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Compass Cope Airframe Design History

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The Boeing Cope was a 90-ft span remotely piloted high altitude reconnaissance design flight tested in 1973-1974. Two airplanes were constructed in Seattle and both were flown at Edwards Air Force Base. Contract funds intended for the complete program of design, construction, and flight test of both prototypes were less than ten million dollars. The program is believed to be significant because of the extent of achievement within this austere funding level. A major program accomplishment was an airframe of new concepts, without a subscale test program, and yet free of inherent arrangement or detail deflects in flight test. The program results include a laminar flow wing, all bonded primary structure components, integral insulated and bonded fuel tank, high lift to drag ratio, superior flying qualities, and toal adaptiveness to electronic mission equipment.

Introduction

THE Compass Cope remotely piloted vehicle (Figs. 1, 2, 3) airframe design history emphasizes the preliminary design phase of one year from mid-1970 to demonstrator contract award in mid-1971. The effect of the preliminary design decisions on the project phase (1971-1972) and flight test (1973-1974) will be discussed. The events of these time periods are shown in Fig. 4.

Boeing's 1969-1970 Cope organization consisted of a management/marketing/avionics oriented group. Small military contracts were obtained for remotely piloted flight test demonstrations using standard light airplanes. There was no airframe design team involvement in the original avionics oriented remotely piloted developments. This initial Cope management group had the long term objective of obtaining development and production contracts for new, remotely piloted surveillance and remotely piloted weapon systems. Addition of airframe design capability was necessary for this objective.

The airframe design group that was formed was a small project and technology staff group of approximately 15 engineers. These engineers had a previous history of common assignments. Project, staff, and manufacturing reported to the same supervisor with no technical ties to their parent functional organizations due to the security characteristics of the program. The experience of this group had been fundamentally on manned aircraft, bombers, fighters, and military-commercial transports.

A few months of initial airframe design studies and customer contacts resulted in formulation of prototype demonstrator requirements. These were early flight verification of: 1) conventional take-off and landing; 2) 24 h of remotely piloted vehicle (RPV) flight above 55,000 ft; 3) 1000 lb equivalent surveillance payload; and 4) total prototype demonstration costs of less than 5.0 million.

Airframe design group objectives resulting from the flight demonstration requirements were: 1) no crashes; back-up airframe in event of a crash; 2) no propulsion system risk or development funding; 3) no freezing of fuel, hydraulic or other temperature sensitive components or systems; 4) almost no model or system tests; 5) concept choices avoiding all timing, functional and cost risk; 6) avionics oriented airplane arrangement; 7) high L/D and structural efficiency; and 8) high fuel volume.

These objectives had to be met within the following constraints: existing light plane developed autopilot; existing electrical power control actuators; and existing data link system. A judgment was made that program timing and budget precluded the use of other options for these components. The preliminary design proceeded from the aforementioned background of objectives and constraints.

Cope Airframe Decisions and Rationale

A. Organization Related Decisions

The budget and timing environment of Cope required that the normal engineering-manufacturing iteration process be drastically shortened. To achieve this, a manufacturing and tooling supervisor with engineering and graphic skills was assigned directly to the design team. Each initial airframe drawing was graphically planned, detailed, and tooled for the manufacturing process using three-dimensional hand sketching techniques of each step and tool involved (Fig. 5). These seemingly simple sketches were significant proof of having thought out a realistic and complete design and manufacturing concept. In this process, the engineering design concept had suggested modification or complete change. These suggestions were incorporated in the second engineering drawings or were replaced by a new and collaborated concept. This was a continuous interactive process that was considered effective and easy by project, technical staff, and manufacturing specialists. Concentrated effort was required by manufacturing but it was in an environment of total engineering cooperation.

The decision to concentrate manufacturing, project, and staff in one configuration organization reduced the time and cost of the Cope definition process. It was a necessity, since there was no time or money for the total organization interaction normally involved in configuration definition. The self-contained configuration definition organization offered an opportunity, or risk, for accountability that is diffused in large programs.

B. Mission Related Decisions

Before the design process could efficiently proceed, the question of man aboard or not on the demonstrator had to be decided. Those involved in this decision oscillated between both positions. For man-aboard capability: manned airplanes almost never crash as compared to RPVs and manned flight could be the assured standby until remote piloting is debugged. Against man-aboard capability: manpower, time,

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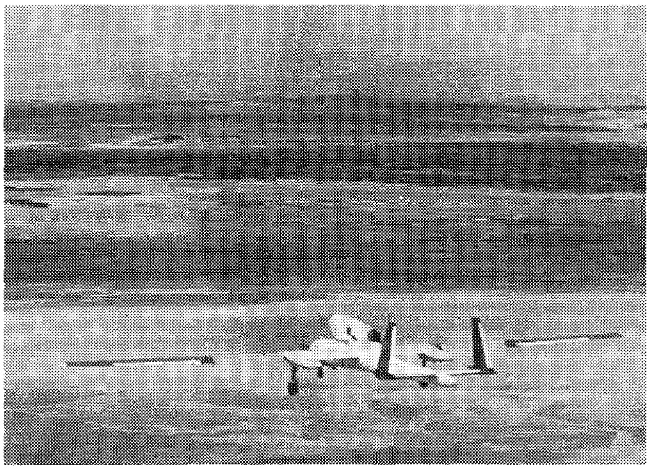


Fig. 1 YQM-94A compass Cope landing approach.

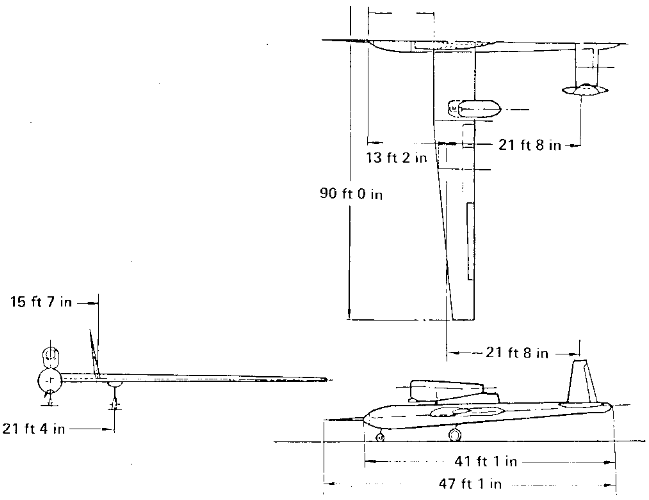


Fig. 2 General arrangement of YQM-94A.

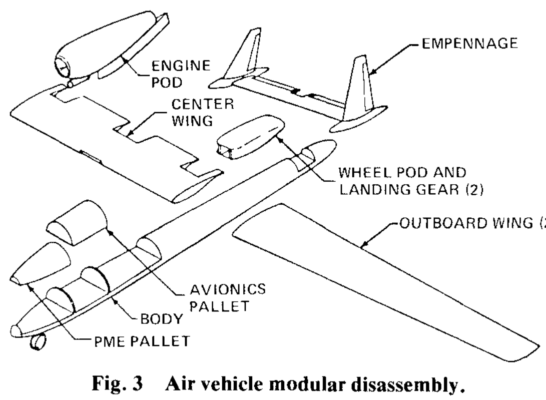


Fig. 3 Air vehicle modular disassembly.

and money required to include man-aboard, RPV, and mission equipment criteria in basic design is excessive superposition of requirements in the low priority environment of the program; and if our intended business is remotely piloted surveillance, let us assure serious work on unmanned flight safety by eliminating the man crutch right from the start.

A decision between these two positions was made early with the judgment that man-aboard capability was not compatible with the size of the airplane or the program resources. In addition, light plane RPV experience leading to the Cope design provided some substantiation that man-aboard was not a necessity. In the absence of man-aboard capability, serious

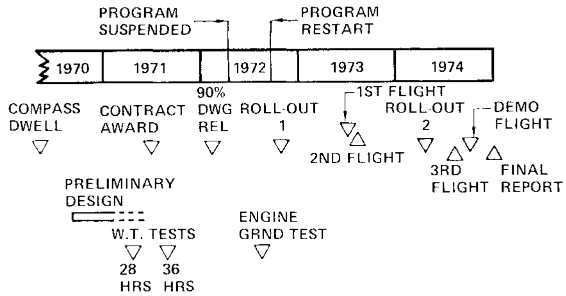


Fig. 4 Compass Cope prototype program schedule of events.

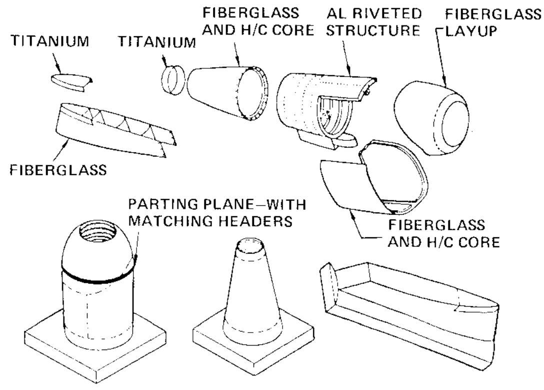


Fig. 5 Preliminary design tooling plan.

and concentrated effort on RPV flight safety did take place. This work was in the form of data link and guidance redundancy and situation displays/signals plus corrective action controls to the pilot. However, the off-the-shelf data links inherent to the Cope definition had limited channels which in turn limited the sum total of displays and controls available to the remote pilot.

Successful remotely piloted landings of light planes by Boeing led to the Cope development and were a part of the initial management/marketing Cope concept. However, consideration was also given to autoland systems as a potential for improved prototype and operational flight safety. The autoland system cost estimates were in the neighborhood of 20% of the initial Cope budget. The design team decided that autoland was necessary for operational vehicles but was too expensive for the prototype.

C. Engine and Power Plant Design

Cope performance requirements of 1000 lb payload, 24 h above 55,000 ft were a result of what could be done with existing engines and advanced technology airframe concepts. Many engines in the 4000-10,000 lb class were considered. These were over a broad range of development, service experience, altitude qualification, and cycle characteristics. A J-97 turbojet (5000 lb sea level static rating) combined with advanced airframe concepts could meet the Cope demonstrator objectives of 24 h above 55,000 ft. The J-97 was high altitude qualified, available, and on loan without fee for the demonstration program. The J-97 was not a step in the development of a production power plant installation for Cope but that was accepted.

The engine, nacelle and airplane arrangement chosen was an overwing pod, (Fig. 6) mounted directly to the wing structure. This podded arrangement had the following characteristics/objectives: 1) engine cooling system definition without testing and without risk; inlet and nozzle definitions without testing and without risk: there would be no inlet flow distortion or airframe contributed nozzle losses; 2) freedom from interference drag with other components; 3) future alternate engines of any size or weight could be pod mounted in the same location without introducing configuration problems in other components; 4) no interference with radio signal reception of body mounted antennas; and 5) engine pod development costs approximately 10% of Boeing experience on turbojets without thrust reversers.

Of these characteristics, perhaps the most significant objective and achievement was the very low power plant development costs relative to Boeing's previous experience. This was achieved through reduction of criteria as compared to long life manned airplanes and through appropriate choice of materials and manufacturing processes. Fiberglass was chosen as the basic nacelle structure for all components except engine support structure and high temperature components. An engine cooling system was chosen with "no test," "no risk" characteristics that included a fiberglass nacelle afterbody fairing. The only nacelle exterior that was not fiberglass was the basic support ring and the ejector nozzle. The engine support ring was built up from existing extrusions and single curvature sheet metal. The elimination of sheet metal forming on this component was made possible by straight lining the nacelle lines over the structural ring portion of the nacelle. A 2 deg break in contour at the structural ring was accepted for this major manufacturing advantage. Engine cooling was a risk area since lack of time and funds precluded model testing. To minimize this risk, the cooling system inlet was located to eliminate any boundary-layer ingestion. The cooling air inlet was unsightly as compared to a boundary-layer ingestion inlet but it was a technically assured low cost solution. The inlet contained an internal diffuser to assist in pressure recovery before dumping to the inlet cowl plenum. The inlet cowl plenum was large in volume relative to the cooling flow and permitted installation of uniform, 360 deg distribution holes into the compartment surrounding the engine. The ejector nozzle generating cooling flow was conservatively chosen from NACA Rept. RME53J13, dated Jan. 20, 1954. It was made divergent corresponding to full expansion at cruise and loiter conditions. This ejector nozzle was titanium and was bolted to the fiberglass shell. The engine exhaust pipe was insulated to keep heat away from the fiberglass nacelle afterbody shell and to prevent exhaust gas temperature reduction. The unshielded exhaust pipe could reduce the exhaust gas temperature sufficiently to increase the fuel consumption by 3%. Tests for cooling adequacy consisted of "hand" feeling the exterior shell during engine ground run. The only warm spot discovered was on the fiberglass shell near the nozzle. A simple internal modification removing a flow blockage eliminated the warm spot.

The maximum flight Mach number of 0.6 led to a very simple inlet design. The inlet entry velocity could be kept moderate without inducing spillage drag, and the inlet lips could have large radii. The combination of circular inlet of moderate entry velocity and large lip radius assured distortion free flow to the engine without test and without risk. The design group was satisfied that inlet caused flameouts could be eliminated without an inlet drag increment. The podded installation permitted a circular nozzle with negligible airframe flow impingement and with no base drag.

The engine pod location immediately above the wing and with large pod curvatures on the bottom to enclose the accessories results in a potential for high local velocities in the wing-body-nacelle juncture. The interference drag from this region was minimized by: 1) straightlining the body lines through the root airfoil to eliminate body generated super velocities; 2) positioning the leading edge of the nacelle strut aft of the maximum nacelle cross section; 3) cantilevering the engine except for tail pipe completely forward of the engine support ring; 4) area ruling the nacelle strut by extending the strut closure aft of the nozzle exit plane; and 5) upward cant of the exhaust nozzle which minimized nacelle to body divergence angle while at the same time, positioned the exhaust flow well above the tail surfaces.

The engine location above the wing had no interference with radio reception. The podded location near the balance center of the airplane made for obvious adaption to alternate power plants without configuration disruption.

In summary, the power plant decisions were relatively easy because there were no apparent alternate solutions and no apparent risk areas or problems.

D. Wing Design

Selection of the J-97 as the Cope demonstrator power plant with its turbojet specific fuel consumption required the compensation of an extremely efficient wing (*L/D*, weight, fuel volume) to meet Cope demonstrator performance. The design team accepted the following concepts in choosing the wing.

1) Wing costs are related to the type of mission, simplicity, or complexity of design, and within limits, unrelated to size or

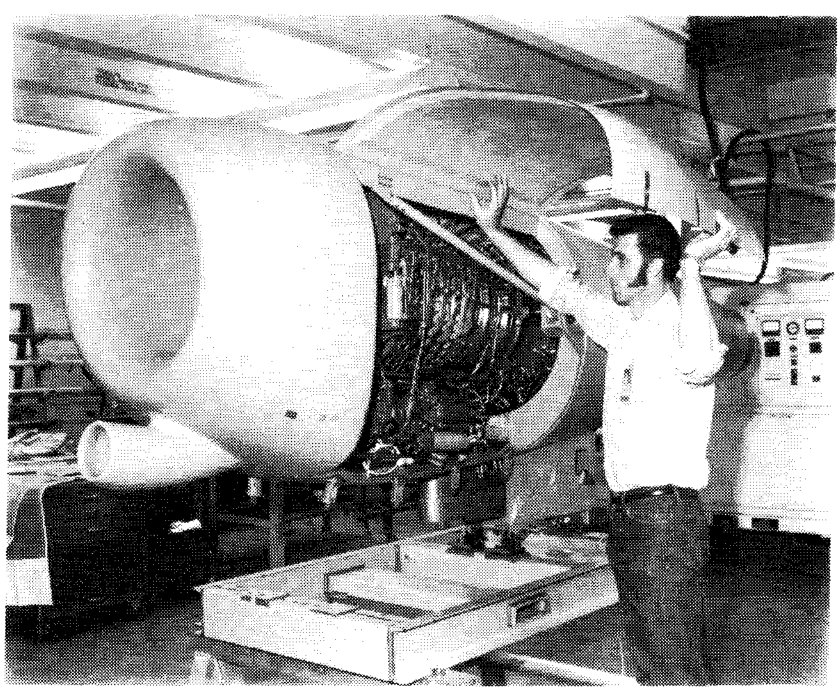


Fig. 6 J-97 over-wing pod.

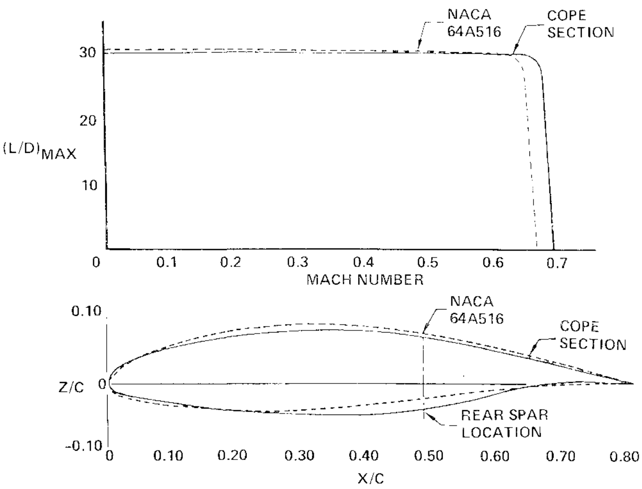


Fig. 7 Airfoil selection.

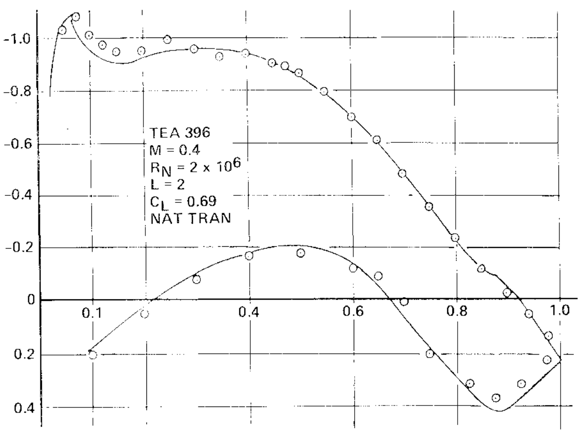


Fig. 8 Compass Cope airfoil Cp vs X/C for theory and experiment.

weight. This cost philosophy is a deviation from normal cost estimating methods.

2) Wing loading of the design must be based on conventional M^2C_L values of approximately 0.20. The airfoil design would be only a small deviation from older NACA sections (Fig. 7).

3) Detail design of the airfoil section would be done by computer using both Mach and boundary-layer effects. The pressure gradients at 0.6 Mach and 0.65 C_L would be such as to permit 50% of natural laminar flow in the presence of a smooth surface (Fig. 8).

4) Basic wing structure would be aluminum skinned, fiberglass core honeycomb (Fig. 9). This combination would provide surface smoothness permitting laminar flow, was structurally optimum for the loads, and provided light weight, insulated (fiberglass core), integral fuel containment.

5) Wing planform would be such that additional span and internal fuel could be added at any time in the program without changing the basic tooling.

6) Honeycomb panel inner skins would be contoured and joined at the edges to the outer skins to prevent fuel contamination of the honeycomb. Inner and outer face sheets of the honeycomb would be constant gage for any one sheet to eliminate cost and complexity of chem-milling.

7) All variation in required wing bending material would be designed into the spar caps since spar cap machining was a low cost concept as compared to honeycomb panels with variable skin gages.

8) Spar caps and rib caps would be hot bonded to the skins with no mechanical fasteners. Upper wing panels would be one piece but lower wing panels would split at the midspar to facilitate assembly. These structural concepts provided the potential for laminar flow to 50% chord. There was no weight penalty or significant manufacturing cost to provide laminar flow potential. Wing structural concepts would be the same even if laminar flow were not operationally desirable.

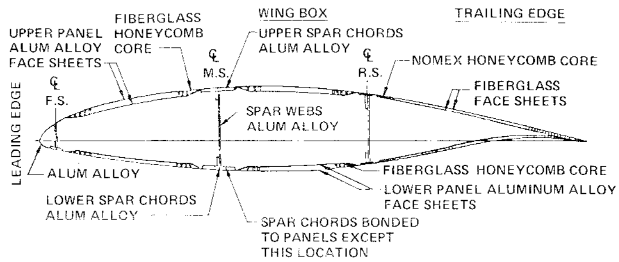


Fig. 9 Typical cross sections inboard wing, approximately BL 70 shown.

9) Wing area would be chosen for the fuel volume needed to meet the 24 h endurance requirement with J-97 engines.

10) The wing would be split in three equal span pieces with a constant chord center section and tapered outer panels. The airfoil section would be unchanged from root to tip with no twist. Spars would be on constant percent lines. Subassemblies of skins, spar and rib caps would be all bonded, but final assembly would be mechanical fastener. The constant chord center section allowed the wing structure to be continuous through the body intersection and greatly simplified the wing-to-body and nacelle-to-wing attachment.

11) The very real hazard of fuel freezing was unique to low speed, high altitude, long endurance designs. Fiberglass honeycomb cells were chosen for the wing panels specifically for insulation. Aluminum honeycomb core was a much lower manufacturing risk, but it had seven times the heat conductivity of fiberglass core. Inner and outer skins of the honeycomb panels were selected as aluminum for fuel containment, weight, and smoothness reasons. Even with the fiberglass core insulation, the heat transfer was sufficient to freeze the fuel toward the end of the flight. Additional heat needed to be added to the fuel. A satisfactory design solution was achieved by cooling the engine oil with fuel and running the warm fuel pipes nearly full span at the low spots in the fuel tank near the lower center spar cap.

12) The wing was sized to hold the fuel required for a 24 h mission using the J-97 engine. This initially resulted in a 84 ft span design. As the preliminary design progressed, the requirements and solutions grew to where the performance was marginal. The constant chord center section was extended 6 ft providing an additional 10% fuel volume and increasing the L/D by 6%. This was done with no skin gage, spar fittings or other changes except for length of skin panels and spar caps. The distributed fuel and wide tread landing gear permitted such a simple change to take place. Flutter and stability and control characteristics remained acceptable.

The wing design proceeded smoothly, primarily because the design team was completely prepared to assume primary structure bonding and honeycomb as basic design concepts, and because Boeing had a 22 × 55 ft autoclave for hot bonding. The Cope technical staff preliminary design supervision was structural dynamics and loads oriented which provided the early and necessary attention to these critical aspects. Although the wing was a major departure from past experience, the wing contribution to program risk was believed to be low.

E. Landing Gear Design

No program funds were available for a landing gear design and development specifically for Cope. It was mandatory that Cope use a landing gear already in production. Design team thoughts relative to the gear were: it must be a wide tread tricycle to provide lateral support of fuel and to assure safe, wing low, and crabbed landings; and it should be forward or aft retracting to insure wing box and body simplicity.

A review of light civil and military aircraft showed the Rockwell Aero Commander gear to meet the objectives for an airplane in the 15,000 lb class. These gear were available at production prices and had ideal geometries for the Cope design.

The Rockwell Aero Commander gears were mounted wide on the central wing section for lateral stability and to eliminate potential vortex interference with the twin vertical tails. The drag of aft mounted gear pods was known to be very low. It was most fortunate to find a production gear ideally meeting requirements. There was no other production gear satisfying the Cope team's ideas of good design practice.

F. Body Design

The primary purpose of the body was to house the primary mission equipment and the control and guidance systems. To achieve this objective, the following decisions evolved.

1) The body would be fiberglass basic structure making it radiowave transparent for placement of internal antennas. The fiberglass body was also a least-cost solution since the compound curves of the forebody could be easily made using a low cost mold.

2) Propulsion, main landing gear and fuel system must be located outside of the body and with no lower hemisphere interference with the body antenna systems.

3) To simplify manufacturing and access, the body would be axisymmetric with a split line on the horizontal C (Fig. 10). Wing and tails would be mounted above the C and the only metal elements in the lower body half would be the nose gear. Antenna interference due to the nose gear was acceptably small.

4) The symmetric horizontal split line made it possible to use two full length aluminum extrusions at the upper and lower body joint. These extrusions served the multiple function of primary longerons, upper and lower body connector, and hard points for the wing and tail attachment. Wing and tail attachments were very simple due to the full length extrusion and the straight-through, constant chord structure of wing and tail.

The axisymmetric body with a symmetrical horizontal split line was the result of major interaction between engineering and manufacturing members of the design team. This concept remained unchanged throughout the program life and adapted easily to a broad range of missions and equipment. Having the landing gear and propulsion out of the body allowed alternate mission equipment such as side looking radar to be pod-mounted on the lower belly with perfect balance and low drag.

G. Empennage Design

Cope tail surfaces had very modest stability and control requirements to meet because of the single engine, small c.g. range, and common low and high speed configuration. To insure the tail's successful functioning, the following decisions were made.

1) All flying V-tails were eliminated from consideration because of unknown power-on effects and lack of power-on test time or funds.

2) The flat horizontal tail and twin fins were chosen to eliminate risk due to power effects.

3) The twin fin flutter problem would be solved with low sweep (forward c.g.) of the fin and inboard tip cant.

4) The horizontal tail would be constant chord for cost and tip stiffness reasons.

5) The twin fin, horizontal tail intersection was a classical aerodynamic interference problem which would be solved by an intersection fairing.

6) Variation in pitch trim moments were low, which along with cost and drag considerations, eliminated the requirement for a movable stabilizer.

7) The landing gear pod would be placed on the wing outboard of the empennage to eliminate the risk of unfavorable tail interaction with the pod wake.

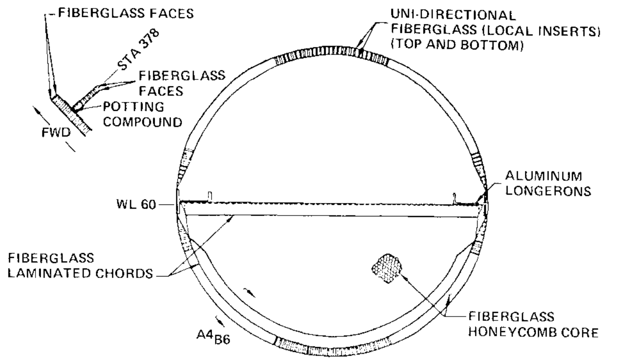


Fig. 10 Typical body cross section rear view.

8) Drag brakes would be placed on the outer wing in a manner to eliminate wake, buffeting, and pitching moment effects on the tail.

The all fiberglass tail which resulted from these decisions was low cost, and had no schedule or flight test risk.

H. Flight Control Design

The most unconventional aspect of Cope flight controls was the cost-necessary dictation of existing 1/10 hp electrical control surface actuators. These low power actuators for a 90 ft span, 400 knot airplane caused the Cope design team significant concern. These concerns resulted in the following decisions.

1) The Cope design would have positive short period, surface fixed, stability on all axes even though it totally relied on autopilot functioning.

2) All structural dynamic modes would be damped with no autopilot input to the control surfaces and checked for characteristics with autopilot on.

3) All flight controls including drag brakes would be aerodynamically balanced to near neutral computed hinge moments. Balancing concept for primary controls would be internal balanced plates plus flexible seals.

Cope was capable of flying at altitudes above 70,000 ft for short missions or for end of mission operation. At these altitudes, the engine minimum rpm limit increased to the rpm required for level flight. Pulling back the engine risked a flame-out. Adding drag by stalling or diving also imposed risks. Drag brakes were chosen as a descent device from extreme altitude until conditions were reached where safe throttle reductions were possible. It was a natural action of the Cope design team to examine the wide variety of sail plane dive brakes as Cope options. The drag brake chosen was a modified Italian sail plane concept. This concept had very simple actuation and structure, maximum drag area per foot of span, and negligible hinge and pitching moments. Drag brake design choices included central, interconnected actuation and an outer panel wing location clear of the empennage. The drag brake system appeared to be without risk.

Flight safety concept for the flight control system were dualized antenna links, autopilots, and surface actuators. These were operated as separate, single channels. The pilot had the choice of switching from one complete system to the other complete system upon suspicion of a malfunction. Automatic voting, automatic circuitry, and airborne component malfunction detection systems were not chosen for cost, data link limits, and reliability reasons. The airframe design team had confidence that the defined control system was low risk when provided with assured and appropriate commands.

Wind Tunnel Tests

The only budget for testing was 40 h in the Boeing transonic tunnel in early 1971. Cope low and high speed configuration was identical and simple which reduced the testing requirements. However, the time allocated was less than 1%

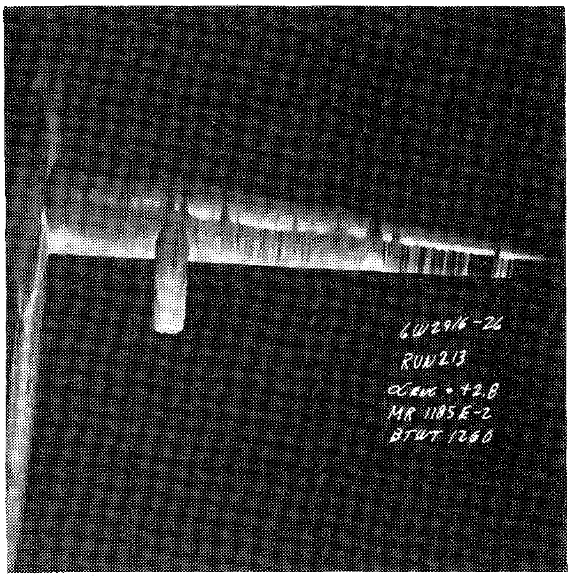


Fig. 11 Laminar flow tests, Boeing transonic wing tunnel; Reynolds number, 1.5×10^6.

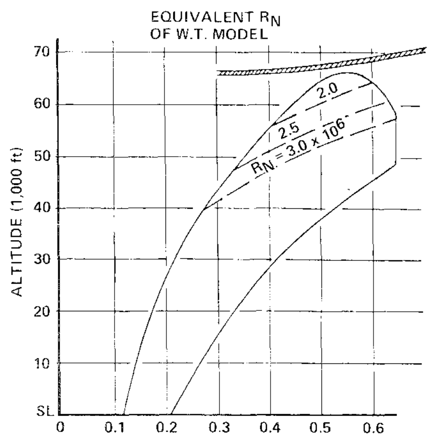


Fig. 12 Compass Cope Reynolds number.

of that normally used by Boeing in manned configuration development. These tests provided the following confirmation: 1) 50% wing laminar flow was achieved at 60% of full scale Reynolds number (Figs. 11 and 12). 2) The airfoil pressures matched theoretical prediction. 3) Nacelle pod interference and Mach drag was acceptable (see Fig. 13). 4) Inlet pressure recovery had only skin friction losses for all important velocity ratios and angles of attack. 5) Tail intersection fairings were required to eliminate interference drag. 6) Stability and control characteristics were stable, linear, and as predicted. The landing gear pods had no effect on directional stability. 7) The drag brake had the predicted high value of drag and minimal pitching moments. 8) Wind tunnel L/D corrected only for mounting plate tare was 29.

The configuration external geometry was proven to have no deficiencies in this test. This allowed the detail design to proceed with surety and confidence even with a very austere funding level.

Project Design and Manufacturing (1971-1973)

Upon demonstrator contract award in June 1971, most of the preliminary airframe design team transferred to the newly formed project organization. New supervision and manpower were added to provide the muscle necessary for program implementation.

The first action of the Cope project organization was to inspect the previous design decisions and make improvements

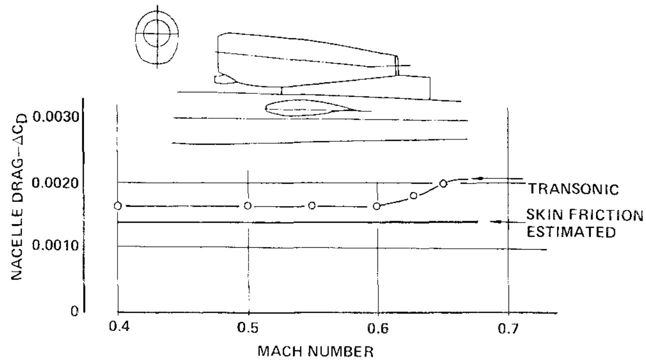


Fig. 13 J-97 nacelle drag.

as necessary. Transfer of the preliminary design team into the project organization made the inspection and improvement process efficient. The results of the inspection and improvement process were changes in the design details but no component or total concept change. The Cope project design team concurred with the preliminary design configuration rather than being forced by schedule to accept the initial definitions.

Manufacture of the prototype parts proceeded for the most part on the cost and schedule plan estimated in preliminary design phase. Two examples of airframe design decisions that caused problems in manufacturing implementation were:

1) Bonding of complex inner wing skins to the contoured fiberglass honeycomb. This was a new experience for Boeing. More bonding stages were required than initially envisioned and there was significant difficulty in assuring conformation of the skins to the contoured honeycomb. A better design decision would have been to use constant thickness core with additional density in those regions of sparcap attachment. The initial decision to avoid square edge honeycomb because of fuel contamination problems could have been wrong, especially for RPVs with their more limited expected life than manned airplanes.

2) Eliminating the landing gear doors and using a rubber boot seal for the wheel to airframe fairing. The Rockwell Aero Commander landing gear had hydraulic retraction power adequate to overcome the gear weight, but it did not have the excess required to overcome the rubber boot seal resistance. With the boot cut away to minimize the resistance, it was no longer a good aerodynamic fairing. Landing gear doors would have been a lower drag solution with less program cost.

These and other possible mistakes did not prevent a general feeling for the total correctness of the project airframe design. This freedom from significant deficiencies resulted in high morale leading to cost and schedule benefits in the combined design and manufacture of the airframe.

Flight Test

Compass Cope has been initially scheduled to fly 18 months after contract award. A 5-month program shutdown, caused by cessation of funding, extended the first flight date to July 28, 1973.

Taxi tests prior to the first flight exposed a major problem in the drag brake system. At midthrow position, at 70 knot airspeed, a divergent oscillation occurred which structurally damaged the drag brake actuation support components. The instability was potentially due to wake separation frequencies at 70 knots and partial deflection being equal to the natural frequency of the drag brake system (Fig. 14). It was a classical structural dynamics problem that had been overlooked in the design process. Installation of rotary dampers attached directly to the drag brakes changed the characteristics to an acceptable, damped oscillation.

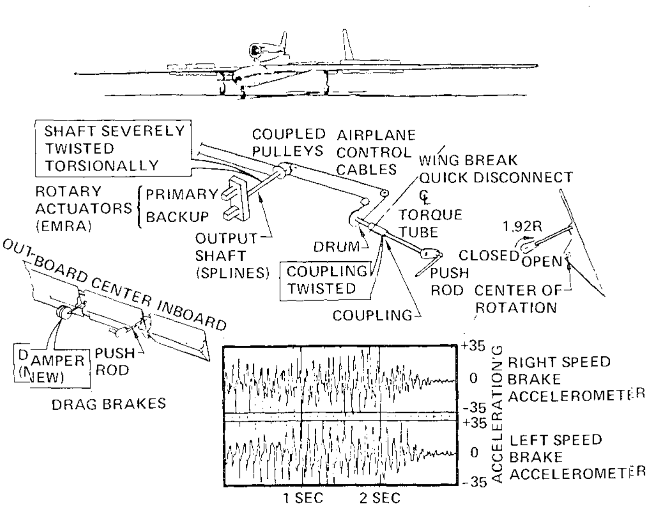


Fig. 14 Compass Cope, drag brake oscillation.

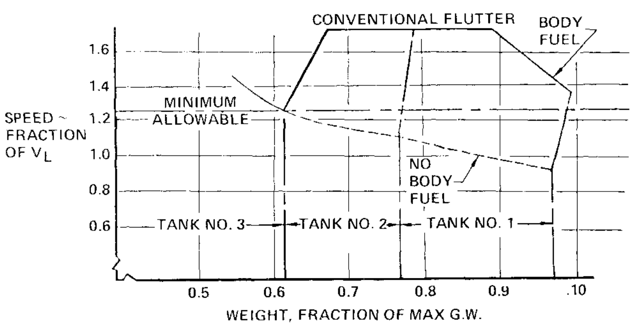
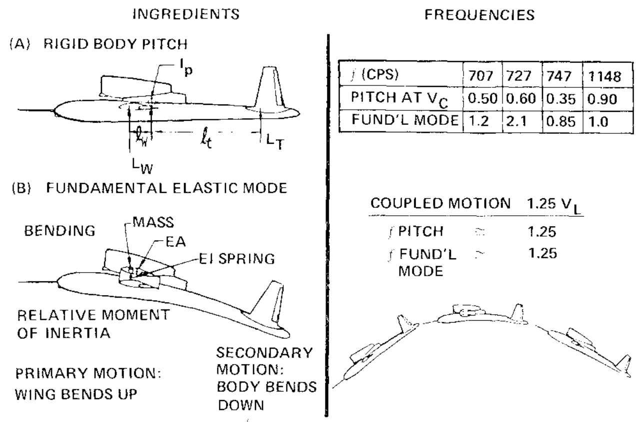


Fig. 15 Pitch bending instability, fuel usage.



f (CPS)	707	727	747	1148
PITCH AT V_C	0.50	0.60	0.35	0.90
FUND'L MODE	1.2	2.1	0.85	1.0

Fig. 16 Pitch-bending instability.

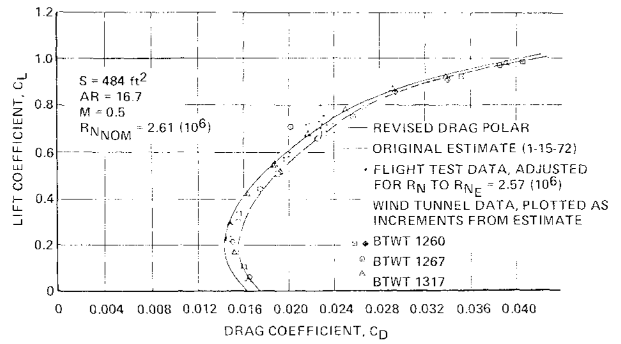


Fig. 17 Comparison of predicted drag with wind tunnel and flight results.

The first flight was without incident. The flying qualities were tame and as predicted. Take-off and landing transition were not problems. The widetread tricycle gear with nose wheel steering and good main wheel braking produced superior taxi and transition characteristics. The Cope team members were pleased and full of confidence.

The second flight on Aug. 4, 1973 had an expanded program to provide verification data for the following long endurance demonstration flight. This test was run entirely on battery power since the engine powered alternator did not function at the low rpms of medium altitude flight. The flight was scheduled for 4 h and the battery capacity was estimated at 4½ h. Fuel capacity was loaded for approximately 5 h.

This flight went very smoothly through all of the altitude test conditions. The speed brake oscillation was acceptably damped at all speeds up to 140 knots indicated airspeed. The series of altitude tests shows all test characteristics to be clean. The test conditions took slightly more time than planned. Descent with speed brakes out was uneventful except for growing concern about remaining battery capacity and fuel quantity. It was decided that a go-around would be an excessive battery and fuel risk and that the approach must proceed to landing. The final approach was confused by an angular misalignment of the airplane axis with the runway centerline. There was no cross wind, the rudder pedals were neutral, the airplane was pointed toward correcting runway lateral misalignment, but it would not make the necessary lateral displacement correction for touchdown on the runway centerline.

Postflight examination revealed that two minutes prior to flight termination, a lateral accelerometer had failed with a resultant hard-over right rudder. This could not be diagnosed from the remote cockpit. The stick and rudder pedals were both in neutral, implying no surface deflection. The final approach had been made, unknowingly, with a full hard-over rudder.

The pilot's station and the autopilot system were redesigned to prevent future serious incidents from developing from this type of autopilot failure. This redesign was inadvertently tested during flights of Cope 2 when the elevator went hard over and automatic switching to the backup system recovered the airplane from a 4-*g* maneuver. The airframe was undamaged, providing significant proof of Cope bonded structure integrity.

An undamped "seagull" mode of body-wing bending (Fig. 15) pitch axis motion had been theoretically solved by installing a small aft body fuel tank (Fig. 16). This and all other structural dynamic characteristics were satisfactory in flight test.

Flight test drag characteristics were estimated using rpm, outside air temperature, altitude, Mach number, and engine manufacturer's thrust values. The flight L/D resulting from this method was in excess of 30 (Fig. 17). This value could imply the achievement of 50% wing laminar flow intended with the airfoil and surface smoothness design. However, the thrust instrumentation was less than required for good accuracy so that L/D values and laminar flow proof are not assured. Boundary-layer measurements were not taken which would have given proof of wing laminar flow.

Lack of proof of laminar flow did not prevent the design team from forming certain positions. 1) Getting the wing surface to laminar flow smoothness quality was not a major flight test problem. 2) Operational laminar flow for special missions appears practical but daily routine missions may not warrant the constant care required. 3) Designing Cope for laminar flow was judged as neither a virtue or mistake.

Remotely piloted flight testing was on a much reduced schedule and, consequently, more limited on knowledge acquisition than with a manned airplane. The Cope airframe design team was a party to the decision to eliminate man-aboard capability. If man-aboard capability had been a solid requirement, there would probably have been no contract award and no flight vehicle. Having the chance to fly with a remote pilot was a desired adventure.

Summary

The Cope design and flight test resulted in the following major airframe accomplishments: 1) podded single engine airplane arrangement that is a potential standard for future single engine high altitude surveillance designs, manned or unmanned, 2) composite and bonded primary structure with maximum use of honeycomb and fiberglass, 3) integral wing fuel containment within honeycomb basic structure, 4) fiberglass core wing panels for fuel insulation, 5) potential for operational laminar flow, 6) superior performance and flying qualities with essentially no test program, 7) propulsion installation costs at a fracture of previous experience due to use of fiberglass as the primary cowling material, 8) Low total airframe and flight test costs due to absence of significant defects and need for change, and 9) strong program marketing position due to adaptability of the design to a wide variety of missions without modifying the airframe arrangement.