GRAVITY LIMITS
FORWARD 1050 mm AFT OF DATUM
AFT 1380 mm AFT OF DATUM
DATUM – WING ROOT LEADING EDGE
11. ESSENTIAL FLIGHT CREW ONLY TO BE CARRIED
12. ENGINES ARE TO BE OPERATED TO THE LIMITATIONS OF THE ENGINE OPERATORS MANUAL
13. ANY DEFECTS ARE TO BE NOTIFIED TO AIRWORTHINESS SA/NT REGION WITHIN 48 HOURS.
14. A COPY OF THIS PERMIT IS TO BE CARRIED ON ALL FLIGHTS.

I hereby direct that the following provisions of regulations 108 and 113 of the Air Navigation Regulations are not applicable to such flight:—
108 (1)(b), (c).

Date 30/7/87 Signed [signature]
*Delegate of the Secretary ~~*Authorised Person~~

RECOMMENDED CENTRE OF GRAVITY RANGE

Reference:

D-1 Southern Cross Replica - Static Structural Design By C.W. Whitney, 5 October 1981

1. The position of the centre of gravity (CG) of the aircraft can be defined in reference to either;

   a. a fixed physical datum on the aircraft,

   b. a percentage of the design range (Reference D-1), or

   c. a percentage of the mean aerodynamic chord (MAC).

2. Table 1 relates the recommended forward and aft limit positions of the CG. The fixed datum is taken as the centreline of the front main spar and corresponds to an x-z plane 25 mm forward of the aft face of the main cabin bulkhead. Appendix 1 to this Annex derives the position of the CG in terms of percentage MAC.

TABLE 1 - CENTRE OF GRAVITY LIMITS

Ser.	Limit	CG Position		
		Aft of Fixed Datum (mm)	Design Range (%)	MAC (%)
(a)	(b)	(c)	(d)	(e)
1	Forward	400	0	22.6
2	Aft	600	60	29.0

3. Appendix 2 to this Annex presents a load sheet for use when operating the aircraft. This will be included in the Flight Manual and is based on the TI 953 recommended CG limits. It should be noted that with only the pilot and co-pilot onboard the aircraft, and with no passengers or baggage, the CG will move forward of the forward limit when fuel remaining reduces to less than 560 litres.

DERIVATION OF CG IN TERMS OF % MAC

Reference:

D-1 Southern Cross Replica - Static Structural Design By C.W. Whitney, 5 October 1981

1. The location of the mean aerodynamic chord was not contained in the extracts of Reference D-1 available to ARDU and therefore it has been derived from the following:

MAC c = 10.179 ft (from Ref D-1, p2-1)
= 122.15"
= 3102.6 mm

A.C. of wing = 0.217 $\bar{c}$
= 0.217 x 3102.6
= 673.3 mm

Ref D-1 also gives A.C. as 40.3" (1023.6 mm) aft of wing leading edge therefore;

L.E. of MAC is 1023.6 - 673.3 = 350.3 aft of wing L.E.

Ref D-1 gives the design CG range as 1050 to 1380 mm aft of wing L.E. which corresponds to 400 to 730 mm aft of the centreline of the main spar.

Therefore: Main spar C/L ≈ 1050 - 400
≈ 650 mm aft of wing L.E.
≈ 650 - 350.3
≈ 299.7 mm aft of L.E. of MAC.

$$\text{Therefore CG in terms of \%MAC} = \frac{X + 299.7}{3102.9} \times 100$$

where X = CG position in mm aft of C/L main spar datum.

LOADING SYSTEM - SOUTHERN CROSS - VH-USU

ITEM	ITEM WEIGHT (kg)	WEIGHT (kg)	ARM (mm)	MOMENT (kgmm)/1000
BASIC AIRCRAFT	3911	3911	+ 415	+ 1623
OIL (1 + 3)	(1 + 1)x0.9		+ 390	+
OIL (2)	(1)x0.9		-1880	
PILOTS	77	154	- 390	- 60
SEATS (ROW 1)	7		+ 730	
PAX (ROW 1)	77		+ 580	
SEATS (ROW 2)	7		+1515	
PAX (ROW 2)	77		+1365	
SEATS (ROW 3)	7		+2270	
PAX (ROW 3)	77		+2120	
SEATS (ROW 4)	7		+3010	
PAX (ROW 4)	77		+2860	
LUGGAGE	70 MAX		+4395	
ZERO FUEL TOTAL	*****************		******	
ZERO FUEL CG	(1)x0.71		+ 725	+
MAX FUEL WT	*****************		******	
TOTAL CG ARM				

CG RANGE 400 - 600 mm

Therefore CG as a % of range = $\frac{(\text{Total Arm} - 400)}{200} \times 100$

= %

Datum for ARM - Centreline of front spar. This corresponds to a plane 25 mm forward of the aft face of the cabin main bulkhead.

WARNING - With pilot and co-pilot onboard and no pax or baggage, the CG will be outside the forward limit when fuel remaining is less than 560 litres.

PREPARED BY.............................DATE....................

PILOT IN COMMAND........................DATE....................

PRESSURE ERROR CORRECTION RESULTS

1. DEFINITION OF AIRSPEEDS

TAS - True Airspeed
EAS - Equivalent Airspeed = TAS $\sqrt{\text{Relative Density}}$
CAS - Calibrated Airspeed. Over the envelope of this aircraft CAS = EAS
RAS - Rectified Airspeed, RAS + PEC = CAS
IAS - Indicated Airspeed as seen on ASI, IAS - Instrument error = RAS
K - Preceeding above abreviations indicates knots.

2. ASI CALIBRATION

The following table contains the ASI calibration data dated 28 June 1987 as supplied by the FAA.

	Pilot ASI S/NO 74336		Co-pilot ASI S/NO 217740	
Actual (Kn)	Indicated (Kn)	Error (Kn)	Indicated (Kn)	Error (Kn)
30	30	0	30	0
40	39	- 1	39.5	- 0.5
50	53	+ 3	48	- 2
60	62	+ 2	58	- 2
70	70	0	69	- 1
80	79.5	- 0.5	79.5	- 0.5
90	89	- 1	88	- 2
100	99	- 1	98	- 2
120	119.5	- 0.5	118	- 2
140	140	0	141	+ 1

NOTE: Error = Indicated - Actual

3. ASI SYSTEM PEC

The reduced flight test data is presented in Figures 1,2 & 3 and includes the test points to indicate the scatter of the test results. Much of the scatter resulted from having to test in other than ideal calm conditions due to the short time scale available for the flight test program. The lines of best fit apply to the combined data for both the pilot and co-pilot independent systems. The mean aircraft all up weight for the PEC runs was 4950 kg.

4. USE OF PEC GRAPHS

Note that the graphs do not include PEC data against IAS as this graph would change if the instruments were adjusted during calibration or were replaced. The recommended method of use:

To obtain KTAS from KIAS

a. Read KIAS from ASI,

b. Correct KIAS for individual instrument calibration to get KRAS,

c. From Figure 3 read off KCAS corresponding to KRAS, and

d. Apply normal altitude density correction to KCAS to get KTAS.

ASI PRESSURE ERROR CORRECTIONS - PEC Vs RAS

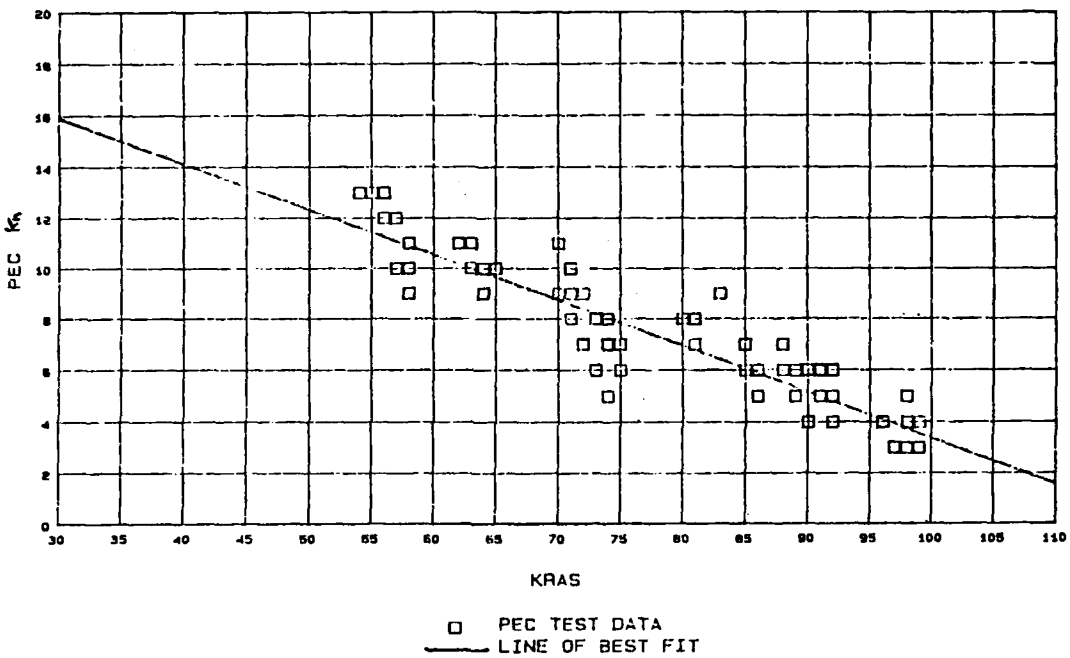


Figure 1

ASI PRESSURE ERROR CORRECTIONS - PEC Vs KCAS

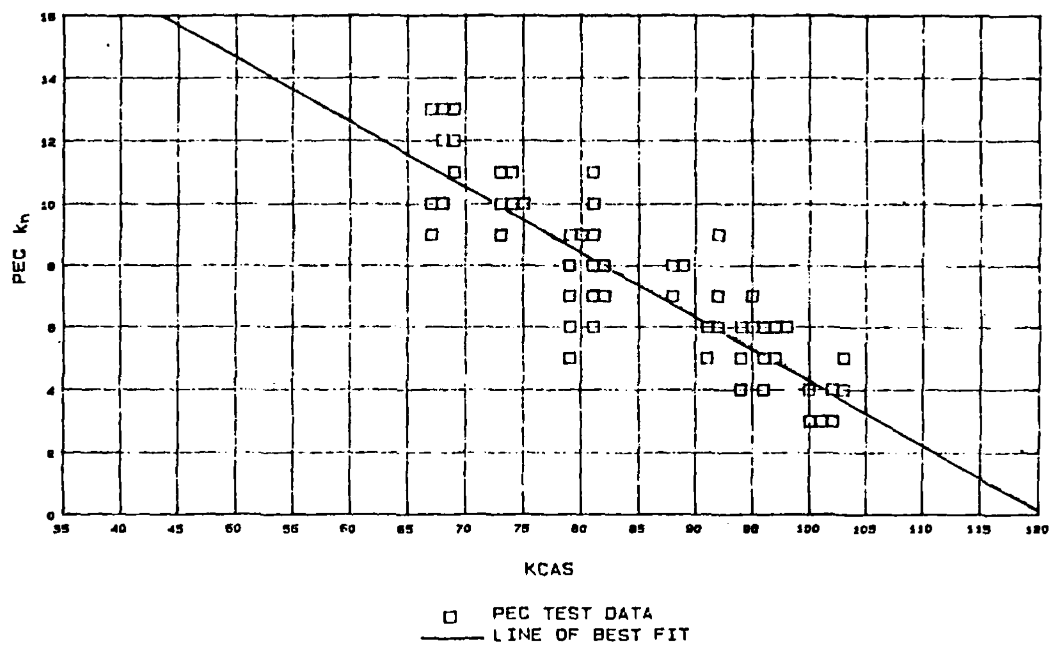


Figure 2

ASI PRESSURE ERROR CORRECTIONS - KCAS Vs KRAS

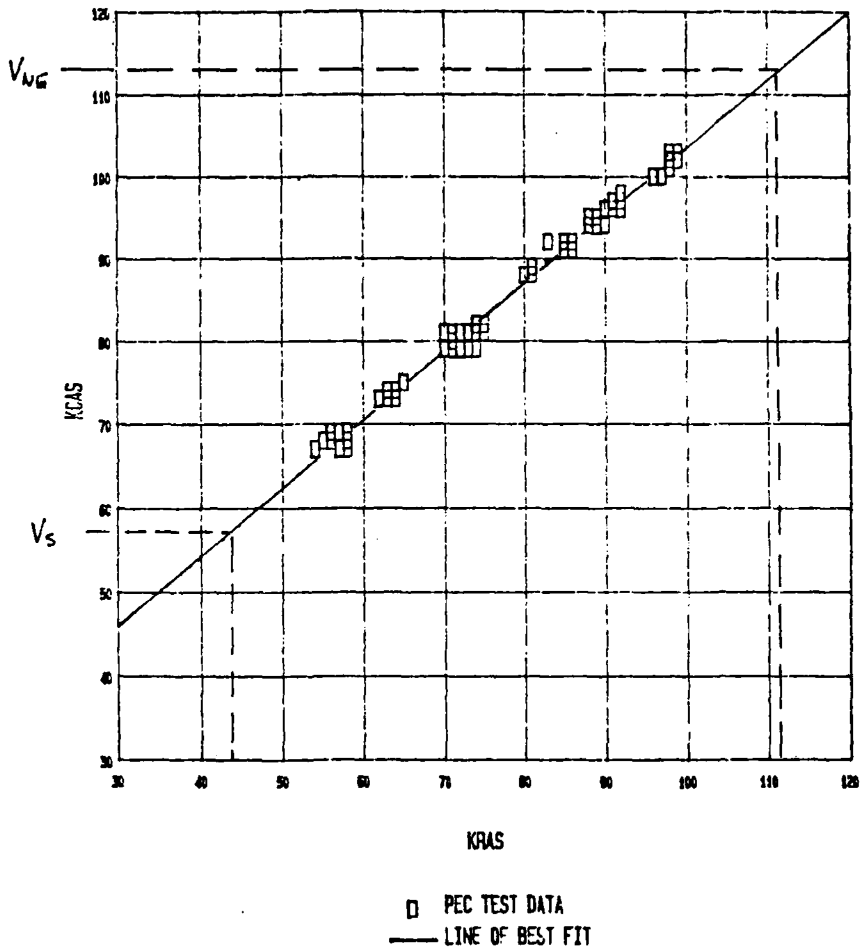


Figure 3

AIRSPEED INDICATOR MARKINGS

Reference:

F-1 Southern Cross Replica - Static Structural Design By C.W. Whitney, 5 October 1981

1. FAR 23.1545 requires that the ASI be marked to indicate V_{NE}, V_S and V_{NO} or V_C in terms of calibrated airspeed. Because of the large PEC it is recommended that these markings should be in terms of IAS so as to be of immediate use to the pilot.

2. The following derives the speeds to be marked. Reference F-1 gives the basic speeds for the flight envelope as:

V_S 57 KEAS

V_C 90 KEAS

V_A 102 KEAS

V_{NE} 113 KEAS

3. Appendix 1 to this Annex justifies changing V_A from 102 KEAS to a recommended value of 90 KEAS. If this recommendation is accepted V_C and V_A will both be 90 KEAS. The following table derives the IASs corresponding to the basic speed.

Speed	KEAS	PEC	KRAS	Pilot ASI	Co-pilot ASI
V_S	57	+ 13	44	45	43
V_C or V_A	90	+ 6	84	83	83
V_{NE}	113	+ 2	111	110	109

4. So that both pilot and co-pilot ASIs can have the same markings it is recommended that the most conservative values be chosen and used to mark both ASIs as follows:

V_S 45 KIAS

V_A or V_C 83 KIAS

V_{NE} 109 KIAS

RECOMMENDED AMENDMENT TO MANOEUVRE SPEED

References:

F-1 Southern Cross Replica - Static Structural Analysis by C.W. Whitney 5 October 1981.

F-2 Reports and Memoranda No 1096 'Full Scale Measurements of Lift and Drag of the Fokker FV11-3M Monoplane' by J.K. Hardy BA April, 1927.

F-3 FAR 23

1. Reference F-1 (Pages 5-1 and 5-2) determined manoeuvre speed V_A for structural analysis in the following way:

a. Stall speed, power off $V_S = \left(\frac{W}{\frac{1}{2}\rho \; C_{LMAX} \; S}\right)^{1/2}$

$$= \left(\frac{12500}{\frac{1}{2} \times .00238 \times 1.56 \times 725}\right)^{1/2}$$

$= 96.37$ ft/s

$= 57.06$ KEAS

where $C_{LMAX} = 1.56$ (power off) from Reference F-2

b. Limit manoeuvre load factor (n) from Reference F-3 (FAR 23.337-a-1)

$$n = 2.1 + \frac{24000}{W + 10000}$$

$$= 2.1 + \frac{24000}{12500 + 10000}$$

$= + 3.17$ g

c. Manoeuvre Speed: $V_A = V_S \sqrt{n}$

$= 57.06 \sqrt{3.17}$

$= 101.6$ KCAS

2. References F-2 also gives C_{LMAX} 'power on' as 1.96, substantially higher than for 'power off.' With a V_a of 101.6 KEAS and power on the manoeuvre load factor achievable (n') would be :

$$n' = n \frac{C_{LMAX}\text{ (power on)}}{C_{LMAX}\text{ (power off)}}$$

$$= n \times \frac{1.96}{1.56}$$

$$= 3.98 \text{ g}$$

This is 26% in excess of the limit manoeuvre load factor used in the stress analysis (Ref F-1).

3. For the aircraft to stall at the design limit manoeuvre load factor of 3.17 g V_A should be re-evaluated as follows:

$$V_S \text{ (power on)} = 57.06 \times \frac{C_{LMAX}\text{ (power off)}}{C_{LMAX}\text{ (power on)}}$$

$$= 57.06 \times \frac{1.56}{1.96}$$

$$= 50.39 \text{ KEAS}$$

Therefore $V_A = V_S \text{ (power on)}. \sqrt{n}$

$$= 50.39 . \sqrt{3.17}$$

$$= 90.6 \text{ KEAS}$$

4. It is recommended that for operational aspects V_A should be changed from 102 KEAS to 90 KEAS.

LONGITUDINAL STATIC STABILITY RESULTS

1. Stability and control testing was carried out with the aircraft in the following configurations:

TABLE 1 - STABILITY TEST CONFIGURATIONS

Ser. (a)	Configuration (b)	Trim Airspeed (KIAS) (c)	Power/Engine (in Hg/RPM) (d)	Take-Off Weight (kg) (e)	Take-Off CG (%MAC) (f)
1.	Cruise (CR1)	80	18-20/1950	4503	23.2
2.	Cruise (CR2)	"	"	5093	28.9
3.	Climb (CL1)	70	24/2050	4503	23.2
4.	Climb (CL2)	"	"	5093	28.9
5.	Powered Approach (PA1)	70	12-13/ approx 1700	4503	23.2
6.	Powered Approach (PA2)	"	"	5093	28.9

2. The longitudinal static stability test results are shown in the following tables. The presented control forces are estimations of the underlying stability forces and do not include the dominant breakout and friction forces. Due to the similarity of results only the aft CG, cruise configuration case (CR2) is presented graphically.

TABLE 2 - STABILITY TEST RESULTS

Ser. (a)	Configuration (b)	Test Airspeed (KIAS) (c)	Average Longitudinal Displacement (mm) (d)	Control Force (lbf) (e)
1	CR1	80	0	0
		85	-1	0
		90	-1	0
		95	-2	-2
		100	-2	-2
		105	-5	-5
		100	-3	-3
		95	-2	0
		90	-1	0
		85	0	0
		80	0	0
		75	1	0
		70	2	0
		65	4	3
		60	6	7
		65	6	5
		70	3	0
		75	0	0

TABLE 3 - STABILITY TEST RESULTS

Ser. (a)	Configuration (b)	Test Airspeed (KIAS) (c)	Average Longitudinal Displacement (mm) (d)	Control Force (lbf) (e)
1	CR2	80	0	0
		85	-1	0
		90	-1	0
		95	-2	0
		100	-2	-2
		105	-3	-5
		100	-3	-3
		95	-2	0
		90	-1	0
		85	0	0
		80	0	0
		75	1	0
		70	2	0
		65	3	2
		60	5	5
		65	4	3
		70	3	0
		75	0	0

TABLE 4 - STABILITY TEST RESULTS

Ser. (a)	Configuration (b)	Test Airspeed (KIAS) (c)	Average Longitudinal Displacement (mm) (d)	Control Force (lbf) (e)
1	CL1	70	0	0
		75	0	0
		80	-2	0
		85	-1	0
		90	-3	-3
		95	-3	-5
		90	-1	-5
		85	0	0
		80	0	0
		75	0	0
		70	0	0
		65	2	0
		60	4	0
		55	4	5
		60	3	0
		65	0	0

TABLE 5 - STABILITY TEST RESULTS

Ser. (a)	Configuration (b)	Test Airspeed (KIAS) (c)	Average Longitudinal Displacement (mm) (d)	Control Force (lbf) (e)
1	CL2	70	0	0
		75	0	0
		80	-2	0
		85	-2	0
		90	-3	-5
		95	-4	-5
		90	-2	0
		85	-2	0
		80	0	0
		75	0	0
		70	0	0
		65	2	0
		60	4	0
		55	6	5
		60	3	0
		65	2	0

TABLE 6 - STABILITY TEST RESULTS

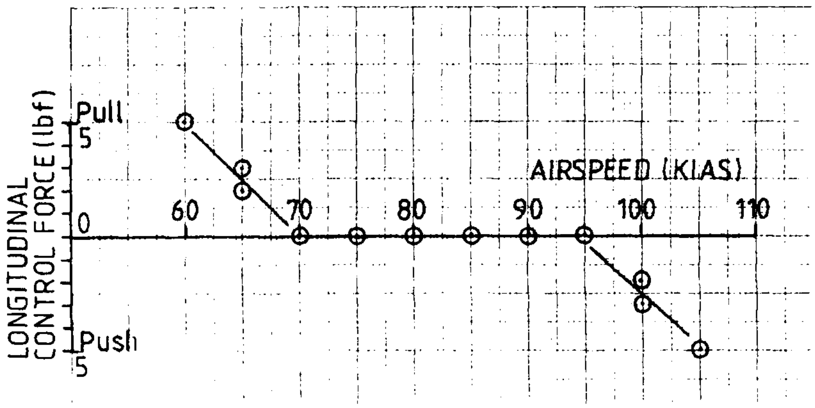
Ser. (a)	Configuration (b)	Test Airspeed (KIAS) (c)	Average Longitudinal Displacement (mm) (d)	Control Force (lbf) (e)
1	PA1	70	0	0
		75	-2	0
		80	-4	0
		85	-4	-5
		90	-6	-8
		95	-6	-6
		90	-3	-5
		85	-3	-3
		80	-1	0
		75	-2	0
		70	-1	0
		65	0	0
		60	2	0
		55	4	5
		60	4	3
		65	1	0

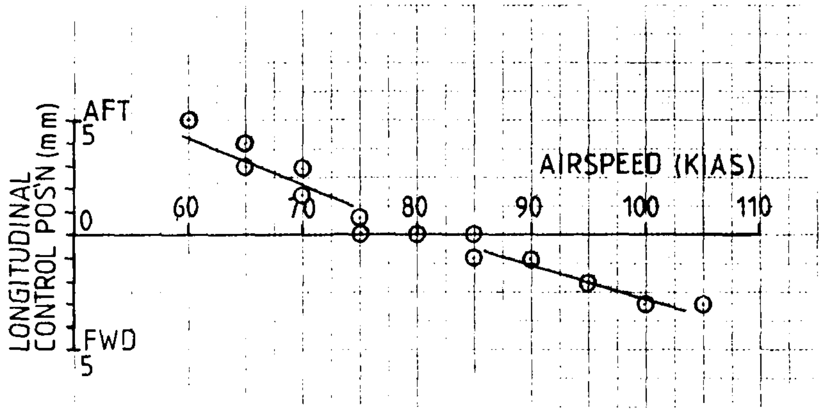
TABLE 7 - STABILITY TEST RESULTS

Ser. (a)	Configuration (b)	Test Airspeed (KIAS) (c)	Average Longitudinal Displacement (mm) (d)	Control Force (lbf) (e)
1	PA2	70	0	0
		75	-1	0
		80	-2	0
		85	-2	0
		90	-3	-3
		95	-3	-5
		90	-3	-5
		85	-3	0
		80	0	0
		75	0	0
		70	2	0
		65	4	0
		60	5	0
		55	5	5
		60	3	0
		65	3	0

LONGITUDINAL STATIC STABILITY

Aircraft:	Fokker VII6-3M Replica
Configuration:	CR2
Altitude:	2000 Ft
Trim Speed:	80 KIAS





LATERAL/DIRECTIONAL STATIC STABILITY RESULTS

1. The lateral/directional characteristics were checked in the following configurations:

TABLE 1 - STABILITY TEST CONFIGURATIONS

Ser.	Configuration	Trim Airspeed (KIAS)	Power/ Engine (in Hg/RPM)	Take-Off Weight (kg)	Take-Off CG (%MAC)
(a)	(b)	(c)	(d)	(e)	(f)
1.	Cruise (CR1)	80	18-20/1950	4503	23.2
2.	Cruise (CR2)	"	"	5093	28.9
3.	Climb (CL1)	70	24/2050	4503	23.2
4.	Climb (CL2)	"	"	5093	28.9
5.	Powered Approach (PA1)	70	12-13/ approx 1700	4503	23.2
6.	Powered Approach (PA2)	"	"	5093	28.9

2. Steady heading side-slips (SHSS) produced the results shown in the following tables. The presented forces are estimations of the underlying control forces and do not include the dominant breakout/friction forces. Side-slip was estimated by measuring the change in aircraft heading on release of the SHSS. Because of the similarity of results only the rear CG, cruise configuration (CR2) case is presented graphically.

TABLE 2 - STEADY HEADING SIDE-SLIP RESULTS

Ser.	Config.	Side-slip	Aileron		Rudder		Angle
			Displ't	Force	Displ't	Force	of Bank
		(deg)	(deg)	(lbf)	(cm)	(lbf)	(deg)
(a)	(b)	(c)	(d)	(e)	(f)	(g)	(h)
1	CR1	7R	10R	2R	4L	50L	5R
2	"	12R	15R	3R	8L	140L	15R
3	"	5L	8L	0	3R	50R	5L
4	"	12L	15L	2L	8R	140R	12L

TABLE 3 - STEADY HEADING SIDE-SLIP RESULTS

Ser.	Config.	Side-slip	Aileron		Rudder		Angle
			Displ't	Force	Displ't	Force	of Bank
		(deg)	(deg)	(lbf)	(cm)	(lbf)	(deg)
(a)	(b)	(c)	(d)	(e)	(f)	(g)	(h)
1	CR2	6R	10R	0	4L	50L	5R
2	"	10R	15R	2R	7L	140L	15R
3	"	5L	7L	0	4R	60R	7L
4	"	12L	20L	2L	7R	130R	15L

TABLE 4 - STEADY HEADING SIDE-SLIP RESULTS

Ser.	Config.	Side-slip	Aileron Displ't	Aileron Force	Rudder Displ't	Rudder Force	Angle of Bank
		(deg)	(deg)	(lbf)	(cm)	(lbf)	(deg)
(a)	(b)	(c)	(d)	(e)	(f)	(g)	(h)
1	CL1	5R	10R	0	4L	50L	5R
2	"	15R	25R	3R	8L	130L	15R
3	"	5L	8L	0	4R	50R	5L
4	"	12L	20L	2L	8R	130R	12L

TABLE 5 - STEADY HEADING SIDE-SLIP RESULTS

Ser.	Config.	Side-slip	Aileron Displ't	Aileron Force	Rudder Displ't	Rudder Force	Angle of Bank
		(deg)	(deg)	(lbf)	(cm)	(lbf)	(deg)
(a)	(b)	(c)	(d)	(e)	(f)	(g)	(h)
1	CL2	5R	7R	0	4L	50L	5R
2	"	12R	20R	2R	7L	130L	13R
3	"	5L	8L	0	3R	50R	5L
4	"	13L	25L	3L	8R	140R	15L

TABLE 6 - STEADY HEADING SIDE-SLIP RESULTS

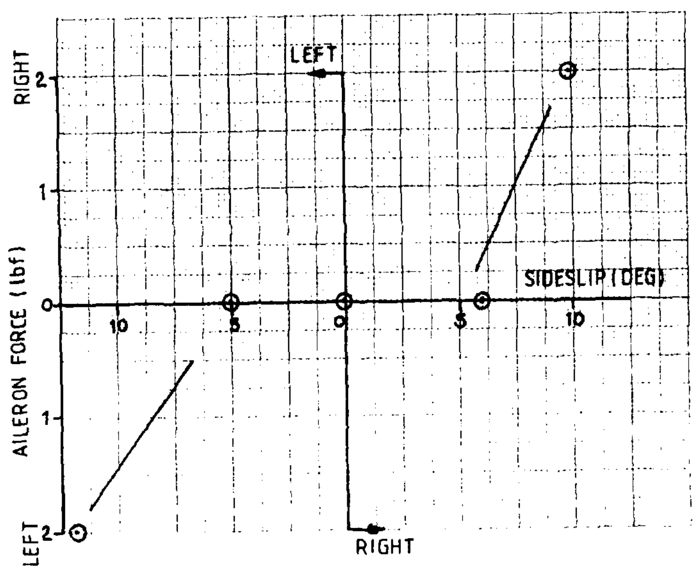
Ser.	Config.	Side-slip	Aileron Displ't	Aileron Force	Rudder Displ't	Rudder Force	Angle of Bank
		(deg)	(deg)	(lbf)	(cm)	(lbf)	(deg)
(a)	(b)	(c)	(d)	(e)	(f)	(g)	(h)
1	PA1	7R	10R	0	4L	50L	7R
2	"	15R	20R	3R	9L	140L	17R
3	"	8L	9L	0	4R	50R	8L
4	"	13L	25L	2L	8R	130R	15L

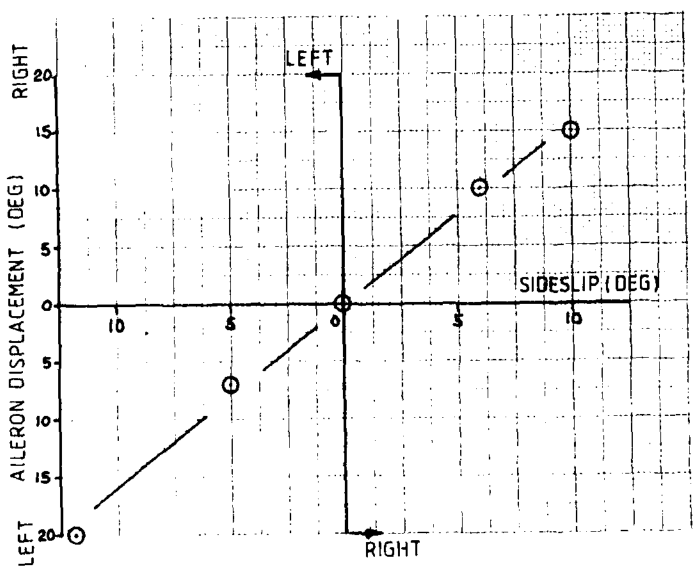
TABLE 7 - STEADY HEADING SIDE-SLIP RESULTS

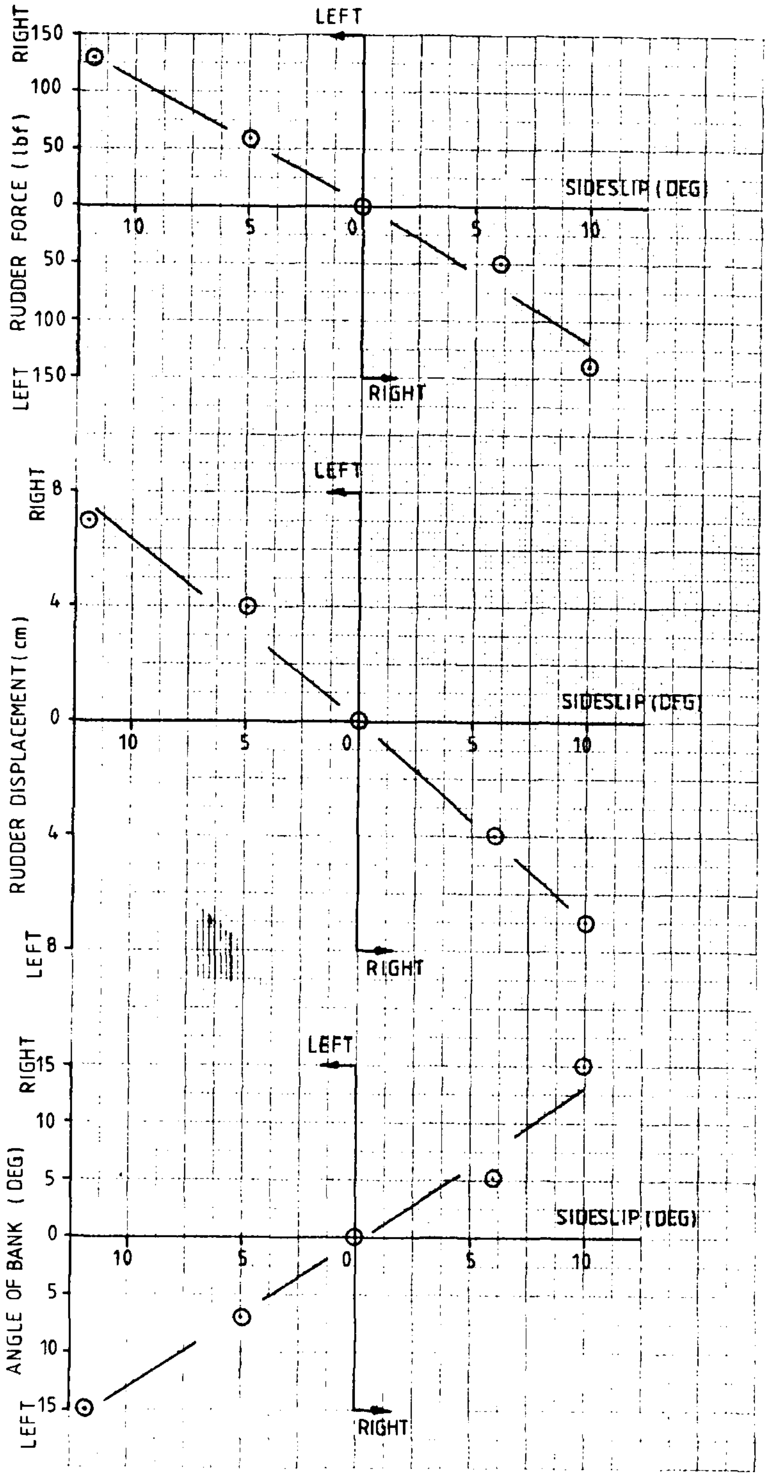
Ser.	Config.	Side-slip	Aileron Displ't	Aileron Force	Rudder Displ't	Rudder Force	Angle of Bank
		(deg)	(deg)	(lbf)	(cm)	(lbf)	(deg)
(a)	(b)	(c)	(d)	(e)	(f)	(g)	(h)
1	PA2	7R	10R	0	4L	50L	8R
2	"	15R	20R	3R	8L	140L	15R
3	"	7L	15L	0	4R	50R	7L
4	"	13L	25L	3L	8R	140R	15L

LATERAL/DIRECTIONAL STATIC STABILITY

Aircraft: Fokker VII6-3M Replica
Configuration: CR2
Altitude: 2000 Ft
Trim Speed: 80 KIAS






RUDDER FORCE (lbf)
RIGHT
LEFT
150
100
50
0
50
100
150
LEFT
RIGHT
SIDESLIP (DEG)
10
5
0
5
10
RUDDER DISPLACEMENT (cm)
RIGHT
LEFT
8
4
0
4
8
LEFT
RIGHT
SIDESLIP (DEG)
ANGLE OF BANK (DEG)
RIGHT
LEFT
15
10
5
0
5
10
15
LEFT
RIGHT
SIDESLIP (DEG)

TAKE-OFF AND LANDING PERFORMANCE

TABLE 1 - TAKE-OFF AND LANDING DISTANCES TEST RESULTS

Test	Weight (kg)	Temp (DEG C)	Pressure (hPa)	Head Wind (kn)	G, Roll (m)	A, Dist (m)	Total (m)
Take-Off 1	4525	11	1026	8	290	181	471
Take-Off 2	4510	11	1026	8	219	236	455
Take-Off 3	4490	11	1026	8	226	251	477
Take-Off 4	5680	8	1023	-	353	364	717
Take-Off 5	5650	8	1023	-	402	295	697
Landing 1	4510	11	1026	8	585	759	1344
Landing 2	4490	11	1026	8	553	637	1190
Landing 3	5690	8	1023	-	782	607	1389
Landing 4	5665	8	1023	-	822	519	1341

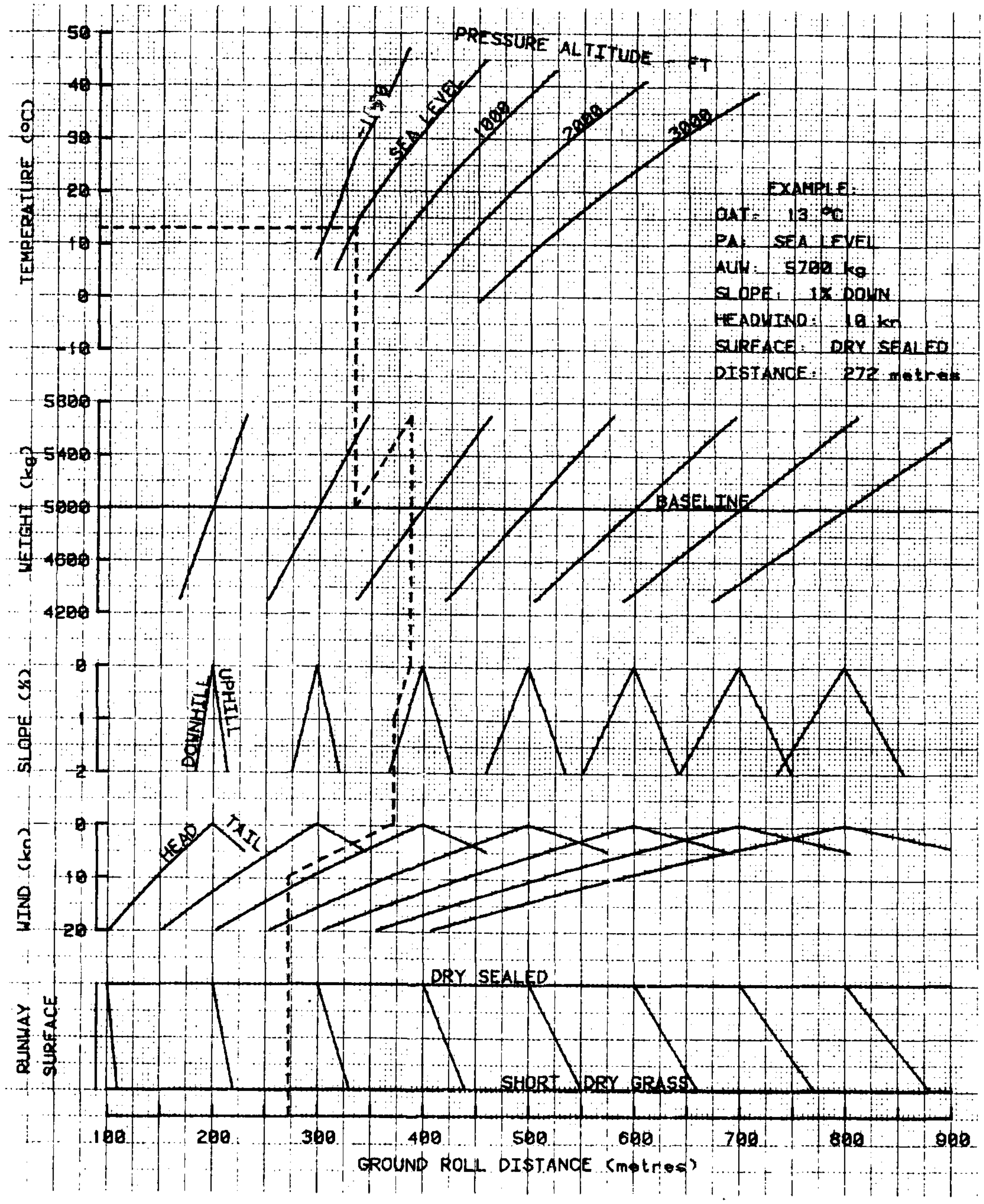


Figure 1 - Fokker VH-USU Take-off Ground Roll

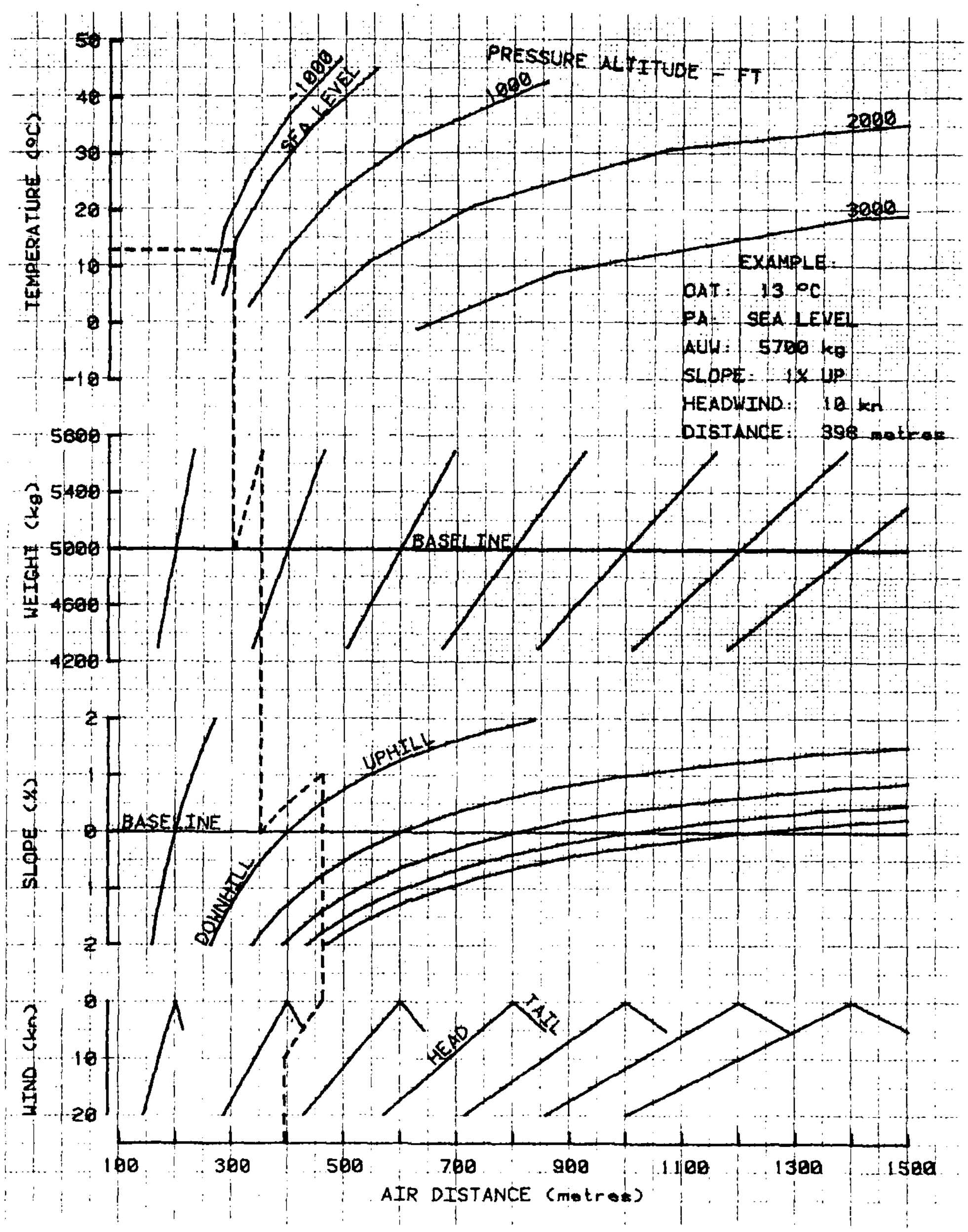


Figure 2 - Fokker VH-USU Take-off Air Distance

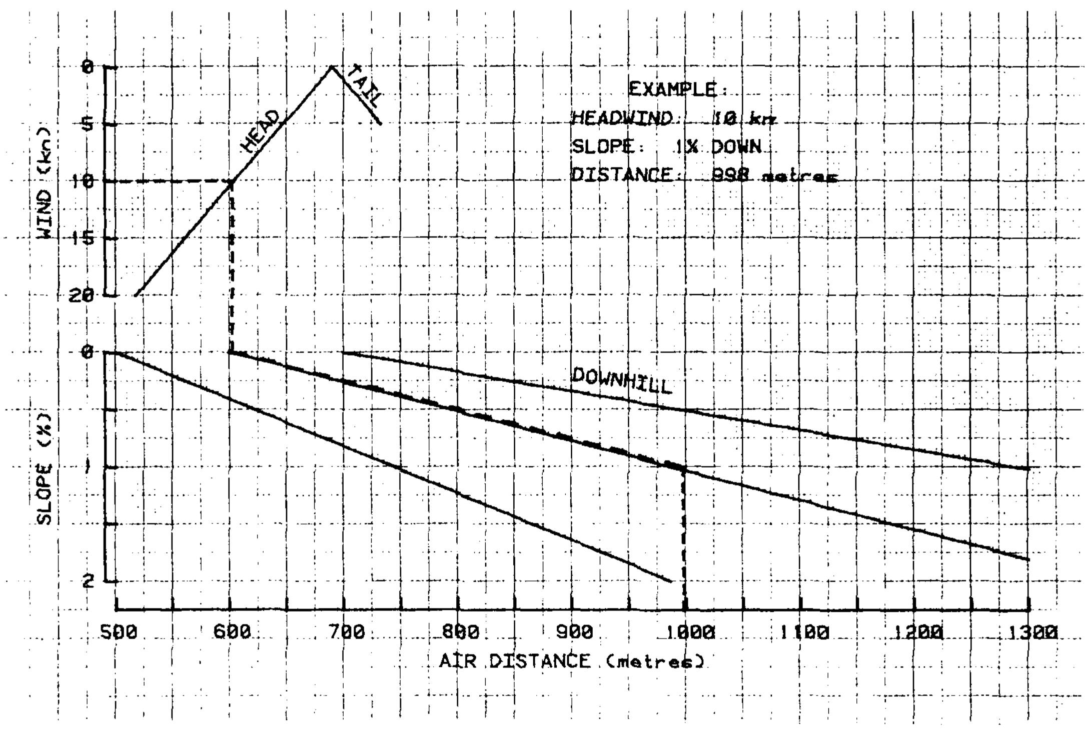


Figure 3 - Fokker VH-USU Landing Air Distance

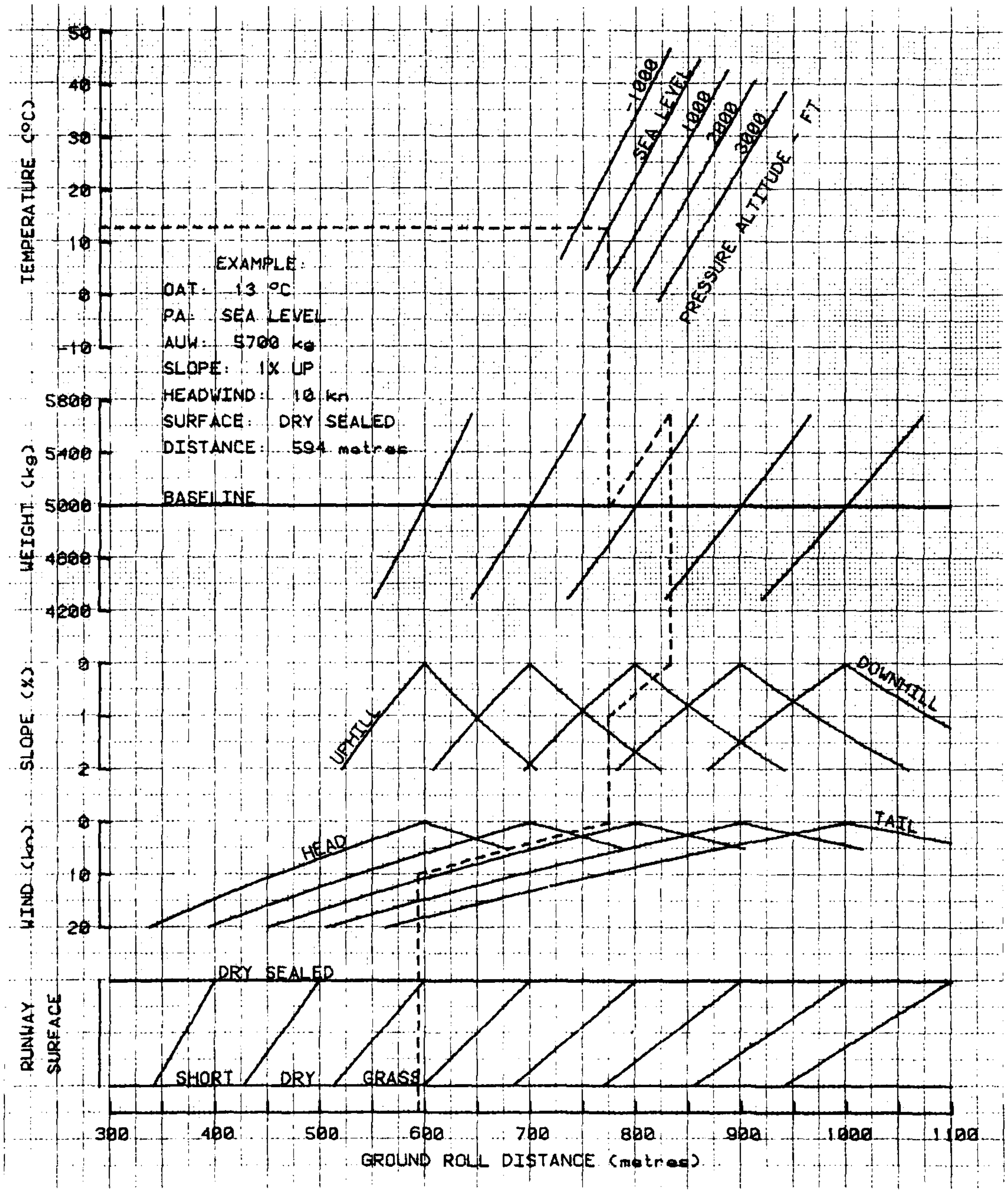


Figure 4 - Fokker VH-USU Landing Ground Roll

CLIMB PERFORMANCE DATA REDUCTION AND TEST RESULTS

CLIMB DATA REDUCTION

Reference:

J-1 United States Naval Test Pilot's School Flight Test Manual 104, Dated July 1977.

J-2 Engineering Science Data Units Item No 83001.

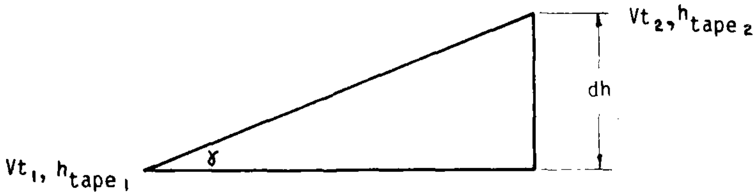
Introduction

1. The climb data is reduced using the specific excess power technique detailed in Reference J-1. The method involves conducting a sawtooth climb through a range of speeds and gross weights. The data is reduced and expanded as detailed below.

2. Data Reduction

2.1 Data Acquistion The climb data is acquired at a constant manifold air pressure (MAP) of 24 inches, until full throttle height is reached. The indicated airspeed is kept constant to minimise the change in true airspeed. The data recorded is included at page J-4.

2.2 Specific Excess Power. Specific excess power (P_s) is the rate of change of energy height (E_h) with respect to time. The P_s is reduced to sea level standard conditions and standard weight using the procedure outlined below.



$$E_h = h_{tape} + \frac{Vt^2}{2g}$$

$$Ps = \frac{dE_h}{dt} = \frac{dh}{dt} + \frac{Vt \times dVt}{g \times dt}$$

The second equation illustrates the point that if true airspeed is constant, then the rate of climb equals the P_s. The changes in true airspeed are included to determine the test value Ps.

2.3 Weight Correction. The weight correction is incorporated to account for changes in induced drag due to operations at weights apart from standard. Assuming constant thrust, the relationship for P_s, weight and drag, is defined below.

$$P_s\,(std) = P_s\,(test) * \frac{W_{test}}{W_{std}} * \frac{Vt_{std}}{Vt_{test}} + \frac{Vt_{std}}{W_{test}}\,(\Delta D)$$

where,

$$D = 2 \times \frac{W_{std}^2 \times \cos\gamma^2 - W_{test}^2 \times \cos\gamma^2}{\pi \times eAR \times \rho_{ssl} \times Ve^2 \times S}$$

2.4 Altitude Correction. The altitude correction is used to account for thrust variation with altitude. The relationship of thrust with altitude is calculated using Reference J-2. The delta thrust correction is applied in a similar manner to the delta drag correction.

$$P_s(std) = P_s(test) * \frac{W_{test}}{W_{std}} * \frac{Vt_{std}}{Vt_{test}} + \frac{Vt_{std}}{W_{test}}\,(\Delta T)$$

3. Data Expansion.

3.1 Maximum Rate of Climb. The maximum rate-of-climb is selected from the graph of P_s versus TAS at sea level which is included at figure 1.

3.2 Altitude Effects. The maximum rate-of-climb can be calculated for differing altitudes and temperatures using the delta thrust correction in a reverse manner to the equation of paragraph 2.4. The relationship of thrust with altitude and temperature can be calculated using Reference J-2.

$$P_s(alt) = \left(P_s(std) - \frac{Vt_{std}}{W_{std}} \times (\Delta T)\right) \times \frac{Vt_{alt}}{Vt_{std}} \times 1.6889$$

3.3 Weight Effects. The weight effects on the rate-of-climb can be calculated as described in paragraph 2.3.

4. Data Presentation.

4.1 Maximum Rate of Climb. The maximum rate-of-climb data is presented in figure 2. The data is presented at the optimum climb speed and at constant climb power. The data is only valid below 4000 ft, as it is based on constant climb power, which decreased with altitude above 4000 ft.

4.2 Engine-Out Climb Performance. The delta thrust calculated in paragraph 3.2 allows calculation for two engines operating at maximum rated power. The data is presented in figure 2 for altitudes below 4000 ft.

CLIMB PERFORMANCE NOTATION

Symbol	Description	Units
E_h	Energy Height	ft
h_{tape}	Tapeline Altitude	ft
Vt	True Airspeed	ft/sec
Vt_{test}	Test Airspeed - Test Conditions	ft/sec
Vt_{std}	True Airspeed - Standard Cond	ft/sec
g	Gravitational Acceleration	ft/sec^2
W_{test}	Aircraft Test Weight	lbf
W_{std}	Aircraft Standard Weight	lbf
$P_s(test)$	Specific Excess Power - Test Cond	ft/sec
$P_s(std)$	Specific Excess Power - Std Cond	ft/sec
D	Drag	lbf
T	Thrust	lbf
ρ_{ssl}	Standard Sea Level Air Density	lbf/ft^3

CLIMB PERFORMANCE

TEST RESULTS

Test	Airspeed KIAS	Time (secs)	Weight (kg)	Temp (DEG C)	Pressure Alt (ft)	Power Setting MAP(in)	RPM
1	75	0	5614	14	2520	24	2050
	74	116	5614	13	3050	24	2050
	76	243	5614	13	3550	24	2050
	76	359	5614	13	4050	24	2050
	77	469	5614	12	4550	23	2050
2	66	0	5582	13	2520	24	2050
	64	89	5582	13	3050	24	2050
	64	166	5582	13	3550	24	2050
	63	255	5582	13	4050	24	2050
	63	345	5582	12	4550	23	2050
3	56	0	5550	14	2520	24	2050
	55	85	5550	14	3050	24	2050
	56	170	5550	13	3550	24	2050
	55	249	5550	13	4050	24	2050
	55	326	5550	11	4550	24	2050
4	75	0	4470	12	2020	24	2050
	75	58	4470	12	2520	24	2050
	76	119	4470	12	3050	24	2050
	76	172	4470	12	3550	24	2050
5	85	0	4444	12	1520	24	2050
	84	91	4444	12	2020	24	2050
	85	193	4444	12	2550	24	2050
	85	296	4444	12	3050	24	2050
	84	396	4444	11	3550	24	2050
6	66	0	4420	12	1520	24	2050
	65	50	4420	12	2020	24	2050
	67	107	4420	12	2550	24	2050
	68	159	4420	12	3050	24	2050
	66	205	4420	11	3550	24	2050
7	58	0	4396	12	1520	24	2050
	56	40	4369	12	2020	24	2050
	56	90	4396	12	2550	24	2050
	59	138	4396	12	3050	24	2050
	58	180	4396	11	3550	24	2050

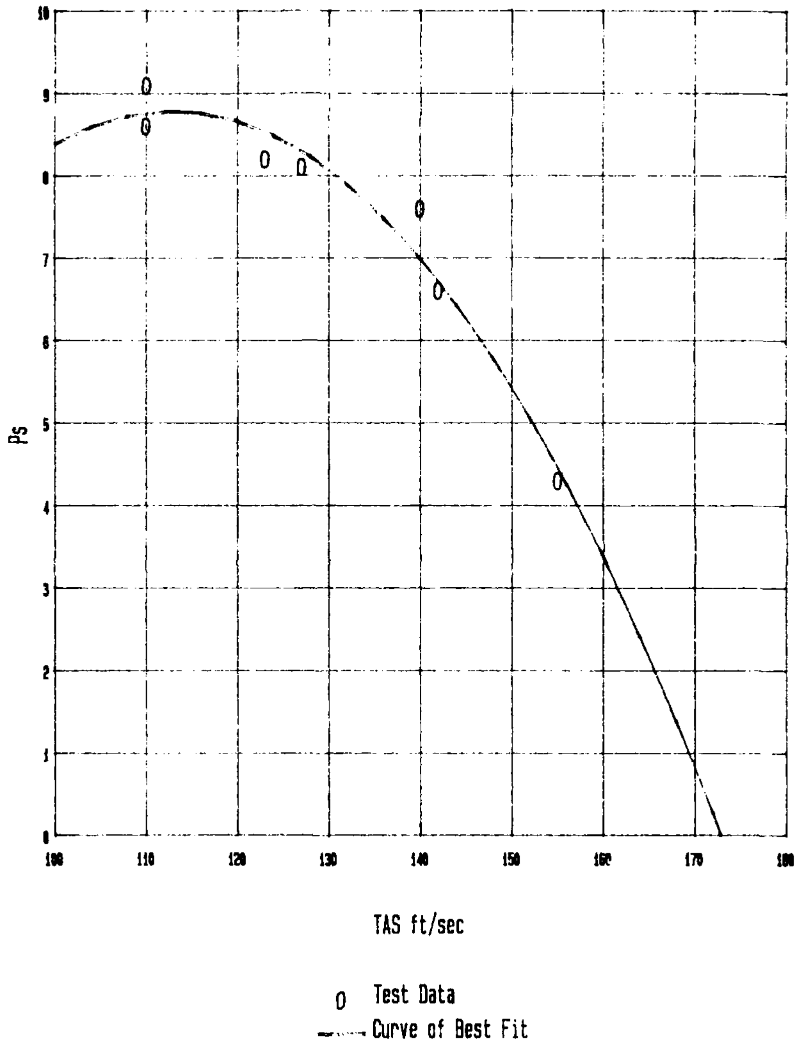


Figure 1 - Specific Excess Power Vs True Airspeed @ SL ISA

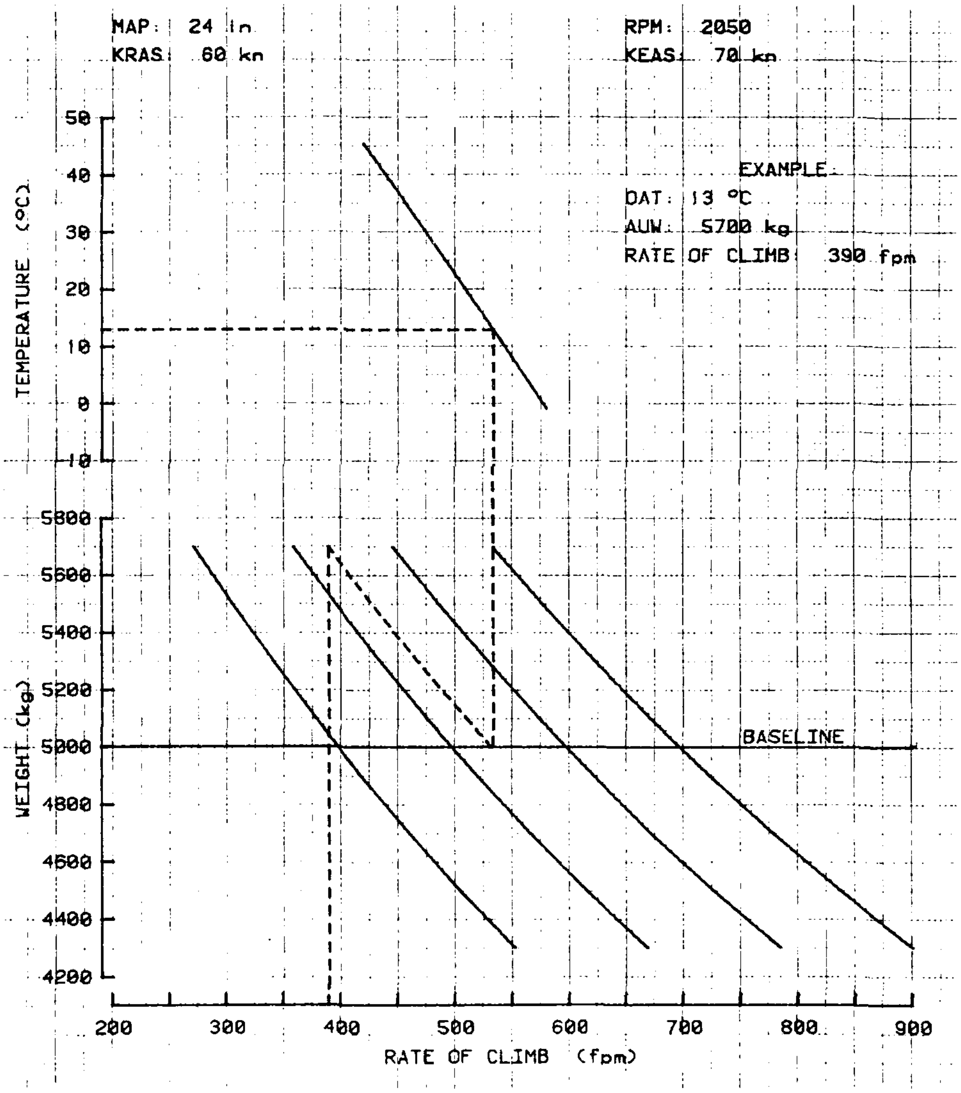


Figure 2 - Fokker VH-USU Rate of Climb

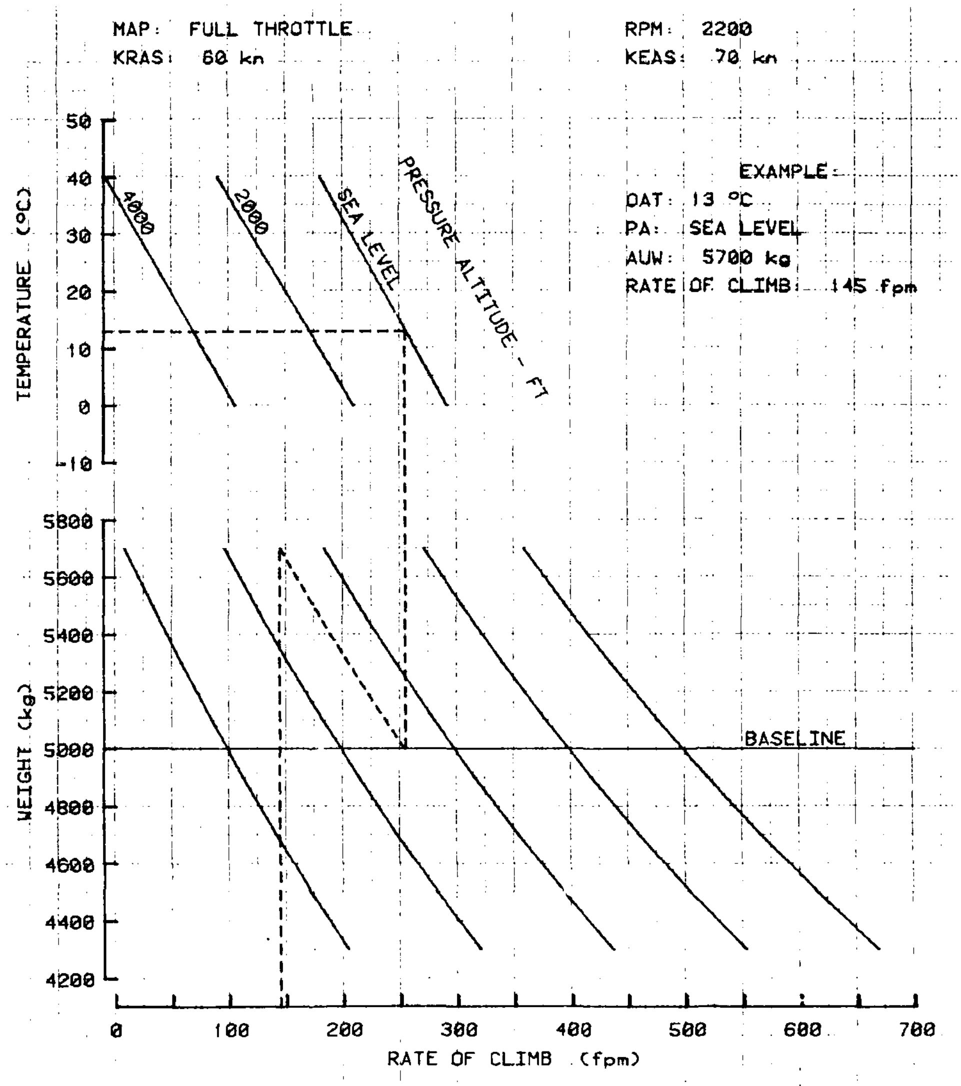


Figure 3 - Fokker VH-USU Rate-of-Climb - Engine-Out

CRUISE PERFORMANCE DATA REDUCTION AND TEST RESULTS

CRUISE PERFORMANCE

Reference:

K-1 United States Naval Test Pilot's School Flight Test Manual 104, Dated July 1977.

K-2 Operators Manual, Jacobs Aircraft Engine Models, R-755A, R755B and R-755S dated 1974.

Performance Technique

1. The Reference K-1, level flight performance technique was used to evaluate cruise performance.

Detailed Data Reduction

2. The following is a summary of the data reduction method contained in Reference K-1.

a. Compute brake horsepower for test point:

$$BHP = f(MAP, RPM, Alt, T_a)$$

calculated from manufacturers' data, R-755A performance chart of Reference K-2.

b. Compute equivalent brake horsepower required by:

$$BHP_e = BHP \times \sqrt{\sigma}$$

c. Compute ratio of standard aircraft gross weight to test aircraft gross weight (ie. W_s/W)

d. Correct equivalent brake horsepower required and equivalent airspeed to standard gross weight by:

$$BHP_{ew} = BHP_e \, (W_s/W)^{**}3/2$$

$$V_{ew} = V_e \, (W_s/W)^{**}1/2$$

e. Compute V_{ew}**4 and V_{ew}*BHP_{ew} and plot graph of data (figure 1).

f. The data is faired by fitting the line of best fit. The equation for the line of best fit can be re-arranged to give BHP_{ew} as a function of V_{ew}. The faired data is presented in figure 2.

g. The fuel flow data as a function of brake horse power is included in manufacturers data in Reference K-2. The data was graphed for three cruise engine speeds throughout the power range of the engines (figure 3).

h. The fuel flow can be plotted against V_{ew} (figure 4).

i. The specific air range (SAR) can be plotted by calculating V_t and dividing by fuel flow and plotting against V_{ew} (figure 5).

j. The data can then be unreferred to give cruise data in terms of fuel flow and SAR against V_e. Figures 6 and 7 include unreferred data throughout the altitude, temperature and speed range evaluated.

CRUISE PERFORMANCE NOTATION

Symbol	Description	Units
P_a	Ambient Pressure	hPa
P_{ssl}	Standard Sea Level Pressure	hPa
MAP	Manifold Air Pressure	in/hg
RPM	Engine Speed	-
BHP	Brake Horsepower	HP
BHP_e	Equivalent Brake Horsepower	HP
BHP_{ew}	Equivalent Weight Brake Horsepower	HP
T_a	Ambient Temperature	^oK
T_{ssl}	Standard Sea Level Temperature	^oK
V_e	Equivalent Airspeed	kn
V_{ew}	Equivalent Weight Airspeed	kn
V_t	True Airspeed	kn
W	Aircraft Gross Weight	kg
W_s	Standard Aircraft Gross Weight	kg
ρa	Ambient Air Density	kg/m^3
ρssl	Standard Sea Level Air Density	kg/m^3
δ	P_a/P_{ssl}	-
θ	T_a/T_{ssl}	-
σ	$\rho a/\rho ssl$	-

CRUISE PERFORMANCE

TEST RESULTS

Sortie	Weight (kg)	Temp (°C)	Pressure Alt (ft)	IAS (kn)	RAS (kn)	TAS (kn)	MAP (in)	RPM	BHP
3	4950	2	3130	86	92	95	20	2050	546
	4950	2	2850	73	80	82	16.5	1900	375
4	4890	4	3260	90	96	100	21	2000	564
	4870	4	3040	74	82	85	17.3	2000	432
	4860	4	3050	65	74	77	16	1725	315
	4850	4	3050	60	67	69	15	1725	278
	4848	4	3000	97	102	106	24	2050	689
16	4395	11	4000	96	101	108	24	2050	702
	4390	11	3950	89	96	102	21	1950	558
	4385	11	3930	75	82	87	17	1780	366
	4380	11	3870	70	78	83	15.7	1900	354
	4375	11	3990	58	67	71	15	1800	318
	4370	11	3980	51	61	65	14	1650	252
	4365	11	4020	39	50	54	16.4	1820	369
	4360	11	3880	50	60	64	15	1740	297
	4355	11	3940	58	67	72	15.4	1830	336
	4350	11	3980	75	82	87	16.6	1950	410
20	5506	12	4070	94	99	106	24	2050	702
	5503	12	4000	90	96	103	22.7	2050	660
	5500	12	4050	79	86	92	18.5	1950	474
	5496	12	4020	70	78	84	16.7	1950	411
	5492	12	4000	62	70	75	15.5	1850	345
	5488	12	3850	51	61	65	17.9	1950	450
	5484	12	3850	55	64	68	18.6	1950	474
	5481	12	3950	58	67	72	18.7	1950	477
	5478	12	4000	70	78	84	17	1950	426
	5474	12	4000	78	85	91	18.4	1950	468
	5468	12	4000	93	98	105	24	2050	702

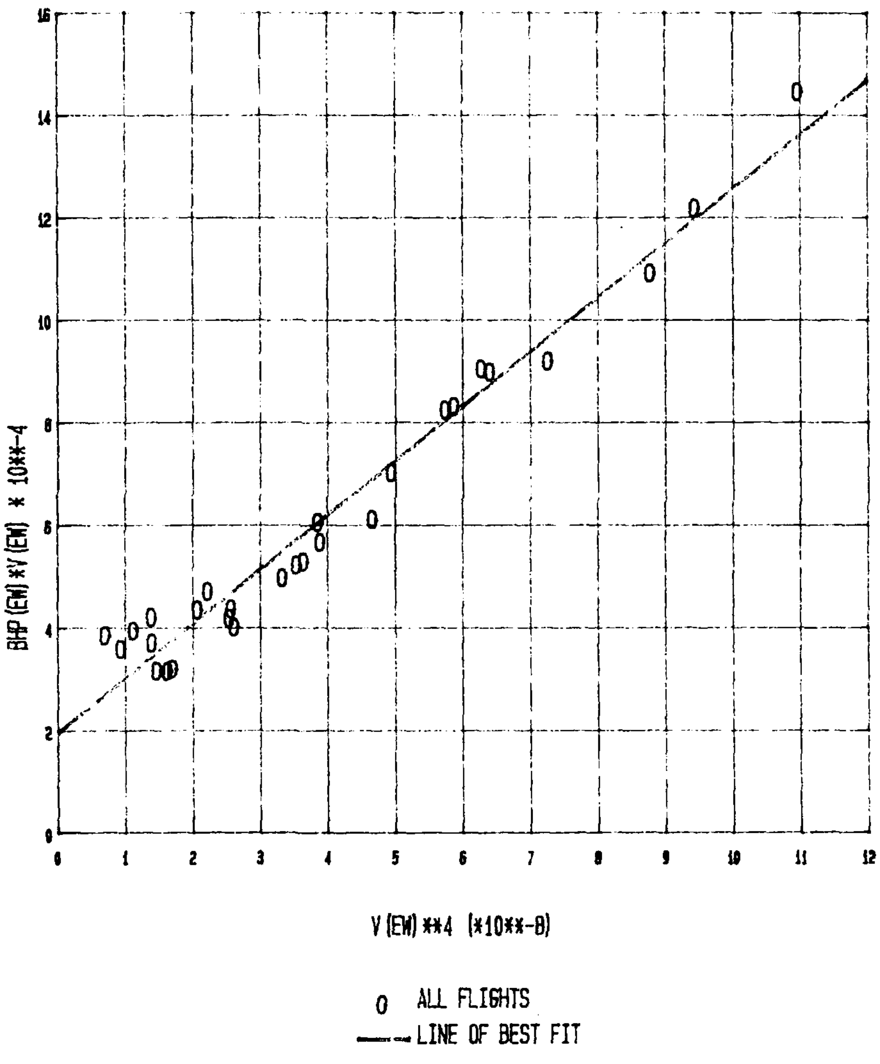


Figure 1 - Fokker VH-USU BHP (EW)*V (EW) Vs V (EW)**4

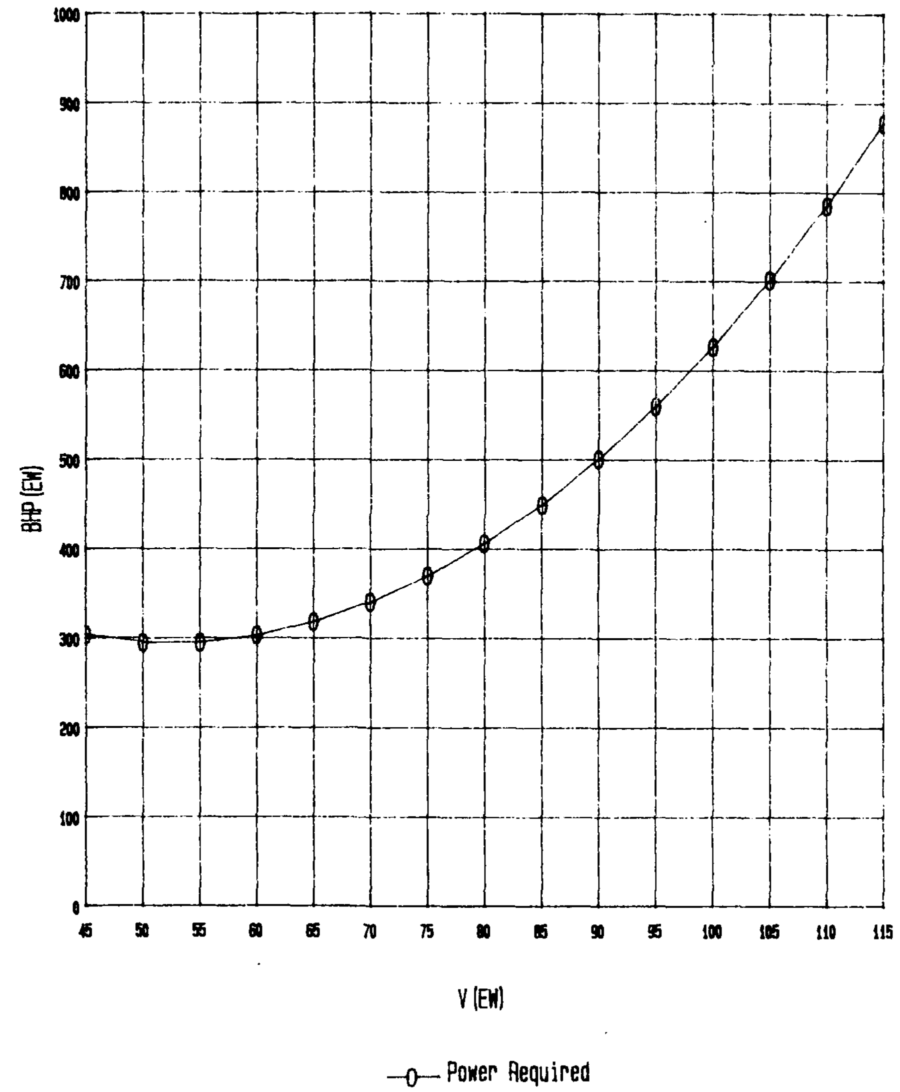


Figure 2 - Fokker VH-USU BHP (EW) Vs V (EW)

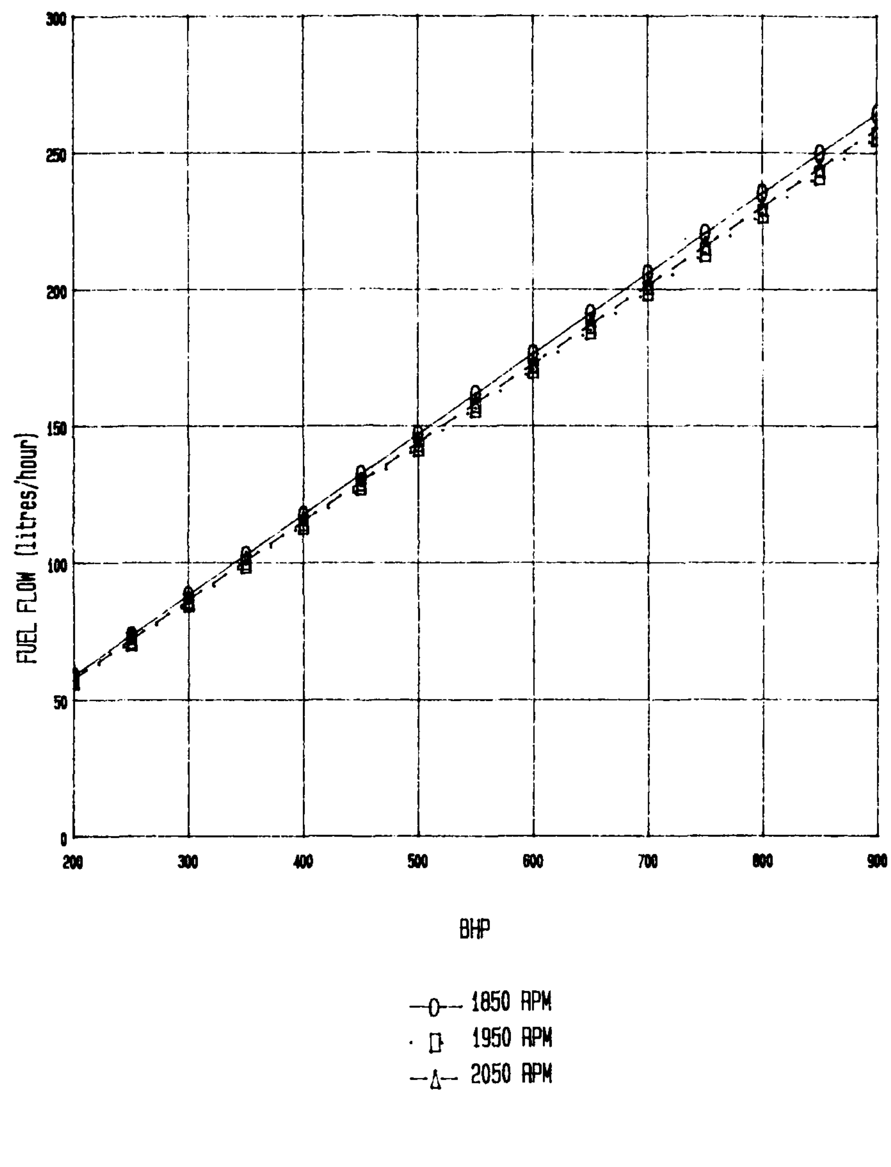


Figure 3 - Fokker VH-USU Fuel Flow Vs BHP

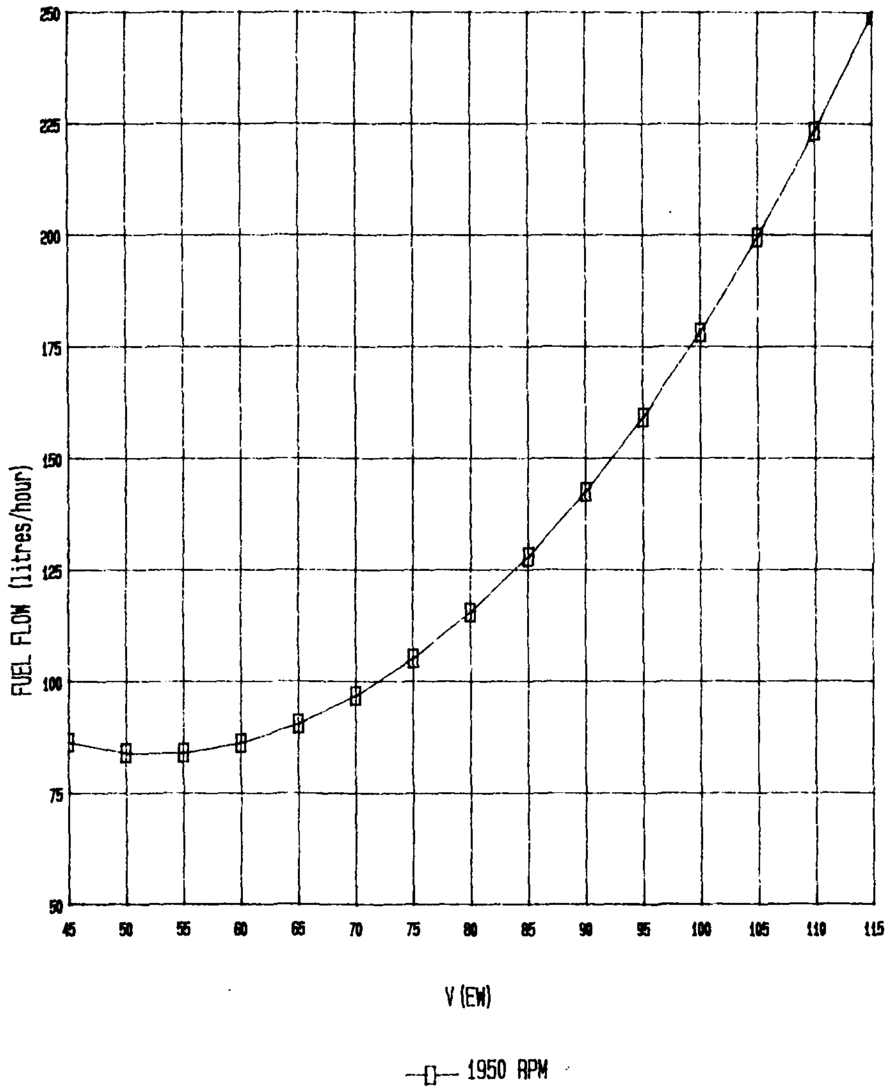


Figure 4 - Fokker VH-USU Fuel Flow Vs V(EW)

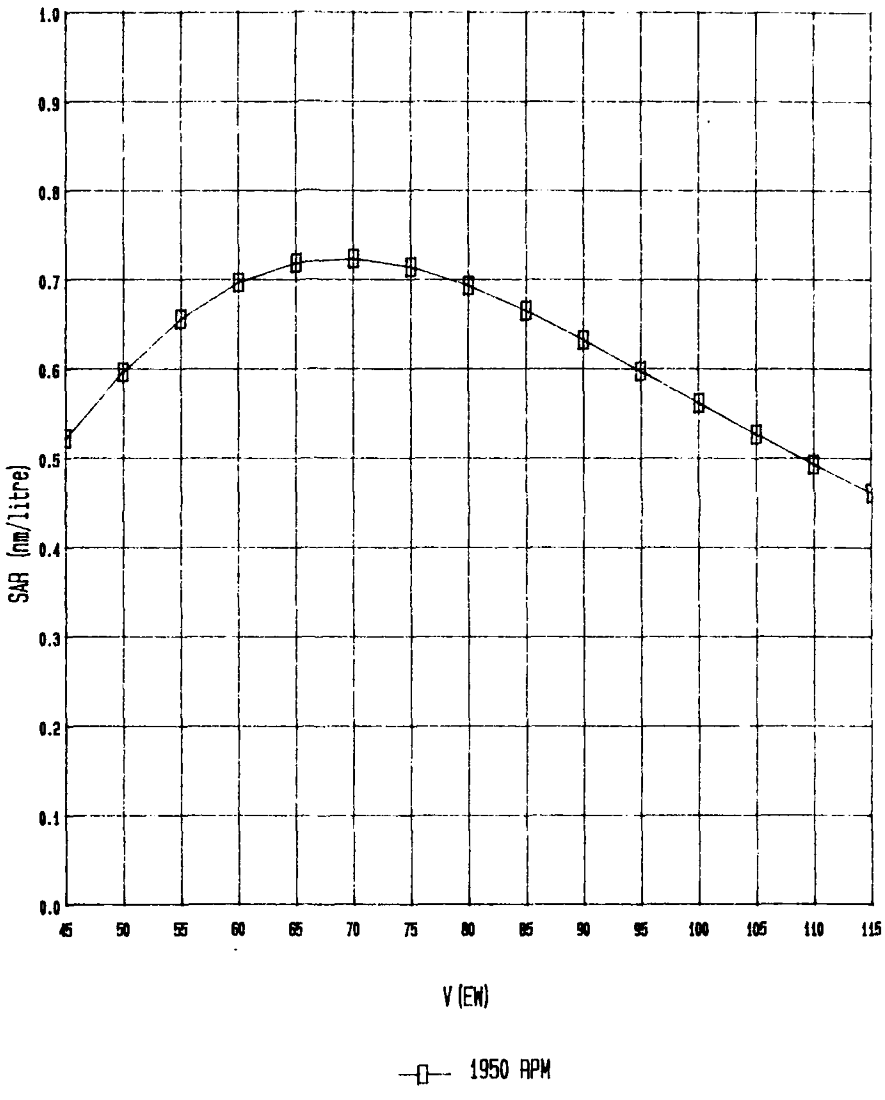


Figure 5 - Fokker VH-USU SAR Vs V (EW)

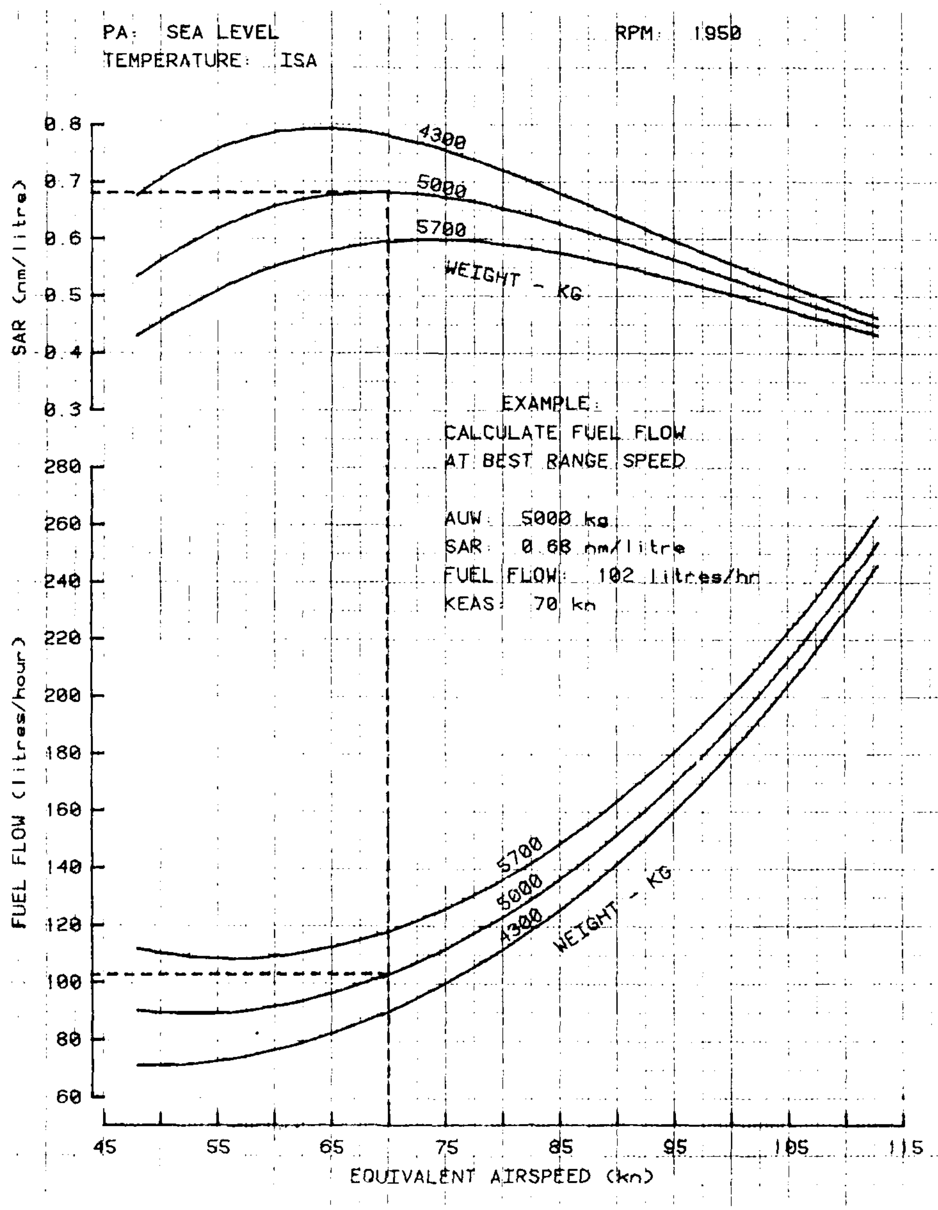


Figure 6 - Fokker VH-USU Cruise Performance

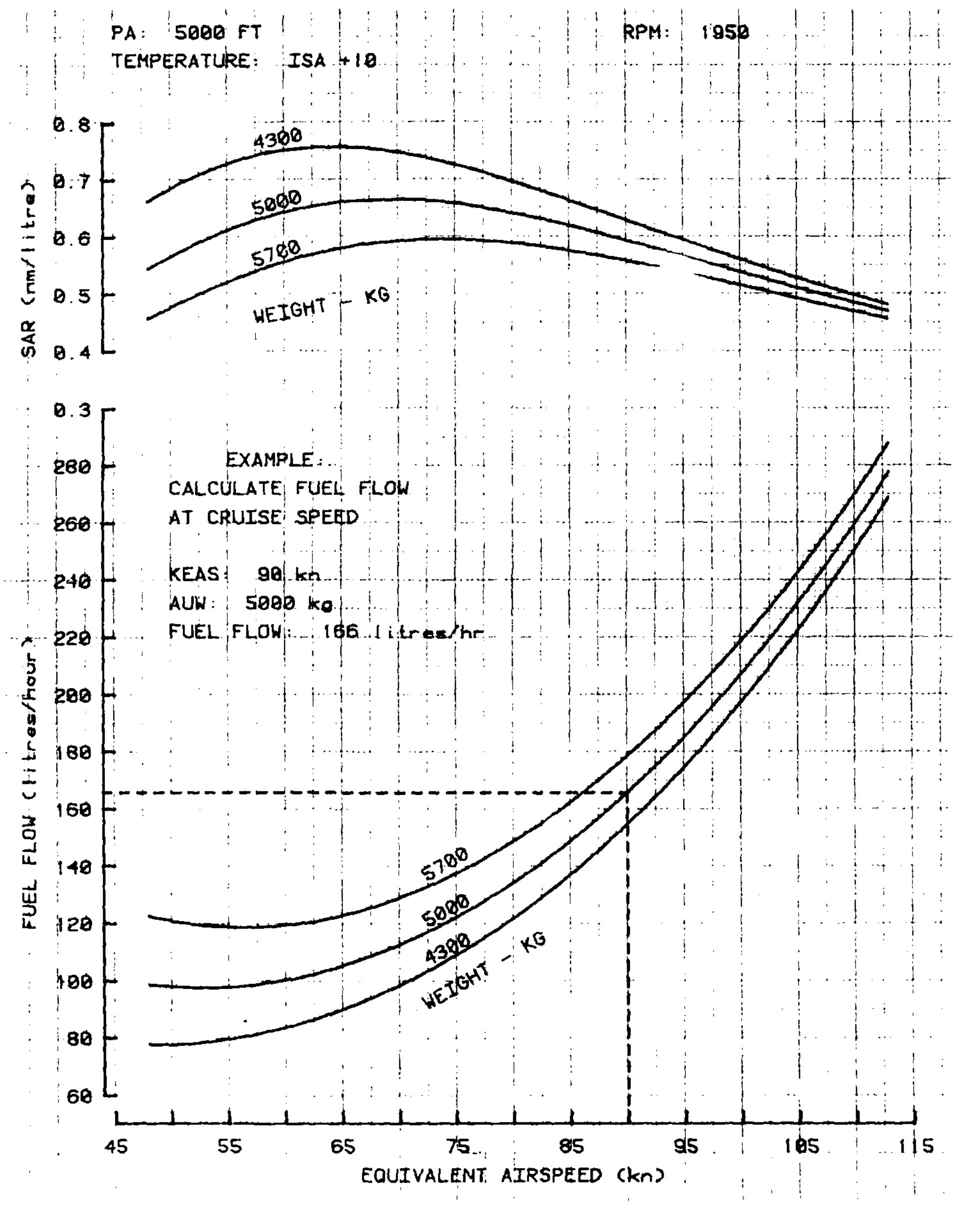


Figure 7 - Fokker VH-USU Cruise Performance

AIRCRAFT RESEARCH AND DEVELOPMENT UNIT

TECHNICAL INVESTIGATION NO 953

FLIGHT TESTING OF THE SOUTHERN CROSS REPLICA AIRCRAFT

DISTRIBUTION EXTERNAL

		Copy No
Department of Defence		
Air Force Office:	DCAS	1
	DGOR-AF	2
	DAFS	3
	DRR-AF	4
	TP3-AF	5
Central Office:	DTRIALS	6
Defence Information Services Branch, Document Exchange Centre		7-24
Support Command		
Headquarters Support Command (SLENGO)		25
Civilian Establishments		
Chief Certification Engineer, Airworthiness Branch, Flight Standards Division, Civil Aviation Authority, CANBERRA		26
Assistant General Manager, Aviation Standards, Civil Aviation Authority, ADELAIDE		27
Mr J.E. Schofield, Project Manager, Southern Cross Museum Trust		28
Defence Science and Technology Organisation Library, SALISBURY		29

DISTRUBUTION INTERNAL

Commanding Officer	30
Project Officer	31
Library	32
Archive Copy	33
Spare Copies	34-40