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Development and Testing of the Perseus Proof-of-Concept Aircraft

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Report: AR-9301
Title: Development and Testing of the Perseus Proof-of-Concept Aircraft
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ABSTRACT

Many areas of global climate change research could benefit from a flexible, affordable, and near-term platform that could provide *in situ* measurements in the upper troposphere and lower stratosphere. To provide such a capability, the Perseus unmanned science research aircraft was proposed in 1989. As a first step toward the development of Perseus, a proof-of-concept (POC) demonstrator was constructed and tested during 1990 and 1991. The POC was a full scale Perseus airframe intended to validate the structural, aerodynamic, and flight control technologies for the Perseus within a total budget of about \$1.5 million. Advanced propulsion systems needed for the operational Perseus were not covered in the POC program due to funding limitations. This report documents the design, development, and testing of the Perseus POC.

Key words:

Unmanned air vehicle
Atmospheric science

Figure i: November 1991. The Perseus proof-of-concept (POC) in flight over El Mirage, CA.

Figure ii: April 1991. POC engine testing in Virginia.

Figure iii: Wing proof test of POC wing.

Figure iv: Inside view of POC fuselage without skin covers showing (L-R) engine bay, fuel tank, avionics, payload bulkhead, dropsonde dispenser, ballistic parachute, and batteries.

Figure v: Three-view of the Perseus POC.

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A number of individuals contributed significantly to this program. First and foremost is James G. Anderson of Harvard, who has provided the vision and the scientific expertise that initiated and guided the Perseus program in its efforts to bring flexible and affordable new tools to all members of the science community. Jarvis Moyers, Sherwood Rowland, Joe Knox, Mac McFarland, and Bill McCabe backed the program by arranging funding. Lenny Solomon, Joyce Makrides, and Alisa McManus helped the funds arrive on a timely basis.

A large number of individuals participated in the Perseus POC design process. Mark Drela provided analysis and design guidance throughout the program. Siegfried Zerweckh, Peter Kämpf, and Peer Frank designed the aircraft structure. Benjamin Russ had primary responsibility for the propulsion system. Scott Brown, Clinton Church, and Steven Waide were the primary fabricators. Tom Clancy, Peter Bahniuk, Martin Gomez, Dave Vos, and Chuck Walcher designed, built, and tested the flight control system. Steve Battelle, Alan Coconi, Jim Murray, Joe Demusz, Steve Finberg, and Conrad Hoepfner formed an ad-hoc outside review team that provided many valuable suggestions. Greg Zack and Tim Just piloted the aircraft. Dave Vos, Benjamin Russ, Clint Church, Martin Gomez, and Caroline Becker assisted in the preparation of this report.

DEVELOPMENT AND TESTING OF THE PERSEUS PROOF-OF-CONCEPT UNMANNED AIRCRAFT

1. BACKGROUND

1.1. Motivation

In a landmark paper, (Molina and Rowland 1974) hypothesized that chlorine released from the breakdown of CFCs in the stratosphere might initiate catalytic reactions that could result in significant losses of ozone. (Farman and others 1985) reported that ground-based observations indicated a cyclical decline of ozone each spring within the Antarctic vortex, a phenomenon subsequently dubbed the "ozone hole". Confirmation that this annual ozone loss was indeed induced by chlorine chemistry, with the chlorine released from a breakdown of CFCs, came largely as a result of the 1987 Airborne Antarctic Ozone Expedition, when sensitive instruments were flown through chemically active regions aboard NASA's ER-2 and DC-8 aircraft (Anderson and others 1988). The evidence from this and other studies led subsequently to the Montreal Protocol under which many nations agreed to limit and then phase out CFC production (Rowland 1989).

The ability to conduct simultaneous, *in-situ* measurements of chemical, radiative, and dynamic properties rather than observing remotely was a key in diagnosing the cause of the ozone hole. The tools used in the Antarctic, however, subsequently proved more limited in other regions of the world. The manned ER-2 aircraft can reach an altitude of about 20.5 km (68,000 feet) - high enough to reach the chemically active regions in Antarctic, but not in the Arctic or at lower latitudes. Further, as a manned aircraft, the ER-2 is subject to numerous limitations in the interest of flight safety, as well as limited range and duration. Supersonic aircraft such as the SR-71, which holds the world record of 25.9 km (85,068 feet) cannot be used for atmospheric research because their shock waves would destroy sensitive free radicals before they could be measured. Scientific balloons are able to reach altitudes as high as 40 km (131,000 feet) but launch opportunities are limited by the need for extremely calm conditions at the surface, while their flight trajectories are determined by prevailing winds rather than by scientific measurement strategies.

Following the 1987 AAOE missions, Prof. James G. Anderson of Harvard University was asked by NASA headquarters to study ways of measuring free radicals deeper and higher into the Antarctic vortex than was possible using the ER-2. The idea was to develop a lightweight version of Anderson's chlorine monoxide instrument, and launch it on a small balloon from McMurdo Station in Antarctica. The goal was to produce vertical profiles of chlorine monoxide (ClO), bromine monoxide (BrO), ozone (O₃), temperature (T), and pressure (P) from sea level to about 25 km. Due to the costs of recovery and the need to build several instruments to cover loss or damage in recovery, Anderson estimated the cost of a 24-flight campaign at roughly \$3 million.

Anderson's studies coincided with the conclusion of the MIT Daedalus Project, a three year program in education and research that resulted in a record 116 km (72 mile) flight under human power. The extremely low Reynolds numbers at which Daedalus flew were typical of those that would be encountered by an aircraft flying very high in the atmosphere, so the aerodynamic and structural advances from Daedalus offered new possibilities for high-altitude aircraft. In 1988, Daedalus project manager John Langford studied the feasibility of a solar-powered derivative of Daedalus (Langford 1989). As an interim step, he considered a technology demonstrator powered by lithium thionyl-chloride batteries. Conversations between Anderson and Langford in 1988 and 1989 indicated that such a battery-powered aircraft might have significant utility in itself, as a substitute for balloons in Anderson's proposed Antarctic expedition. Thus, in the spring of 1989 a project was launched to develop a lightweight unmanned aircraft specifically for atmospheric science research. The effort was dubbed Perseus, and a new organization, Aurora Flight Sciences, was formed to conduct it.

1.2. General applications

As other scientists learned of Perseus, it quickly became clear that such an aircraft might have other applications besides measuring vertical profiles of ozone and its related chemistry. In 1990 Aurora initiated a systematic effort to identify and explore other areas of global change research that might benefit from a flexible, low-cost platform (Anderson and Langford 1991; McGeer and others 1990). Beginning with the research priorities identified by the U.S. Committee on Earth and Environmental Sciences, nine primary areas were identified where unmanned aircraft could be an enabling technology making possible key scientific measurements not otherwise attainable in a timely or affordable manner. These areas include:

Stratospheric ozone depletion. The 1985 discovery of the Antarctic Ozone Hole shocked both the general public and the scientific community. Not predicted by any models, the chemical causes of the hole were determined through careful aircraft-based investigations in 1987. Subsequent missions to the Arctic in 1988/89 and 1991/92 have revealed record levels of chlorine monoxide, and shown the potential for massive ozone destruction in areas where large percentages of the world population resides. Yet of the twelve catalytic cycles identified to date that destroy ozone, only five have been measured in the atmosphere - and these five only over about 30% of the altitude range in which they are known to be effective. What is urgently needed is a means of making directed measurements of key chemical species at altitudes between 20 and 30 km - a mission ideally suited to unmanned aircraft.

Global Warming: the role of clouds in the earth's radiation budget. Water vapor and clouds are the dominant regulators of radiative heating of the planet. Lack of understanding of the nature of the cloud-climate feedback is one of the fundamental sources of uncertainty in climate prediction. Long-term measurements at or near the tropopause, combined with vertical profiles from the ground up to 30 km, are required. In addition, one of the best "greenhouse laboratories" is in the western tropical Pacific. The unmanned aircraft is the only known technology that can provide sufficient temporal and spatial coverage to address this issue.

Hurricane Research and Forecasting. Hurricanes are among the most destructive natural phenomena; the two deadliest natural catastrophes in U.S. history were both hurricanes. In 1989 Hurricane Hugo did \$7 billion in damage to the U.S. Yet present forecasts are so uncertain that large stretches of coastline must be evacuated needlessly, leading to economic loss and complacency. The upper-level winds that actually steer the hurricane are unavailable to researchers: present storm-research aircraft cannot reach higher than about 25,000 feet, while satellites cannot sense the steering winds which operate above the visible cloud formations. An unmanned aircraft flying above a storm and sampling its environment with lightweight, expendable dropwindsondes would lead to substantial improvements in our understanding of hurricane formation (cyclogenesis), intensification, and track prediction.

The Impact of Supersonic Aircraft on the Stratosphere. Twenty years ago, environmental concerns contributed to the demise of the U.S. supersonic transport (SST) project. Today, NASA is planning an updated high-speed civil transport as its centerpiece in aeronautics, with over \$1 billion in spending projected by 1995 and \$10 billion by the end of the decade. Yet many of the same issues from the previous debate remain to be resolved. Specifically, is there a "safe" level of aircraft emissions? How does this level vary with altitude? These questions can only be answered by improved models of the lower stratosphere, calibrated by *in situ* observations of many chemical species. The unmanned aircraft is the only tool that can accomplish this task within the desired time frame (1995) to support a government decision whether or not to proceed with this massive project.

Stratosphere-troposphere exchange. A critical component in the infrared flux field, the basis for predictions of global warming, is the water vapor mixing ratio in the middle/upper troposphere and the lower stratosphere. What is the mechanism responsible for the exchange of water vapor between the stratosphere and the troposphere? What is the turnover time of the stratosphere? What controls exchange of material across the tropopause at low, middle, and high latitudes? The key to answering these questions is the ability to make long-term and wide-ranging measurements throughout the upper troposphere and the lower stratosphere, including multiple flights around and through the large convective towers believed to be the primary mechanism for strat/trop exchange. These tasks are too dangerous and require too great a range/duration for manned aircraft, and thus are ideally suited for unmanned aircraft.

Global sources and sinks of CO₂ and CH₄. Predictions of global warming rates are very inaccurate without knowledge of carbon dioxide and methane sources and sinks - both natural and anthropogenic. Considering the tremendous political and economic forces at work, it is a source of deep concern that even rudimentary understanding of many scientific questions is not available. It is difficult to imagine the nations of the world agreeing to substantive controls of CO₂ emissions before we have defensible data on the current geographical distribution of sources of CO₂ and CH₄ and on the breakdown of emissions between industrial and agricultural practices and between natural and manmade sources. Accomplishing this will require measuring fluxes on a wide range of length scales (from living organisms, to the local ecosystem, to regional, continental, and finally global scale). The unmanned aircraft is the only platform capable of accomplishing these roles using a common set of instruments.

Tropospheric Oxidation. The hydroxyl radical, OH, is in many ways the key element in the atmosphere's ability to clean itself of many chemical pollutants. Because of measurement difficulties, however, there are only nine measurements of stratospheric OH, all taken from one balloon site in Texas. There are even fewer measurements in the troposphere. Ground-based measurements suggest, however, that OH is declining gradually on a worldwide basis. Unmanned aircraft equipped with lightweight laser-based instruments recently developed at Harvard University would be the first tools capable of addressing this crucial problem.

Operational meteorology. As is the case with hurricanes, extratropical forecasts suffer from a grave lack of hard data over the oceans. Satellite-derived winds and temperatures have had little influence on improving forecasts; there are some studies that show a degradation of forecasts initialized with supplementary satellite data. Many of the radiosonde stations, and most all the radiosonde ships, that made up the worldwide observing network in the 1960s have been phased out. A small fleet of unmanned aircraft flying in the lower stratosphere and deploying dropsondes, or perhaps making occasional excursions into the lower troposphere, would provide a vastly improved data base for operational forecasting, at much smaller expense than can be achieved with satellites.

Calibration and support of satellite instruments. A very large fraction of our national resources for global change research are directed towards large satellite programs. How accurate and precise are satellite observations of trace constituents in the stratosphere and troposphere? In order to test the accuracy and precision of data sets from satellites such as UARS and EOS, a serious research effort is required using techniques independent of those on the satellite, within volume elements observed simultaneously by both methods. Since much of the altitude band observed by satellites is above that accessible by manned aircraft, the unmanned aircraft is the primary candidate for performing this role. In addition, unmanned aircraft may play a useful role by flying space-type instruments in the field before they are qualified and launched into space. This technique could significantly shorten and improve the satellite instrument development process.

1.3. A family of platforms

Aurora's studies made it clear that no single platform would meet all the various needs of global change research, and that a family of different designs was required. Building around the initial Perseus design developed for Anderson, Aurora evolved a family of five basic designs, including:

- a small payload (50-100 kg), very high altitude (25-30 km) aircraft
- a medium payload (100-200 kg), high altitude (20-25 km) aircraft with long duration (>24 hr)
- a long-range, low-altitude (<100 m) platform
- a heavy (200-400 kg) payload, very high-altitude (25-30 km) platform
- a small (<50 kg) payload, very long duration (>2 weeks) platform

These designs became the basis for a family of aircraft shown in figure 1.1. The family would be built around a common technology base of lightweight structures, computational aerodynamics, and microprocessor-based flight control systems. Using a common airframe and control system, three operational versions of Perseus could be built: Perseus A, to carry 50 kg payloads to 30 km for short durations; Perseus B, using a turbocharged engine to carry 100-200 kg payloads to 20 km altitudes for 1-3 days, primarily in support of radiation budget studies; and Perseus C, designed for extended flights at low altitudes measuring surface conditions and boundary layer fluxes. Beyond Perseus, a larger-twin engine aircraft known as Theseus would carry 200-400 kg payloads to altitudes of 25 km or above for ranges of 10,000 km or greater. Finally, a solar-powered platform, known as Odysseus, was envisioned that could carry 50-100 kg payloads to 20 km altitudes for periods of weeks or months.

Figure 1.1: Planned lineage of the Aurora family of atmospheric research aircraft.

1.4. Role of the Proof-of-concept

Despite the early endorsements Perseus received from many prominent members of the scientific community, it was unclear where funds for such a project could be obtained. Within NASA, Perseus was viewed as a science project by the aeronautics office, and as an aeronautics project by the science office. The National Science Foundation was reluctant to devote science program funds to an engineering development project, while the conventional wisdom within the aerospace industry and commercial financiers was that the atmospheric science community was far too small a market to justify investment of private capital. Finally, there were obvious questions about Aurora's technical competence and the realism of its cost targets. Previous NASA and DoD unmanned aircraft projects had cost between \$40 million and \$1 billion, without resulting in a single operational system.¹

It was to break this gridlock that the Perseus proof-of-concept (POC) project was initiated in the spring of 1990. The project had three basic goals: 1) to develop and validate the aerodynamic, structural, and flight control technologies upon which future unmanned science aircraft could be built, 2) to prove that viable unmanned aircraft could be built at costs that would be affordable by the scientific community, and 3) to build the infrastructure, experience, and credibility of the technical team at Aurora Flight Sciences. The \$1.5 million cost of the project was shared between five major sponsors: Harvard University, the National Science Foundation (NSF), DuPont, the National Institute for Global Environmental Change (NIGEC), and Aurora Flight Sciences. Table 1 summarizes the key events in the POC's development.

Table 1: Chronology of the Perseus POC program.

1989

April	Perseus concept initiated
June	Full-time work starts. First proposal submitted to NASA and NSF.
September	First design review. Aircraft size frozen.
October	Start of tooling fabrication

1990

February	Initial proposal to NASA is rejected
May	Perseus program split into three parts: POC, A, and B
June	Tooling completed, fabrication of flight structure begins
October	Structure completed. Software coding begins.
November	Fuselage construction begins.

1991

February	Engine test runs begin.
April	Avionics design review. Complete drive train test.
August	POC rollout
September	Taxi testing begins
November	First flight

¹ The sole unmanned aircraft system operational with the U.S. military at the time Perseus was proposed had been developed by the Israelis.

2. PRELIMINARY DESIGN

Most of the preliminary design for Perseus was conducted before the project was split into three sections. The considerations that sized the POC, therefore, are those for the high-altitude mission.

2.1. Design requirements

The original design goals for the Perseus project were:

- a 50 kg payload capacity (weight of the Anderson ClO/BrO/O₃ payload)
- 25 km altitude capability (minimum needed for polar vortex research)
- total program cost (24 flights) of under \$3 million
- unobstructed airflow for the nose-mounted ClO instrument
- slowest possible flight speed consistent with wind margin (to minimize ducting for instrument)
- provisions for shipment to and launch from McMurdo Station, Antarctica

2.2. Design considerations

In contrast to most other aircraft designs, the initial mission specifications for Perseus placed a premium on altitude rather than on range or duration. To maximize altitude requires:

$$\begin{aligned} h &= \int_{\text{take-off}}^{\text{fuel exhaustion}} v dt \\ &= \int_{\text{take-off}}^{\text{fuel exhaustion}} \frac{\text{engine power} - \text{drag power}}{W} dt \\ &\approx \frac{E}{mg} - \int_{\text{take-off}}^{\text{fuel exhaustion}} \frac{\text{drag power}}{W} dt \end{aligned}$$

where W is the gross weight, and E is the total energy delivered by the propeller. Notice that according to this policy, fuel is exhausted at the top-of-climb, and the aircraft must glide back to base. This is the plan for Perseus. The power required to balance aerodynamic drag is a function of air density (ρ , determined by altitude), wing loading (W/S), aerodynamic endurance parameter $C_L^{3/2} / C_D$:

$$\text{drag power} = \sqrt{\frac{2}{\rho}} \sqrt{\frac{W}{S}} \frac{C_D}{C_L^{3/2}} W$$

The air density, ρ , is determined primarily by altitude. At 25 km, the standard atmosphere quotes a density of 0.0394 kg/m³, about 3% that at sea level. Since Perseus was designed for use in polar conditions, a "high latitude, winter" version of the standard atmosphere was used (N.O.A.A. and others 1976). In this case, the density at 25 km is 0.0334. The primary design variables are (in approximate order of importance) thus low weight, high lift coefficient, low drag, and large wing area. These are generally expressed as wing loading (W/S) and "power factor" ($C_L^{1.5}/C_D$). In the case of Perseus, low wing loading is achieved primarily by the use of composite materials in a lightweight sandwich construction, while power factor is achieved through extensive use of computational codes - both derived directly from the experience of the Daedalus human-powered aircraft.

2.3. Propulsion options

At the slow speeds needed by Perseus for scientific measurements, a large, slowly turning propeller is the most efficient means of aerodynamic propulsion and was the only option considered for

Perseus. The power produced by an unsupercharged, four-stroke engine at constant speed is given as a function of altitude by (Taylor 1989) as:

$$\frac{P_2}{P_1} \approx \frac{bmep_2}{bmep_1} = \frac{R_i - \frac{fmep}{imep_1}}{1 - \frac{fmep}{imep_1}}$$

where $bmep$ is the brake mean effective pressure, $fmep$ is the friction mean effective pressure, $imep$ is the indicated mean effective pressure (the ratio $fmep/imep$ approximates mechanical losses in the engine core), and:

$$R_i = \frac{P_2}{P_1} \sqrt{T_1/T_2}$$

Assuming the mechanical efficiency of the engine is roughly constant at about 80%, figure 2.1 shows that the power produced by an air-breathing declines rapidly as a function of altitude. There are basically two options for overcoming this effect: either increasing the pressure of the ambient air by turbocharging and/or supercharging, or using a system that does not depend on outside air.

Figure 2.1: Propulsive performance of a standard aircraft engine declines rapidly as a function of altitude. The performance needed for Perseus requires a pressure ratio of more than 60:1. - technically possible but too expensive for Perseus.

During the preliminary design phase, a variety of propulsion alternatives were considered. These included:

- turbocharged internal combustion engine
- oxygen-fed internal combustion engine
- electric motor driven by battery
- electric motor driven by fuel cell
- monopropellant engine

Table 2.1 presents a summary comparison of each option. Monopropellants (hydrazine) were rejected because of handling and shipping hazards. The fuel cell and turbocharged options were judged to be prohibitively expensive for Perseus, while the oxygen-fed internal combustion engine and the battery-driven electric motor were competitive. Early life-cycle cost analyses favored lithium thionyl-

chloride primary batteries, and these were used for all preliminary design and sizing. Continued analysis on the oxygen-fed internal combustion engine showed it to be more flexible in terms of accommodating growth in the mission requirements, especially in desired payload or range, and it later became the system of choice.

Table 2.1: Summary comparison of propulsion options considered for Perseus. "Specific power" is defined to include engine core, prop, transmission, cooling, oil, and ignition systems. "Specific energy" is consumables plus tanks.

Option	Specific Power (W/kg)	Specific Energy (W-hr/kg)	Cost through first 25 flights
Turbocharged IC (1989)	237	3,300	\$1.2 million
Turbocharged IC (1991)	282	4,716	\$2.8 m
Oxygen-fed IC	211	618	\$0.6 m
Battery-electric	342	325	\$0.6 m
Fuel cell electric (LH ₂ /LOx)	284	1,765	>\$10 m
Monopropellant (N ₂ H ₄)	647	339	>\$1 m

2.4. Configuration

The design of Perseus not only had to fulfill the requirements of the physics of flight, but also had to be a versatile and reliable payload platform for the intended users. The requirement for clean flow to the payload dictated a pusher design. The selection of an M-2 shipping container (internal dimensions: 8' x 8' x 20') dictated the maximum allowable size for any component. The goal of rapid field assembly using a minimum number of people and no heavy equipment also suggested limits for the weight of any one component and dictated the design of connection joints.

Within these constraints a variety of configurations are possible. Among those considered for Perseus were:

- single fuselage with aft-mounted prop
- flying wing
- joined wing
- twin-tail design (conventional and inverted V)
- twin engines mounted on wings

Some of these are illustrated in figure 2.2. The twin engine design was ultimately rejected because of the additional weight, complexity, and cost of the additional propulsion system, while the twin tail design was eliminated due to concern for its higher parts count, possible vibration modes, and large boom spacing needed to accommodate the large diameter propeller. The joined wing suffered from excessive drag due to the low Reynolds number of its small-chord lifting surfaces. The flying wing and the aft-mounted prop were judged essentially equivalent in weight (the added structural weight of a swept wing roughly equaling that of a long drive train); although the swept wing was considered attractive for its lower overall parts count, the stability and control aspects of the conventional design were viewed to be simpler and it was therefore selected.

Figure 2.2: Four major configurations considered during preliminary design phase of Perseus. Clockwise from upper left: conventional, flying wing, joined wing, twin tail. Not shown: twin-engine and inverted "V" twin boom tail configurations.

2.5. Sizing

To achieve the lowest operating costs, the original Perseus airframe was optimized for the lowest battery mass that could carry a 50 kg payload to 25 km. A computer program was written that used airfoil properties as a function of Reynolds number and battery energy density as a function of discharge rate and then parametrically varied wing loading and overall power-to-weight ratio, iteratively computing for each case the minimum battery mass fraction needed to reach the specified altitude goal. As seen in figure 2.3, the optimum is fairly flat, so a design point was selected that allowed use of a commercially available electric motor.

At this design point (170 N/m^2 , $P/W = 8 \text{ W/N}$) the original Perseus had a span of 17.9 m, an area of 16 square meters, and a gross weight of 278 kg. A 21.7 kW electric motor was required, to be driven by 85 kg of lithium-thionyl chloride batteries. An aspect ratio of 20:1 allowed good aerodynamic performance while still disassembling into three main pieces for shipment in the M-2 container.

Figure 2.3: Carpet plot used to select Perseus wing area.

2.6. Unique features of the POC

The estimated cost for the custom batteries was about \$500,000 - about 1/6 the total projected budget for the entire Perseus project, and about twice what any one sponsor was willing to put into the project in 1990.

The Perseus proof-of-concept program was therefore initiated to demonstrate and validate the aerodynamic, structural, and flight control technologies needed for Perseus prior to committing the larger investments needed for development of custom high-altitude propulsion systems. The POC was identical to the planned operational variants in size and geometry, with the same tooling developed for the POC usable for later models. The flight control system was to be very similar in architecture to that planned for later Perseus models, except that it was not ruggedized for the high-altitude environment and it lacked certain navigational features. The biggest differences lay in the areas of propulsion and landing gear. With no particular altitude requirement, the POC could use a 35 kW two-stroke engine and wooden propeller sold commercially for use in ultralight aircraft. Also, since weight and drag were of less concern than in later models, a fixed tricycle landing gear (to provide good ground handling) was adopted for the POC.

3. STRUCTURAL DESIGN

3.1 Design load diagram

There are presently no design or certification guidelines for unmanned aircraft. In designing the POC, the basic airworthiness criteria for manned glider designs were used as a starting point, and modified where it was judged appropriate. Thus the aircraft was designed to a positive limit load of 4.7 g's and ultimate load of 7 g's. Subsequent increases in the gross weight of the POC reduced the effective load factor to a limit load of 3.2 g's and an ultimate of about 5.6. The design load envelope is shown in figure 3.1.

Figure 3.1.: Load envelope for the Perseus POC in flight configuration.

3.2 Weight and balance

The weight breakdown for the Perseus POC is shown in Table 3.1.

Table 3.1: Weight distribution of the Perseus proof-of-concept during its November, 1991 flight tests.

Component	Mass (kg)	Mass fraction (%)
Wings	85.3	19.8
Tails	17.7	4.1
Fuselage	77.8	18.1
Landing gear	37.5	8.7
Structure, total	218.3	50.8
Propulsion system	71.5	16.6
Avionics	87.3	20.3
Empty weight	377.1	87.7
Payload	16.2	3.8
Fuel	36.7	8.5
Gross take-off	430.0	100.0

3.3 Construction details

In general, the structure is built from graphite, epoxy, Kevlar, and Nomex honeycomb produced with a wet lay-up in composite molds; the primary exception being the fuselage which was a steel tubular truss covered with a fiberglass fairing. The wing construction owes much to the design and manufacturing approach of German high-performance sailplanes, except that the finished panels are considerably lighter. Layouts of the horizontal and vertical tails are shown in figure 3.2. The wing layout is shown in figure 3.3. A typical wing cross-section is shown in figure 3.4. Construction techniques are summarized in figure 3.5.

Figure 3.2: Layout of the horizontal and vertical tails.

Figure 3.3: Wing layout.

Figure 3.4: Typical wing cross section consisting of Kevlar-carbon-Nomex honeycomb sandwich skins, carbon-Nomex honeycomb sandwich ribs and webs, carbon unidirectional prepreg spar caps, and carbon fabric and tows for local stiffeners.

Figure 3.5: Wing construction.

3.4 Fuselage

The fuselage used a steel truss design covered with a composite aerodynamic fairing. Although slightly heavier than an all composite monocoque structure, the advantages in terms of manufacturing cost and accessibility far outweighed any weight penalties in this application. Considerable use of Structural Research and Analysis Corporation's COSMOS finite-element analysis package was made to optimize the fuselage space frame.

The landing gear was a tricycle based on components designed for homebuilt kit aircraft, the Task *Silhouette*. The main wheels had independent hydraulic brakes, and the nose wheel was fully castored. Fuselage and landing gear are shown in figure 3.6.

Figure 3.6: Fuselage and landing gear.

4. PROPULSION

4.1. Engine

The engine used in the POC was a Rotax 503 CDI, made in Austria by Rotax-Bombardier GmbH for use in ultralight aircraft (see figure 4.1). The 2-cylinder/2-stroke engine has a displacement of 496 cc. The engine is fan cooled. Lubrication is provided by oil mixed in the fuel. The stock breakerless capacitive discharge ignition and E-jet carburetor system supplied by Rotax were used on the aircraft. The engine used on the POC was equipped with an electric starter, but not with an engine-driven alternator.

Figure 4.1: The Rotax 503 engine was used on the Perseus POC.

The Rotax 503 CDI was installed in an inverted position with cylinder heads facing downward and the crankshaft facing upward. This position accommodates the engine in the fuselage engine bay with the gear box output flange in alignment with the drive shaft in the tail boom. To ensure spark plug performance in the inverted position, gold palladium electrodes were used. An inlet on the bottom of the fuselage provides the fan and the engine with cooling air which exits through the opening for the exhaust system.

Figure 4.2: Rotax 503 manufacturer's performance data.

4.2. Transmission

The transmission consisted of (1) a 2:1 speed reduction gearbox, integrally installed on the engine; (2) a flexible coupling scheme (shown in figure 4.3); (3) a 2.55 m carbon-fiber draft shaft, supported by bearings at three locations (shown in figure 4.4), and (4) a prop coupling hub.

To isolate the carbon-fibre drive shaft from the vibrations of the two-cycle internal combustion engine, a coupling had to be designed to allow the engine to move up to the highest vibration amplitudes without carrying any loads into the drive shaft (see figure 4.3). Automotive-type rubber disc couplings were used to allow for angular deflections, whereas for the axial coupling a design from an ultralight aircraft drive system with cylindrical rubber elements was adapted and modified.

The carbon-fibre drive shaft was designed, manufactured, and tested at Aurora's facility. Three layers of carbon prepreg were used, each 0.35 mm thick. The outer diameter of the shaft is 81.8 mm. A static torque test with a sample section was done with positive results as well as dynamic testing (see section 7.3). Static torque tests were done with all the drive line components including the bonding connections between the carbon-fibre shaft and its connection flanges.

Figure 4.3: POC engine bay with the transmission coupling. The coupling has three flexible members: 2 rubber discs for the angular deflections and one for the axial deflections, which allows the center-shaft to contract or to elongate. All three couplings serve as dampers in the three axes: axial, radial, and torsional.

Figure 4.4: Supporting bearings for the carbon-fibre drive shaft in the POC tail boom.

4.3. Propeller

The propeller used on the POC was a 3-bladed wooden propeller with ground-adjustable pitch and a diameter of 1.27 m (50 inches). The prop was manufactured by GSC Systems for use in ultralight aircraft. Ground testing was done together with the complete transmission (see section 7.5). In order to predict and evaluate the Perseus POC performance, the prop airfoil was measured and its performance computed with the XROTOR software (see section 5.1.3). Figure 4.5 shows the Perseus POC airfoils at various radial stations.

Figure 4.5: Airfoil sections of the Perseus POC prop.

5. AERODYNAMICS AND PERFORMANCE

5.1. Analysis and design methodology

The Perseus POC went from concept to flight without the use of wind tunnel testing. All important analysis and design decisions were made using the computational tools described in this section.

5.1.1 Rapid configuration and mission analysis

The primary analysis tool used for evaluating aircraft performance and for evaluating the impact of specific design changes is a proprietary Aurora program known as QWIKPERF. Written in Basic and executed on a Macintosh computer, QWIKPERF uses a specified aircraft geometry. Inputs to the program include: weights and dimensions of major components (the program assumes a Perseus geometry); airfoil profile drag as a function of Reynolds number, CL, and Mach number; standard or polar atmosphere; and engine fuel consumption as a function of throttle setting. The program analyses a trajectory in discrete time steps through climb, cruise at a specified altitude, and descent phases. The primary output of QWIKPERF is a detailed trajectory analysis plus drag polars at selected design conditions.

5.1.2 Airfoil design

All of the performance prediction in the design process of the Perseus airfoils was performed by the direct/inverse ISES code (Drela 1986); (Drela and Giles 1987b). It has been verified for a number of transonic and low-Reynolds-number airfoils (Drela and Giles 1987a); including extensive field testing under NASA sponsorship during the Daedalus project. ISES is a zonal method that employs the steady compressible Euler equations to describe the inviscid outer flow, and a two-equation lagged dissipation integral formulation to describe the boundary layers and wake. The effect of the viscous regions on the inviscid flow is modeled by the displacement surface concept. The overall discrete equation system is solved by a full Newton method. This solution technique performs stable calculation of flows with limited separation regions without elaborate viscous/inviscid iteration techniques.

The ISES code uses an e^9 (Lighthill 1945) type transition prediction method similar to that of Smith and Gamberoni (Mach 1977), which models the spatial growth rate of Tollmien-Schlichting waves which lead to transition. The growth rates are determined from solutions to the Orr-Sommerfeld equation correlated with the local shape parameter and momentum thickness. The first-order DE, which describes the downstream growth of the most amplified Tollmien-Schlichting disturbance, is discretized and included in the overall viscous-inviscid equation system. This approach appears to give reliable prediction of transition both in attached boundary layers and in separation bubbles (Drela and Giles 1987a).

5.1.3 Propeller design

Propeller analysis and design is performed using the XROTOR code developed by Drela. XROTOR is an interactive program for the design and analysis of ducted and free-tip propellers and windmills. It employs the classical lifting-line representation of the propeller blade, allowing arbitrary profile lift and drag properties to be specified. Prandtl-Glauert compressibility corrections are used for the profile lift characteristics.

XROTOR calculates the induced velocities by numerically solving for the exact perturbation potential flowfield about the helical vortex sheet wake. This formulation is valid for all advance ratios and blade numbers, unlike classical theories which assume a small advance ratio and many blades. XROTOR also makes a first-order correction for the change in the pitch of the trailing sheets due to propeller's self-induction. (These features are particularly important for the Perseus propeller at ceiling.)

XROTOR includes a non-linear beam blade structural model. Rapid stress calculations at selected operating points simplify the overall aero/structural design of the blade. Static aeroelastic deflections can be readily incorporated into the blade's aerodynamic twist distribution.

5.1.4 3-D Flow

Flow over three-dimensional configurations is analyzed using the QUADPAN code. QUADPAN is a low-order panel method for calculating the potential flow about general three-dimensional configurations. It employs the same constant-source and constant-doublet panel formulation used in other popular panel codes like VSAERO. The Prandtl-Glauert compressibility correction is used for subsonic compressible flows.

QUADPAN permits specification of an arbitrary freestream direction as well as steady angular rotation rates. All aerodynamic directional and rate derivatives can thus be obtained by perturbing a flow angle or rate and finite-differencing the calculated loads.

For quick-look and preliminary design studies, the LINAIR code developed by Desktop Aeronautics is used. LINAIR uses extended lifting line theory, and can be run on a desktop PC.

5.1.5 Simultaneous wing aerodynamic and structural load prediction

A calculation method known as ASWING was used for the simultaneous solution of aerodynamic and structural loads on arbitrary high-aspect-ratio wings. The wing aerodynamics are modeled using lifting line theory with roll rate, yaw, and yaw rate effects included. The wing structure is modeled as a nonlinear beam with vertical and horizontal displacement and torsional degrees of freedom. Axial compression effects are also incorporated to permit modeling of wings with external bracing struts or wires. The aerodynamic and structural problems together constitute a coupled nonlinear system for the aerodynamic and structural unknowns, which is discretized and solved using a global Newton method. The overall procedure permits computationally economical prediction of the following: 1) aerodynamic and structural loads for a very wide range of operating conditions, 2) induced drag with static wing twist effects included, 3) lateral and longitudinal wing static stability derivatives and roll-yaw coupling forces incorporating wing deflections, 4) static divergence and aileron reversal speeds, and 5) buckling loads for externally braced wings. The program was validated and used extensively on the *Daedalus* human powered aircraft (Drela 1990).

5.2. Airfoil performance

Perseus uses a custom series of airfoils designed for high performance at low Reynolds number. The root chord airfoil PS02 is closely related to that used on the *Daedalus* aircraft. Airfoil performance as a function of Reynolds Number is shown in figure 5.1.

Figure 5.1: Airfoil performance as a function of Reynolds Number.

5.3. Aircraft drag buildup

Table 5.1 summarizes the drag contribution of various components of the POC at its design cruise condition.

Table 5.1 Drag Breakdown of POC at cruise conditions.

Major element	Cd, referenced to wing
Wing profile drag	.007
Wing induced drag	.026
Horizontal stab	.0016
Vertical stabilizer	.0011
Fuselage	.0035
Landing gear	.0103
Tail boom	.0009
Cooling inlets	.0010
Radiators	.0010
Speed brakes	.0000
Other	.0005
Subtotal, parasite drag	.020
Interference	.002
Total drag coefficient	.055
Lift coefficient	1.2
Lift-to-drag ratio	22:1

5.4. Power predictions

Figure 5.2 shows predicted performance versus speed for the POC, with the drag brakes open and closed.

Figure 5.2: Power versus speed as predicted for the POC using QWIKPERF.

6. FLIGHT CONTROL SYSTEM

6.1. Airborne Flight Control System

A functional block diagram of the basic flight control system is shown in Figure 6.1. The heart of the system is a 3U VME-bus computer with a 68020 microprocessor and a 68881 co-processor. The central computer performs three main tasks: communication, autopilots and navigation, and propulsion system control. In the event of a failure in the primary computer, a secondary computer has all control functions except for autopilots and telemetry.

Ground-to-air communication is provided by a multi-channel UHF digital radio system. There are two 4800 baud data paths. The first path is provided through a Motorola DARCOM radio operating at 920 and 945 MHz. The back-up link was a Motorola RNET configured to operate on 426 MHz.

Airborne sensors continuously measure the aircraft state and this information is provided to the ground based pilot as well as the airborne autopilot and navigation algorithms. Roll and pitch angles are sensed by a vertical gyro while heading is derived from an encoded compass. The angular rates are sensed with three rate transducers. The angle of attack and sideslip are measured with vanes. Indicated airspeed, barometric altitude, and vertical speed are also measured by pressure transducers.

In all operational versions of Perseus, navigation will be performed using data obtained from the Global Positioning System (GPS), onboard heading sensors, and ground based tracking (when available). The airborne 5-channel GPS receiver provides 3-D position and velocity data to the airborne navigation algorithms. This system was installed in the POC but was not integrated into the flight computer.

Primary flight controls are activated by six closed-loop position servos. A single servo is provided for each aerodynamic surface, except on the elevator, which is split into two sections with each section actuated by a separate servo. This configuration allows the aircraft to survive the failure of any one primary flight servo.

Primary electrical power was provided by two redundant 28V battery packs located in the nose of the aircraft.

Figure 6.1: Block diagram of the Perseus airborne electronics.

Figure 6.2: Primary components of the Perseus flight control system include (clockwise from upper left): video transmitter, vertical gyro, flight computer, servo, GPS receiver, data transceiver.

6.2. Ground Control Station

Two separate ground stations were built for the Perseus POC. The primary ground station was housed in a small trailer (see figure 6.3) and had three control stations (one for the pilot and two flight engineer stations). The pilot has three large-screen monitors (see figure 6.4) to present map data, video data, and a "glass cockpit" instrument display. Pilot commands were input through either of two control boxes (one mounted inside and the other on a ten foot cable for outside use). The flight engineer stations were terminals which controlled the aircraft and ground station control configuration, activated such commands as emergency flight termination, and displayed telemetry.

The second ground station was a small portable unit that lacked the large displays and had only one flight engineer station, but otherwise had all the capabilities of the main station (see figure 6.5). During the POC test program, this station was placed outside the main ground station, powered and ready in case of a failure in the main station. It could also be installed in a vehicle (including a manned aircraft), at a forward site, or to control take-off and landing operations when the main site is located elsewhere.

Figure 6.3: Exterior view of the POC ground station.

Figure 6.4: Interior view of the POC ground station. Piloting station is at right.

Figure 6.5: POC mobile ground control station.

6.3. Software

The Perseus flight control system uses a proprietary software known as Ithaca[®]. Ithaca uses the OS-9 operating system and a layered design approach to achieve high reliability while also maintaining flexibility in the hardware configuration. The development, testing, and validation of Ithaca represents one of the major investments in the Perseus program.

The basic system architecture is shown in figure 6.6. Separate processes run continuously for each of the radio links, GPS, A/D processing, D/A processing, and propulsion system control. Data is shared through common modules with higher-order processes such as autopilot, autonavigation, and lost link routines. Numerous system monitoring functions are present, including a software status monitor that can independently restart any process that has failed, and a watchdog timer which can reset the entire computer in flight if needed. The system is designed for maximum flexibility when interfacing with payloads.

Figure 6.6: Overview of software architecture. The actual software implemented in "C" requires about 50,000 lines of code.

6.4. Flight Termination System

The primary purpose of the flight termination system was to insure that the aircraft cannot escape from its controlled trajectory in the event of a failure in the primary flight system. A secondary purpose is the safe return of the aircraft and payload. The POC used a BRS type 4-XP1 ballistic parachute, originally designed for recovery of manned ultralight aircraft. The pyrotechnic igniter could be triggered on ground command by the backup flight computer.

6.5. Aircraft Dynamics

The Perseus Proof of Concept (POC) control systems were designed using Optimal Control techniques employing full state feedback to aid in obtaining good robustness to model uncertainty as well as actuator/sensor failure. These designs are gain-scheduled on dynamic pressure to cover the full operating envelope, where the gain scheduling is accomplished by polynomial curve fitting as functions of dynamic pressure. This allows for continuous smooth gain changing, eliminating the unpleasantness associated with switching gains between discrete operating points. A complete description of the control system design and evaluation is included in Appendix A. The primary derivatives, calculated by two different techniques, are shown in Table 6.1.

Table 6.1: Stability derivatives for POC at $C_L=1.2$ and cruise conditions.

Symbol	Derivative	QUADPAN	LinAir
$C_{L\alpha}$	Lift curve slope	6.35	6.05
C_{Lq}	Lift due to pitch rate	7.8	6.53
$C_{m\alpha}$	Pitching moment due to angle of attack	-1.4	-1.05
C_{mq}	Pitch damping	-20.5	-21.3
$C_{y\beta}$	Sideforce due to sideslip	-0.287	-0.216
C_{yp}	Side force due to roll rate	-0.24	-0.065
C_{yr}	Side force due to yaw rate	0.1975	0.187
$C_{l\beta}$	Rolling moment due to sideslip	-0.377	-0.081
C_{lp}	Rolling moment due to roll rate	-1.62	-0.677
C_{lr}	Rolling moment due to yaw rate	0.55	0.28
$C_{n\beta}$	Yawing moment due to sideslip	0.04	0.023
C_{np}	Yawing moment due to roll rate	-0.222	-0.15
C_{nr}	Yawing moment due to yaw rate	-0.064	-0.03
$C_{L\delta e}$	Elevator power	0.32	0.317
$C_{m\delta e}$	Elevator power	-1.43	-1.22
$C_{y\delta r}$	Side force due to rudder	0.15	0.14
$C_{l\delta a}$	Rolling moment due to aileron deflection	0.35	0.22
$C_{l\delta r}$	Rolling moment due to rudder	0.013	0.004
$C_{n\delta a}$	Yawing moment due to aileron	-0.019	-0.005
$C_{n\delta r}$	Yawing moment due to rudder deflection	-0.07	-0.03

7. INTEGRATED TEST PROGRAM

Prior to being certified for flight, the POC underwent an extensive program of development and validation testing. The most important tests are summarized in table 7.1, and are discussed below.

Table 7.1: Summary of major pre-flight tests.

Test	Date	Level
Wing proof load	November, 1990	3 g's @ 354 kg GTOW
Tailboom & fuselage frame	January 18, 1991	4.7 g's
Drive shaft critical rpm	April 16, 1991	1500-2800 rpm
Drop test	April 11, 1991	0.5 - 2.5 m/s
Full drivetrain test	April 25, 1991	1000-3000 rpm
Avionics design review	April 29, 1991	
Thermal testing	June 1-10, 1991	55°C
Vibration testing	June 8-20, 1991	engine as source
EMI testing	July 5-16, 1991	
Control link validation (ground)	August, 1991	1 nm, with attenuation
Control link validation (air)	August, 1991	to 17 nm
Autopilot ground tests	August, 1991	
Taxi testing (low speed)	September 9 -October 29, 1991	to 20 kts
Flight readiness review	October 14 & 28, 1991	
Taxi testing (high speed)	November 5-7, 1991	to 45 kts
First flight	November 8, 1991	

7.1. Wing proof test

To validate the wing construction, the entire wing was statically loaded with a mass of 1100 kg. This is equivalent to a load factor of about 3 g's for a non-lifting mass of 350 kg. Deflections measured during this tested closely matched those predicted.

7.2. Tailboom & fuselage frame static load test

The attach joint between the steel fuselage truss and the graphite-epoxy tailboom was statically loaded to the equivalent of 4.7 g's by bolting the fuselage frame to a solid base and applying loads with sandbags to the tail surfaces and tail boom. This test also tested rudder and horizontal stabilizer loads at maneuver speed.

7.3. Drive shaft critical rpm test

To obtain temperature data from the driveshaft bearings and to verify that no critical modes were excited, the assembled drive shaft was driven with an electric motor throughout its speed range. Bearings were instrumented and their temperatures recorded. The first test showed excessive vibration which was attributed to a slight curvature in the drive shaft. A new shaft was manufactured and no further vibrations were encountered. The bearing temperatures remained well within acceptable levels.

7.4. Drop test

To verify the ability of the landing gear and other structure to sustain shock loads associated with landing, the entire fuselage (with appropriate weights added for fuel and other systems) was dropped in a series of tests from heights between 10 and 30 cm, providing impact velocities of 0.5 to 2.5 meters per second. Accelerometers located throughout the vehicle provided data on loads experienced by key components, such as avionics. The front landing gear was modified as a result of lessons learned from these tests.

7.5. Full drivetrain test

In April 1991, the entire POC was transported to a local airfield and the propulsion system run through four hours of run time. These tests broke in the engine, fuel system, gearbox, drive train, and prop. These tests went smoothly and validated the transmission design.

7.6. Avionics design review

Once the avionics architecture had been developed, an outside panel of experts was convened to provide a peer review. This two-day workshop resulted in a prioritized list of action items and suggestions, many of which were incorporated in the design.

7.7. Thermal testing

To test the cooling system, the fully integrated fuselage was placed inside Aurora's large composites curing oven. The effectiveness of the cooling system was tracked as the ambient temperature was increased to 55 degrees Celsius. The test was performed in two blocks of six hours each.

7.8. Vibration testing

A series of tests were conducted with the avionics package installed in the aircraft. The engine was run through a full range of speeds, and the resulting levels of vibration observed. Initial vibration levels in the avionics bay were higher than desired, and a series of modifications were developed to the shock mounts of both the avionics package and the engine. These modifications brought vibrations to within acceptable levels.

7.9. Electromagnetic interference (EMI) tests

A systematic survey of electromagnetic sources and sensitivities on the aircraft was conducted using a signal generator and a recording oscilloscope and a spectrum analyser. A number of filters were added throughout the system as a result of these tests.

7.10. Control link validation test (ground)

The aircraft and ground station were positioned approximately a mile apart on the Manassas airfield. Attenuators of increasing strengths were inserted between the antenna and the receiver to simulate long ranges.

7.11. Control link validation test (air)

The completed avionics package was installed in a light aircraft and flown on a simulated trajectory, with signal strength monitored both inside the ground station and onboard the aircraft. This test validated the link margin calculations and verified that the system was free from local radio interference. This test was repeated shortly before flight at El Mirage, the actual flight location.

7.12. Autopilot ground test

A series of tests were conducted on the ground to verify proper operation of the autopilot system. In the first sequence of tests, synthetic sensor signals were fed to the flight computer, and the response of the control surfaces was observed. Further tests were conducted by suspending the aircraft and providing various motion inputs to the sensor (a known roll input, for example) and observing the response of the control systems.

7.13. Taxi tests

An extensive series of taxi tests were conducted at Manassas. These tests were designed (1) to validate proper functioning of all aircraft control systems; (2) to determine the aircraft's ground handling, including maneuverability, acceleration, and braking performance; (3) to provide endurance testing for system components in the aircraft's vibration environment; and (4) to provide training for the pilot and flight control crews. Tests up to 20 kts were conducted at Manassas. Tests up to 45 kts (greater than flight speed) were conducted at El Mirage, with wing fences installed to prevent accidental take-off.

7.14. Flight Readiness Review

Prior to shipment to California, an internal Flight Readiness Review was held to insure that all test objectives had been met and all preparations for flight were completed.

8. FLIGHT TEST PROGRAM

8.1. Objectives

The flight test program for the POC had the following objectives:

- (1) Demonstrate the concept of the unmanned scientific research aircraft by successfully completing a series of test flights that were documented by still photographs and videotape suitable for distribution to the scientific community.
- (2) Validate the structural and aerodynamic design of the Perseus airframe.
- (3) Validate the electronic flight control system.
- (4) Provide pilot training experience on the dry lakebed environment so that future operations could be conducted with confidence from runways.
- (5) Deploy a demonstration science payload consisting of prototype dropwindsondes.
- (6) Maintain the highest levels of safety for personnel and property.

8.2. Flight configuration

The flight configuration is shown in Figure v. The weight statement is shown in Table 3.1.

An on-board data collection system was installed which would store data from dedicated sensors installed for engineering test purposes. These included strain gauges mounted at the wing roots and tailboom root and a set of control position transducers which would read the control surface deflection directly at the surface. The data was collected in a Tattletale Mark 3 computer installed in the payload bay.

The test program was conducted from El Mirage dry lake, a public lakebed approximately 20 miles east of Lancaster, CA. The lakebed is 2,833 feet above sea level. The aircraft was based in a small hangar at the edge of the lakebed and was towed by truck to the center of the lakebed each morning for flight operations..

8.3. Chronology

The first flight of the Perseus POC was made on Friday November 8, 1991. Take-off time was 08:35 am with a GTOW of 429.5 kg (see table 8.1). The duration of the flight was 21 minutes and was performed at altitudes between 1,500 and 2,000 ft. AGL. The objective of the flight was to test the autopilot routines as well as a general validation of the aircraft's structure and its avionics systems. The airspeed regime explored during this flight was between 40 and 46 knots. During this flight all aircraft systems behaved as expected and most of the flying was done with both the lateral and longitudinal autopilot controls. The aircraft was flown entirely from the outside pilot station. The pilots experienced considerable difficulty when attempting to fly the aircraft without the autopilots, particularly in controlling the pitch axis. The aircraft was landed with autopilots on, using throttle to control final descent rate and to flare for touchdown.

The second flight was made on Tuesday, November 12, 1991, when Perseus flew for 51 minutes from 07:10 am to 08:01 am with a GTOW of 429 kg. The primary objectives of this flight were the validation of R/C flying capability by the outside pilot and the first inside flying from the control panel of the ground station via video and instrument panel. The aircraft was much easier to fly without its autopilots in this mode. The Tattletale auxiliary data collection system operated for the first time.

The third flight, made Friday, November 15, 1991, was the most comprehensive of the test program. An extended (>3 minute) test was conducted in autonomous flight mode. Five boilerplate dropwindsondes were deployed during the flight (see figure 8.2). Total flight times was one hour and 25 minutes.

Table 8.1: A summary of flight test highlights.

Flight Test Highlights	
Fall, 1991 deployment	
El Mirage, California	
First equipment leaves Manassas:	Thursday, October 31
First taxi test:	Tuesday, November 5
First flight:	Friday, November 8
Last day of operations:	Friday, November 15
Last equipment returned to Manassas:	Wednesday, November 20
Total days in transit:	10
Total days in ready condition:	11
Days of taxi test:	3
Days of flight ops:	3
Days lost to lakebed availability:	3
Days lost to weather:	1
Days lost to personnel:	1
Crew:	
1. Peter Bahniuk	Back-up ground station
2. Clint Church	Photo, assistant crew chief, driver 1
3. Tom Clancy	Avionics controller
4. Brian Duff	PAO
5. Martin Gomez	Avionics flight engineer, driver 2b
6. Tim Just	#2 pilot
7. John Langford	RSO
8. Steve Waide	Crew chief, driver
9. Chuck Walcher	Avionics checkout inside pilot
10. Greg Zack	prime pilot, driver
11. Siggy Zerweckh	Flight Director, project manager, driver 2a
Minimum crew:	
1. Pilot #1	
2. Pilot #2	
3. Crew chief	
4. Ground station operation (avionics)	
5. Back-up ground station operator (avionics)	
6. Logistics coordinator/project manager	
PLUS any extras required for a) range operation or b) special engineering test or development.	
Flight #1	
Friday, November 8, 1991	
21 minutes	
429.5 kg	
• Outside piloting	
• First takeoff, first landing	
• first use of autopilots; highly successful	
• Speed brakes tested - effective	
• stall behavior & recovery tested - excellent	
• aircraft difficult to fly in pure RC mode	
Flight #2	
Tuesday, November 12, 1991	
51 minutes	
• verified controllability without autopilots	
• inside flying using video & instruments	
• first dropwindsonde deployment	
• short test of lost link	

- first use of Tattletale data acquisition system

Flight #3

Friday, November 15, 1991

85 minutes

- extensive inside flying, with and without video
- extended (3 minute) test of lost link mode (autonomous flight)
- air-to-air still and video photography
- deployment of 3 dropsondes
- speed brakes jammed in open position
- data collection with Tattletale
- first attempt at using GPS
- autopilot validation tests
- quantitative performance measurements

Figure 8.1: The Perseus POC lifts off on its first flight, November 8, 1991.

Figure 8.2: Dropwindsonde being dispensed from the POC during its third flight - November 15, 1991.

8.4. Analysis of flight data

Although the flight tests were abbreviated, several important quantitative results were obtained and compared with design predictions. These included:

- the maximum lift coefficient
- lift-to-drag ratios, and hence, required power
- autopilot performance

8.4.1 Maximum lift coefficient

In the flight tests, the relevant measurement needed to deduce $C_{L_{MAX}}$ is the minimum sustainable indicated airspeed (V_{stall}). Using the known wing loading W/S , we obtain

$$C_{L_{max}} = \frac{2W}{\rho_{ref} V_{stall}^2 S}$$

where ρ_{ref} is the reference air density at which the indicated airspeed sensor is calibrated (assumed to be sea level). Minimum indicated airspeeds observed on the POC were about 32 knots, which for $W/S = 260 \text{ N/m}^2$ corresponds to

$$C_{L_{MAX}} \cong 1.55$$

using $\rho = 1.225 \text{ kg/m}^3$. This is very close to the predicted maximum lift coefficient of the PS 02 airfoil, and indicates that extensive laminar flow is being obtained even at sea level conditions.

8.4.2 Lift-to-drag ratio

The lift-to-drag ratio (L/D) of an aircraft in steady flight can be determined by starting with the force balance along the flight direction :

$$T = D + \frac{\dot{h}}{V} W \quad (8.1)$$

where D is the drag and V is the true airspeed, related to the indicated airspeed by:

$$V = V_i \sqrt{\frac{\rho_{ref}}{\rho}} \quad (8.2)$$

The ambient density ρ at the measured flight altitude was obtained from a Standard Atmosphere chart. From the force balance equation we obtain:

$$\begin{aligned} \frac{1}{L/D} &\approx \frac{\cos \gamma}{L/D} \\ &\approx \frac{T}{W} - \frac{\dot{h}}{V} \end{aligned} \quad (8.3)$$

where $\gamma = \sin^{-1}\left(\frac{\dot{h}}{V}\right)$ is the flight path angle. In the POC flight tests, gamma is sufficiently small so that the approximation $\cos \gamma \approx 1$ can be safely made. Knowing propeller pitch and speed, T can be determined from propeller performance curves. If W is measured before flight and updated for fuel burn, and $\dot{h}$ and V are measured in flight, we can calculate L/D in flight.

Table 8.2 lists the measured flight data of the POC with the deduced propeller thrust and L/D . Based on the low-thrust points with small T/W , the L/D appears to lie between 25-30 with the airbrakes

closed, and between 15-20 with the airbrakes open. As indicated in figure 8.2, these values are considerably better than the values predicted using QWIKPERF.

The primary difficulty in using equation (8.3) is that the righthand side involves taking the difference of two large quantities to obtain the small quantity $1/(L/D)$. The uncertainty in the measurements and the predicted thrust thus leads to very large uncertainties in the L/D , particularly in a climb where both quantities on the righthand side of the equation are large. The uncertainties are the smallest with the engine nearly at idle, in which case the propeller thrust estimate has only a minor impact on the calculated L/D . Even in these cases, however, there is the substantial uncertainty of vertical atmospheric motion which gives a false rate-of-climb reading $\dot{h}$. This problem is all too common when free-flight L/D measurements of high-performance aircraft, notably sailplanes, are attempted.

As an example of the magnitude of errors that may occur, the last column of table 8.2 ($L/D+1^\circ$) column gives the calculated L/D if the measured propeller blade angles are increased by 1° or the zero-lift angles are decreased by 1° . Although a measurement error of this magnitude is believed to be unlikely, it indicates which L/D values are potentially most affected by errors in the prop thrust prediction due to geometry measurement errors. Note that the effect of the blade angle change is the greatest for the high-thrust cases.

Table 8.2: Summary of selected Perseus POC flight test data and calculated L/D .

Time h:mm	Airbrakes	V m/s	Altitude ft	rpm	T/W	h/V	L/D	$L/D+1^\circ$
2:04	no	24.1	3790	3100	0.163	0.132	54	27
2:08	no	23.0	4530	2000	0.021	-.009	33	25
2:15	no	23.2	3190	2225	0.041	0.011	34	25
2:20	no	23.6	4700	2950	0.119	0.100	52	27
2:11	yes	23.0	4240	1575	-.006	-.052	21	20
2:12	yes	23.5	4070	1525	-.010	-.067	18	16
2:23	yes	24.1	4260	1775	0.001	0.059	17	15
2:28	yes	29.8	4750	2750	0.054	0.000	18	14

Figure 8.3: Comparison of predicted and measured performance

8.4.3 Autopilots

Comparison of the lost link flight test data with simulation data demonstrates good correlation, illustrating that the lateral control system performance was as expected. This data combined with video footage of the alpha and beta vanes on the nose boom demonstrates good regulation performance, with turns well coordinated (minimal sideslip) and angle of attack perturbations minimal.

Figures 8.4 and 8.5 illustrate the flight test data for two maneuvers. Figure 8.4 shows the lost link maneuver with simulation data overplotted for comparison. Figure 8.5 is flight test data for the transition in climbout after takeoff from open loop (radio control) to airspeed hold autopilot active. Since the reference inputs to the airspeed hold autopilot are unknown (not recorded at time of flight test) it is not possible to generate simulation performance data for comparison in this case. The well damped response of the aircraft in longitudinal control, however, illustrates the good performance of the airspeed hold autopilot. A further demonstration of the pilot's satisfaction and confidence in the autopilots' (both lateral and longitudinal) performance is the fact that all landings were done with both autopilots active.

Figure 8.4: Lost link maneuver. Shown is flight test and simulation data for comparison, in the lost link maneuver, where the aircraft is commanded into a steady turn in the case of a lost link with the ground station. The time histories show good correlation with the simulation data, indicating satisfactory performance of the lateral autopilot. Note good sideslip regulation and the clearly visible effect of the washout filter in the yaw rate time history.

Figure 8.5: Transition to airspeed hold autopilot active in climbout from takeoff. The transition involves a relatively sharp pitch down maneuver due to two possibilities. The pitch angle is much larger (positive, nose up) than the autopilot expects for straight and level flight and this combined with the low airspeed (35 knots) at which the airspeed hold function appears to have been activated (neutral stick commands ~41 knots in this mode) cause the autopilot to drop the nose of the aircraft rather sharply. In future versions, this transition may easily be made much smoother by low pass filtering the effective "step" inputs to the autopilot when activated with the actual flight condition at variance with the expected (by the autopilot) conditions.

9. SUMMARY AND CONCLUSIONS

The Perseus proof-of-concept program was an unusual experiment to determine whether or not it is practical to design from scratch a modern unmanned aircraft specifically tailored to the needs of the atmospheric science community. Going from an original concept to a flight-tested aircraft in about 30 months, the Perseus proof-of-concept program achieved all of its original goals:

(1) *The POC developed and validated the aerodynamic, structural, and flight control technologies upon which future unmanned science aircraft can be built.* Although it was not possible within the funding constraints of the POC program to obtain a large statistical data base applicable to reliability, the POC program showed conclusively that the basic concept behind the unmanned science aircraft is sound.

(2) *The POC program proved that scientifically useful unmanned aircraft could be built at costs affordable to the science community.* The total cost of the POC program, about \$1.5 million, was financed from within the science community itself. When compared to typical aerospace industry practice, the POC was completed at between 1% and 10% of typical practice. The keys to controlling these costs were: (1) careful attention to cost throughout the design process, (2) use of a small, highly motivated design and construction team, (3) the use of commercial and industrial-grade components wherever possible.

(3) *The POC successfully built the infrastructure, experience, and credibility of the Aurora Flight Sciences team.* During the course of the POC, Aurora went from a one-person office to a fully functional small company with 20 employees and 13,000 ft² design and manufacturing facility. The lessons learned in the program allowed Aurora to win the backing from NASA for the high-altitude version of Perseus and to move rapidly into the full-scale development of that program.

The POC helped establish the viability of the entire concept of the unmanned scientific aircraft. At the time the POC program began, no serious consideration had ever been given to the use of a flexible, low-cost unmanned aircraft as a tool for atmospheric science. By the time the program was completed, the interagency Committee on Earth and Environmental Sciences (CEES 1991; CEES 1992), the Congressional Office of Technology Assessment (Williamson 1993), and the Department of Energy's JASON Advisory Panel (Banks and others 1991) had each studied and endorsed the concept; the White House had twice proposed line items in the Federal budget devoted specifically to unmanned science aircraft; and the National Academy of Sciences had launched a formal study. At least three Federal agencies were known to be contemplating large-scale programs. More than a dozen scientists had submitted proposals or begun construction of a new generation of compact lightweight instruments for use aboard unmanned aircraft, and at least one such instrument had been tested in the field. Whether or not the Perseus program achieves its future goals, it seems likely that the POC program will have permanently influenced the course of atmospheric science and global change research.

10. Appendix A: Control System Design and Evaluation (prepared by D. Vos).

CONTROL SYSTEM DESIGN AND EVALUATION

INTRODUCTION

The Perseus Proof of Concept (POC) control systems are designed using Optimal Control techniques employing full state feedback to aid in obtaining good robustness to model uncertainty as well as actuator/sensor failure. These designs are gain scheduled on dynamic pressure to cover the full operating envelope, where the gain scheduling is accomplished by polynomial curve fitting as functions of dynamic pressure. This allows for continuous smooth gain changing, eliminating the unpleasanties associated with switching gains between discrete operating points.

The control systems are extensively tested by simulation of various flight conditions in a full nonlinear three degree of freedom, six axis rigid body simulation. The simulation includes models for the power plant; all servos, with bandwidth, slew rate and deadband limits imposed; washout filters on all rate sensor signals; an atmospheric model; a model of the airspeed sensor since this has a relatively slow time constant (300 ms); as well as noise corrupting the rate gyro signals. The aerodynamic model is a quasi linear model in the sense that perturbation aerodynamics are used to determine the present time step aerodynamic loads, given the previous time step loads and the perturbations in aircraft motion from the previous time integration step to the present. This may be viewed as using a locally linear model to determine the perturbation airloads over a single integration time step. Note that this does not at all limit the aerodynamic model from being fully nonlinear over the complete aerodynamic flight envelope.

THE AERODYNAMIC MODEL

The aerodynamics are modeled by Professor Marc Drela of MIT, using a potential flow model and numerical solution (QUADPAN). The perturbation model is determined by solving the potential flow problem for specific control or disturbance conditions and linearizing to determine effective perturbation aerodynamic coefficients. Figure 1 shows the full aerodynamic model used, where we are only concerned about operation over a limited range of sideslip and angle of attack, hence the relatively linear dependence on angle of attack.

Figure 1. Aerodynamic model determined from QUADPAN solution to potential flow problem over the specified sideslip and angle of attack ranges.

Figure 2. Stability axis system showing Euler angles and angles of attack (α) and sideslip (β). The velocities and angular velocities $u, p; v, q; w, r$ are aligned with the axes X_s, Y_b and Z_b , respectively. The stability axis system aligns the X_s -axis such that the steady state body Z -axis velocity is zero, i.e. $W=0$.

The aerodynamic stability derivatives of Figure 1 are as defined in [McRuer] for the stability axis system. The first order polynomial representations of the data of Figure 1 with correction to units of radians are determined in a least squares fit as

$$\begin{aligned} CL &= 1.1372 + 0.1116 \alpha \\ CL_u &= 0 \\ CL_q &= (0.1321 - 0.00204 \alpha) 180/\pi \\ CL_{de} &= 0.321 \\ CL_{dt} &= 0 \\ CL_\alpha &= (0.11082) 180/\pi \end{aligned}$$

$$\begin{aligned} Cd &= 0.019374 + 0.01578 CL^2 \\ Cd_u &= 0 \\ Cd_\alpha &= 0.0316 CL CL_\alpha \\ Cd_q &= 0 \\ Cd_{de} &= 0 \\ Cd_{dt} &= 0 \end{aligned}$$

$$C_{t_u} = 0$$

$$C_{y_\beta} = -(0.0053 + 0.000125 \alpha) 180/\pi$$

$$C_{y_p} = (-0.004 + 0.0001 \alpha) 180/\pi$$

$$C_{y_r} = (0.0039 + 0.0002 \alpha) 180/\pi$$

$$C_{y_{da}} = 0$$

$$C_{y_{dr}} = 0.1375 \text{ Sratio}$$

$$C_{l_\beta} = -0.1125 (0.0608 + 0.001 \alpha) 180/\pi$$

$$C_{l_r} = 0.1125 (0.1054 + 0.009 \alpha) 180/\pi$$

$$C_{l_p} = 0.1125 (-0.2499 + 0.00035 \alpha) 180/\pi$$

$$C_{l_{da}} = 0.3495$$

$$C_{l_{dr}} = 0.012 \text{ Sratio}$$

$$C_m = 0$$

$$C_{m_u} = 0$$

$$C_{m_\alpha} = (-0.0283 - 0.0017 \alpha) 180/\pi$$

$$C_{m_q} = (-0.3609 - 0.0011 \alpha) 180/\pi$$

$$C_{m_{de}} = -1.4324$$

$$C_{m_{dt}} = 0$$

$$C_{n_\beta} = -0.1125 (-0.007 - 0.0001 \alpha) 180/\pi$$

$$C_{n_r} = 0.1125 (-0.0106 - 0.0003 \alpha) 180/\pi$$

$$C_{n_p} = 0.1125 (-0.04024 - 0.0025 \alpha) 180/\pi$$

$$C_{n_{da}} = -0.0189$$

$$C_{n_{dr}} = -0.063 \text{ Sratio}$$

where Sratio=1.117 , accounting for 30% increase in rudder surface area. The dimensional derivatives used in the state space linear model for controller design [McRuer] in stability axes with upper case representing nominal values and perturbation quantities in lower case, are given below.

Aircraft mass and inertia properties:

$$\text{Mass} \quad M = 430 \text{ kg}$$

Inertias in body axes

$$\begin{aligned}
 I_{xx} &= 1925 \text{ kg.m}^2 \\
 I_{yy} &= 998 \text{ kg.m}^2 \\
 I_{zz} &= 2788 \text{ kg.m}^2 \\
 I_{xy} &= -9.8 \text{ kg.m}^2 \\
 I_{yz} &= 3.7 \text{ kg.m}^2 \\
 I_{zx} &= -55 \text{ kg.m}^2
 \end{aligned}$$

The equivalent inertias transformed into stability axes are

$$\begin{aligned}
 I_{xs} &= I_{xx} \cos^2(\alpha) + I_{zz} \sin^2(\alpha) - 2 I_{xz} \sin(\alpha) \cos(\alpha) \\
 I_{ys} &= I_{yy} \\
 I_{zs} &= I_{xx} \sin^2(\alpha) + I_{zz} \cos^2(\alpha) + 2 I_{xz} \sin(\alpha) \cos(\alpha) \\
 I_{xzs} &= (I_{xx} - I_{zz}) \sin(\alpha) \cos(\alpha) + I_{xz} (\cos^2(\alpha) - \sin^2(\alpha)) \\
 I_{xys} &= I_{xy} \cos(\alpha) + I_{yz} \sin(\alpha) \\
 I_{yzs} &= I_{yz} \cos(\alpha) - I_{xy} \sin(\alpha)
 \end{aligned}$$

Define, with S the wing reference area, b the wingspan and ρ the air density,

$$\begin{aligned}
 q_q &= \rho S U \\
 q_a &= \rho S U/M \\
 q_b &= \rho S b/4/M
 \end{aligned}$$

then

$$\begin{aligned}
 X_u &= q_a (-C_d - C_{d_u}) & [1/s] \\
 X_{t_u} &= q_a (C_{d_u}) & [1/s] \\
 X_w &= q_a (C_L - C_{d_\alpha})/2 & [1/s] \\
 X_q &= q_a c (-C_{d_q})/4 & [m/rad.s] \\
 X_{d_e} &= q_a U (-C_{d_{d_e}})/2 & [m/rad.s^2] \\
 X_{d_t} &= q_a U (-C_{d_{d_t}})/2 & [m/kg.s^2] \\
 Y_v &= q_a (C_{y_\beta})/2 & [1/s] \\
 Y_p &= q_b U (C_{y_p}) & [m/rad.s] \\
 Y_r &= q_b U (C_{y_r}) & [m/rad.s] \\
 Y_{d_a} &= q_a U (C_{y_{d_a}})/2 & [m/rad.s^2]
 \end{aligned}$$

$$Y_{dr} = q_a U (C_{y_{dr}})/2 \quad [m/rad.s^2]$$

$$Z_u = q_a (-CL - CL_u) \quad [1/s]$$

$$Z_w = q_a/2 (-CL_\alpha - C_d) \quad [1/s]$$

$$Z_q = q_a c (-CL_q)/4 \quad [m/rad.s]$$

$$Z_{de} = q_a U (-CL_{de})/2 \quad [m/rad.s^2]$$

$$Z_{dt} = q_a U (-CL_{dt})/2 \quad [m/kg.s^2]$$

$$L_v = qq \, b (Cl_\beta)/2I_{xs} \quad [1/m.s]$$

$$L_r = qq \, b \, b (Cl_r)/4I_{xs} \quad [1/s]$$

$$L_p = qq \, b \, b (Cl_p)/4I_{xs} \quad [1/s]$$

$$L_{da} = qq \, U \, b (Cl_{da})/2I_{xs} \quad [1/rad.s^2]$$

$$L_{dr} = qq \, U \, b (Cl_{dr})/2I_{xs} \quad [1/rad.s^2]$$

$$M_u = qq \, c (C_m + C_{m_u})/I_{ys} \quad [1/m.s]$$

$$M_w = qq \, c (C_{m_\alpha})/2I_{ys} \quad [1/m.s]$$

$$M_q = qq \, c \, c (C_{m_q})/4I_{ys} \quad [1/s]$$

$$M_{de} = qq \, U \, c (C_{m_{de}})/2I_{ys} \quad [1/rad.s^2]$$

$$M_{dt} = qq \, U \, c (C_{m_{dt}})/2I_{ys} \quad [1/kg.s^2]$$

$$N_v = qq \, b (C_n\beta)/2I_{zs} \quad [1/m.s]$$

$$N_r = qq \, b \, b (C_n_r)/4I_{zs} \quad [1/s]$$

$$N_p = qq \, b \, b (C_n_p)/4I_{zs} \quad [1/s]$$

$$N_{da} = qq \, U \, b (C_{n_{da}})/2I_{zs} \quad [1/rad.s^2]$$

$$N_{dr} = qq \, U \, b (C_{n_{dr}})/2I_{zs} \quad [1/rad.s^2]$$

The "primed" derivatives, eliminating explicit appearance of cross products of inertia terms from the linearized equations of motion, are defined as

$$L_{pv} = (L_v + I_{zxs}/I_{xs} \, N_v)/(1 - I_{zxs}^2/I_{xs}/I_{zs})$$

$$L_{pp} = (L_p + I_{zxs}/I_{xs} \, N_p)/(1 - I_{zxs}^2/I_{xs}/I_{zs})$$

$$L_{pr} = (L_r + I_{zxs}/I_{xs} \, N_r)/(1 - I_{zxs}^2/I_{xs}/I_{zs})$$

$$L_{pda} = (L_{da} + I_{zxs}/I_{xs} \, N_{da})/(1 - I_{zxs}^2/I_{xs}/I_{zs})$$

$$L_{pdr} = (L_{dr} + I_{zxs}/I_{xs} \, N_{dr})/(1 - I_{zxs}^2/I_{xs}/I_{zs})$$

$$N_{pv} = (N_v + I_{zxs}/I_{zs} \, L_v)/(1 - I_{zxs}^2/I_{xs}/I_{zs})$$

$$N_{pp} = (N_p + I_{zxs}/I_{zs} \, L_p)/(1 - I_{zxs}^2/I_{xs}/I_{zs})$$

$$\begin{aligned}
N_{pr} &= (N_r + I_{zxs}/I_{zs} L_r)/(1 - I_{zxs}^2/I_{xs}/I_{zs}) \\
N_{pda} &= (N_{da} + I_{zxs}/I_{zs} L_{da})/(1 - I_{zxs}^2/I_{xs}/I_{zs}) \\
N_{pdr} &= (N_{dr} + I_{zxs}/I_{zs} L_{dr})/(1 - I_{zxs}^2/I_{xs}/I_{zs})
\end{aligned}$$

LONGITUDINAL LINEAR CONTROL DESIGN MODEL

The longitudinal linearized state space model for control design purposes may now be written as below, with the state vector $\underline{x}_{lng}'(t)=[u \ w \ q \ \theta]$ and elevator (de) as actuator with

$$\begin{aligned}
u &= \text{perturbation stability X-axis velocity} \\
w &= \text{perturbation Z-axis velocity (proportional to perturbation in angle of attack)} \\
q &= \text{perturbation pitch rate} \\
\theta &= \text{perturbation pitch angle}
\end{aligned}$$

$$A_{ln} = \begin{bmatrix} X_u & X_w & X_q & -g \\ Z_u & Z_w & (Z_q+U) & 0 \\ M_u & M_w & M_q & 0 \\ 0 & 0 & 1 & 0 \end{bmatrix}$$

$$B_{ln} = \begin{bmatrix} X_{de} \\ Z_{de} \\ M_{de} \\ 0 \end{bmatrix}$$

LATERAL LINEAR CONTROL DESIGN MODEL

As in the case of the longitudinal dynamics, the lateral linearized equations of motion are easily written in state space form as below with state vector $\underline{x}_{lat}'(t)=[v \ p \ r \ \phi]$ and rudder and ailerons $[dr \ da]'$ as actuators with

$$\begin{aligned}
v &= \text{perturbation sideslip velocity (proportional to } \beta) \\
p &= \text{perturbation roll rate} \\
r &= \text{perturbation yaw rate} \\
\phi &= \text{perturbation bank angle}
\end{aligned}$$

$$A_{lt} = \begin{bmatrix} Y_v & Y_p & (Y_r - U) & g \\ L_{pv} & L_{pp} & L_{pr} & 0 \\ N_{pv} & N_{pp} & N_{pr} & 0 \\ 0 & 1 & 0 & 0 \end{bmatrix}$$

$$B_{lt} = \begin{bmatrix} Y_{da} & Y_{dr} \\ L_{pda} & L_{pdr} \\ N_{pda} & N_{pdr} \\ 0 & 0 \end{bmatrix}$$

These state space models are in the standard form $\frac{d}{dt}(x(t)) = A x(t) + B u(t)$ with control $u_{ln}=d(\text{elevator})$ and $u_{lt}'=[d(\text{aileron}) \ d(\text{rudder})]$ for the longitudinal and lateral systems, respectively.

NOMINAL OPEN LOOP DYNAMICS

The open loop system eigenstructure for the flight condition of steady straight and level flight at 20 m/s airspeed at sea level is given below.

Longitudinal

Short Period mode

$$\begin{aligned} \lambda_1 &= -2.5217 + j \ 2.4022 \text{ rad/s} \\ \lambda_2 &= -2.5217 - j \ 2.4022 \text{ rad/s} \\ \xi_{sp} &= 0.724 \end{aligned}$$

Phugoid mode

$$\begin{aligned} \lambda_3 &= 0.0065 + j \ 0.4568 \text{ rad/s} \\ \lambda_4 &= 0.0065 - j \ 0.4568 \text{ rad/s} \\ \xi_{ph} &= -0.01422 \text{ (unstable)} \end{aligned}$$

Lateral

Dutch roll mode

$$\lambda_5 = -0.9403 + j 1.5355 \text{ rad/s}$$

$$\lambda_6 = -0.9403 - j 1.5355 \text{ rad/s}$$

$$\xi_{dr} = 0.522$$

Roll damping mode

$$\lambda_7 = -26.9188 \text{ rad/s}$$

Spiral mode

$$\lambda_8 = -0.0006 \text{ rad/s}$$

These open loop dynamics exhibit the expected benign characteristics of the aircraft. Only the phugoid mode is unstable and very poorly damped, although not an unexpected characteristic of a high L/D aircraft. The frequency of this mode is low enough that this should pose no problem for open loop control by a suitably experienced radio control pilot.

CONTROL DESIGN

The controllers are designed to improve damping of the open loop modes of the system as well as to provide integral servo control on forward speed (U) and bank angle (ϕ) allowing the pilot to operate the system in either open loop radio controlled mode; a single hold function active (bank angle or airspeed hold); or both hold functions active. When either of the hold functions is achieved, automatic stability augmentation of all the other modes for the specific axis is also active. For example, with airspeed hold active, the phugoid and short period modes are augmented to have good (better than critical) damping.

The Linear Quadratic Regulator (LQR) [see e.g. Bryson, Gelb, Kwakernaak and Sivan] optimal control design technique is used to achieve the desired specifications, with integrators appended to the forward speed, bank angle and sideslip states for integral control of these errors. Integral sideslip control is employed to ensure good regulation of this state to minimize the effect of the

strong roll-yaw coupling (mainly through Cl_β) which causes large rolling moments due to relatively small sideslip angles.

Figure 3 illustrates the longitudinal and lateral control systems in block diagram form.

Figure 3. Block diagram illustration of lateral and longitudinal LQR based control systems. Servo functions are on bank angle (lateral) and airspeed (longitudinal), with all other states regulated. Note that the structure easily allows servo action on sideslip (lateral) as well, for cross-wind landing capability.

The state and control penalty weighting matrices are selected in a manner akin to Bryson's rule [Bryson].

Longitudinal

State weights

$$Q_{ln} = \begin{bmatrix} 1 & 0 & 0 & 0 & 0 \\ 0 & 0.05 & 0 & 0 & 0 \\ 0 & 0 & 10000 & 0 & 0 \\ 0 & 0 & 0 & 100000 & 0 \\ 0 & 0 & 0 & 0 & 10 \end{bmatrix}$$

Control weight

$$R_{ln} = 50000$$

where the state vector now includes the stability x-axis velocity integrator state appended, thus $\underline{x}_{lng}'(t)=[u \ w \ q \ \theta \ i_u]$. Note that since the short period mode is well damped, this state penalty is small (0.05), whereas the large penalty on pitch angle (100000) is due to the fact that the phugoid is marginally open loop unstable and penalizing pitch angle heavily serves to damp this closed loop mode.

Lateral

State weights

$$Q_{lt} = \begin{bmatrix} 1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 400000 & 0 & 0 & 0 \\ 0 & 0 & 0 & 100000 & 0 & 0 \\ 0 & 0 & 0 & 0 & 100 & 0 \\ 0 & 0 & 0 & 0 & 0 & 1000 \end{bmatrix}$$

$$R_{lt} = \begin{bmatrix} 640000 & 0 \\ 0 & 320000 \end{bmatrix}$$

with the state vector including the appended integrators on sideslip and bank angle, thus $\underline{x}_{lat}'(t)=[v \ p \ r \ \phi \ i_v \ i_\phi]$. The control vector is $u_{lt}'=[\text{aileron rudder}]$. In the lateral case, the yaw rate (400000) and bank angle (100000) are penalized most heavily to improve the dutch roll mode, which has strong roll-yaw coupling due to the large dihedral effect. Note that in both lateral and longitudinal cases, the control activity is penalized heavily in order to maintain closed loop bandwidth close to open loop frequencies, while improving the damping of the poorly damped modes.

Designs are carried out over a range of airspeeds between 14 and 40 m/s with increments of 2m/s and then polynomial curves fitted to these data to give the final gains scheduled on dynamic pressure. These designs are initially done in continuous time with the models and weights presented above in order to obtain satisfactory frequency responses for the closed loop

systems, as the continuous time domain is a familiar one. Since the actual aircraft controllers are, however, implemented digitally, the complete procedure is redone with discrete time models, i.e. with the equivalent zero order hold discrete time models, using weights adjusted according to the method proposed in [Kwakernaak and Sivan] to obtain the equivalent discrete time controller which will yield the same continuous time behavior.

Figure 4 illustrates the discrete time gains scheduled on dynamic pressure for a digital implementation with a sample frequency of 16Hz.

The polynomial expressions for each gain are listed below.

LONGITUDINAL GAINS

Gu: airspeed to δ (elevator) gain

$$G_u = 4.8434e-14 Q^4 - 1.3238e-10 Q^3 + 1.406e-7 Q^2 - 8.7451e-5 Q + 6.1118e-2$$

Gw: w (proportional to alpha) to δ (elevator) gain

$$G_w = 1.8658e-13 Q^4 - 4.9721e-10 Q^3 + 4.9119e-7 Q^2 - 2.2684e-4 Q + 6.3982e-2$$

Gq: pitch rate to δ (elevator) gain

$$G_q = -3.712e-12 Q^4 + 9.8164e-9 Q^3 - 9.5556e-6 Q^2 + 4.2854e-3 Q - 1.1616$$

Gtheta: pitch angle to δ (elevator) gain

$$G_{\theta} = -2.2482e-12 Q^4 + 5.9523e-9 Q^3 - 5.8291e-6 Q^2 + 2.9096e-3 Q - 2.1711$$

Giu: airspeed error integrator to δ (elevator) gain

$$G_{iu} = 3.5333e-10 Q^2 - 3.0571e-6 Q + 1.368e-2$$

Figure 4. Controller gains scheduled with dynamic pressure ($0.5\rho U^2$). Polynomial curve fitting is used to obtain a suitable representation for implementing. Plotted are both actual data and fitted data.

LATERAL GAINS

Gvda: sideslip velocity to δ (aileron) gain

$$Gvda = -1.5194e-11 Q^3 + 3.5203e-8 Q^2 - 3.1728e-5 Q + 1.3159e-2$$

Gpda: roll rate to δ (aileron) gain

$$Gpda = 1.6293e-13 Q^4 - 4.3150e-10 Q^3 + 4.2054e-7 Q^2 - 1.9152e-4 Q + 6.7532e-2$$

Grda: yaw rate to δ (aileron) gain

$$Grda = 1.3219e-12 Q^4 - 3.5374e-9 Q^3 + 3.5059e-6 Q^2 - 1.5344e-3 Q + 4.6481e-2$$

Gphida: bank angle to δ (aileron) gain

$$Gphida = 1.2983e-12 Q^4 - 3.6203e-9 Q^3 + 3.8485e-6 Q^2 - 2.011e-3 Q + 0.97575$$

Givda: sideslip velocity integrator to δ (aileron) gain

$$Givda = -5.7465e-13 Q^3 + 3.0401e-9 Q^2 - 6.7718e-6 Q + 1.3155e-2$$

Giphida: bank angle error integrator to δ (aileron) gain

$$Giphida = -1.1468e-13 Q^4 + 3.1303e-10 Q^3 - 3.2183e-7 Q^2 + 1.601e-4 Q - 1.2427e-2$$

Gvdr: sideslip velocity to δ (rudder) gain

$$Gvdr = -5.3048e-14 Q^4 + 1.4233e-10 Q^3 - 1.4025e-7 Q^2 + 5.774e-5 Q - 2.773e-3$$

Gpdr: roll rate to δ (rudder) gain

$$Gpdr = -3.8423e-13 Q^4 + 1.0207e-9 Q^3 - 1.0026e-6 Q^2 + 4.4058e-4 Q - 3.2783e-2$$

Grdr: yaw rate to δ (rudder) gain

$$\text{Grdr} = -7.3582\text{e-}13 Q^4 + 1.729\text{e-}9 Q^3 - 1.3171\text{e-}6 Q^2 + 4.1448\text{e-}4 Q - 0.65204$$

Gphidr: bank angle to δ (rudder) gain

$$\text{Gphidr} = -4.0894\text{e-}12 Q^4 + 1.0893\text{e-}8 Q^3 - 1.071\text{e-}5 Q^2 + 4.802\text{e-}3 Q - 1.2924$$

Givdr: sideslip integrator to δ (rudder) gain

$$\text{Givdr} = -5.2137\text{e-}14 Q^4 + 1.4238\text{e-}10 Q^3 - 1.4593\text{e-}7 Q^2 + 6.8524\text{e-}5 Q - 5.323\text{e-}3$$

Giphidr: bank angle error integrator δ (rudder) gain

$$\text{Giphidr} = 5.0428\text{e-}12 Q^3 - 1.9804\text{e-}8 Q^2 + 3.2426\text{e-}5 Q - 5.8636\text{e-}2$$

IMPLEMENTATION

The autopilots are implemented in the form of the polynomial expressions of the gains as functions of dynamic pressure given above. In order to prevent integrator windup and other destabilizing issues the following rules on implementing are imposed:

Anti-aliasing Filters

Single pole low pass anti-aliasing filters of 5Hz bandwidth are implemented on each of the sensor input lines. The bandwidth is below the Nyquist maximum to prevent aliasing, and also is high enough that minimal phase lag is introduced due to the filters (approximately 2-3 degrees phase margin is lost in the short period mode, but since it is well damped, this translates to negligible degradation in performance. In the case of all other modes, the phase loss is further reduced.)

Integrator windup.

Whenever one of the actuators saturates, integration is deactivated until the aircraft reaches a state where the control system does not call for full actuator deflection, whence integration is reactivated.

Washout filters on rate signals.

All the rate gyro signals are washout filtered (high pass) to prevent opposing action by the controller in the case of desired steady rate maneuvers. For example, it is undesirable to have the yaw rate regulator opposing the necessary steady yaw rate in a fixed bank angle steady turn. The role of the washout filter is to block low frequency signals, thus eliminating this problem. The filter transfer function is

$$x_{wo}(t) = x(t) [s/(s + .1)]$$

or in discrete time (sample frequency of 16Hz)

$$\begin{aligned} x_{wo}(k+1) &= 0.99377 x_{wo}(k) + 0.06231 x(k) \\ y_{wo}(k) &= -0.1 x_{wo}(k) + x(k) \end{aligned}$$

where $x(k)$ represents either one of pitch rate, yaw rate or roll rate signals from the respective rate gyros and $y_{wo}(k)$ is the output from the filter, i.e. the "washed out" signal.

SIMULATION OF CONTROL SURFACE FAILURE

Since the design philosophy for the Perseus aircraft is that single point failures should be tolerable, the control systems should be robust to such possibilities, although a secondary backup may in some cases be available in the radio control backup (open loop) mode of pilot control.

This section shows performance of the closed loop system in the case of various single point actuator failures.

Figure 5. Results from the nonlinear simulation for the case of aileron hard over failure. This in fact simulates a dual failure, as the simulation assumes both aileron servos failed hard over. The reference bank angle is zero (wings level) and cruise speed is 20m/s. Servo deadband modeled is 2 degrees total. The rudder easily maintains good bank angle hold, whilst sacrificing sideslip. In the case of such a failure, the sideslip integrator is automatically cut out since the servo fully deflected is a software condition for this action. The bank angle integrator is still active.

Figure 6. Rudder hard over failure simulation. The strong dihedral effect proves this failure to be difficult to recover from by the ailerons, although the controller gives the correct response. The simulation plot shows sideslip, bank angle, rudder and aileron responses and it appears that the system may destabilize. The oscillation present is the dutch roll mode with ailerons fully deflected, attempting to counter the fully deflected rudder effect. Reduced dihedral effect would alleviate this problem, making the control system robust to such a failure.

Figure 7. Simulation of hard over failure of the right elevator servo deflected fully down, leaving only the left elevator servo active. The control system stabilizes the aircraft with greatly reduced elevator authority. This failure introduces disturbances to the lateral dynamics which are easily contained by the lateral controller as apparent in the sideslip and bank angle plots. The simulation includes 2 degree servo deadband and deflection limits of 15 degrees on all control surfaces.

TEST FLIGHT PERFORMANCE

The test flights of November 1991 were fundamentally aimed at system validation and since the autopilots behaved flawlessly from the first flight where the longitudinal autopilot was first activated at approximately one hundred feet AGL, flight testing of the communication systems and procedures and dropping of drop windsondes took priority over flight testing from the point of view of control system performance evaluation. For this reason, the only maneuver which is useful for post flight control performance evaluation, is the lost link maneuver, as this was successfully tested and all the autopilot reference inputs are known.

Comparison of the lost link flight test data with simulation data demonstrates good correlation, illustrating that the lateral control system performance was as expected. This data combined with video footage of the alpha and beta vanes on the nose boom demonstrates good regulation performance, with turns well co-ordinated (minimal sideslip) and angle of attack perturbations minimal.

Figures 8 and 9 illustrate the flight test data for two maneuvers. Figure 8 shows the lost link maneuver with simulation data overplotted for comparison. Figure 9 is flight test data for the transition in climbout after takeoff from open loop (radio control) to airspeed hold autopilot active. Since the reference inputs to the airspeed hold autopilot are unknown (not recorded at time of flight test) it is not possible to generate simulation performance data for comparison in this case. The well damped response of the aircraft in longitudinal control, however, illustrates the good performance of the airspeed hold autopilot. A further demonstration of the pilot's satisfaction and confidence in the autopilots' (both lateral and longitudinal) performance is the fact that all landings were done with both autopilots active.

Figure 8. Lost link maneuver. Shown is flight test and simulation data for comparison, in the lost link maneuver, where the aircraft is commanded into a steady turn in the case of a lost link with the ground station. The time histories show good correlation with the simulation data, indicating satisfactory performance of the lateral autopilot. Note good sideslip regulation and the clearly visible effect of the washout filter in the yaw rate time history.

Figure 9. Transition to airspeed hold autopilot active in climbout from takeoff. The transition involves a relatively sharp pitch down maneuver due to two possibilities. The pitch angle is much larger (positive, nose up) than the autopilot expects for straight and level flight and this combined with the low airspeed (35 knots) at which the airspeed hold function appears to have been activated (neutral stick commands ~41 knots in this mode) cause the autopilot to drop the nose of the aircraft rather sharply. In future versions, this transition may easily be made much smoother by low pass filtering the effective "step" inputs to the autopilot when activated with the actual flight condition at variance with the expected (by the autopilot) conditions.

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