N91-18162



# HIGH ALTITUDE RECONNAISSANCE AIRCRAFT

CALIFORNIA STATE POLYTECHNIC UNIVERSITY, POMONA

At the equator, the ozone layer ranges from 65,000 to 130,000+ ft which is beyond the capabilities of the ER-2, NASA's current high altitude reconnaissance aircraft. The Universities Space Research Association, in cooperation with NASA, is sponsoring an undergraduate program which is geared to designing an aircraft that can study the ozone layer at the equator. This aircraft must be able to cruise at 130,000 ft for 6 hr at Mach 0.7 while carrying 3,000 lb. of payload. In addition, the aircraft must have a minimum of a 6,000-mile range. The low Mach number, payload, and long cruising time are all constraints imposed by the air sampling equipment. In consideration of the novel nature of this project, a pilot must be able to take control in the event of unforseen difficulties. Three aircraft configurations have been determined to be the most suitable for meeting the above requirements, a joined-wing, a biplane, and a twin-boom conventional airplane. Although an innovative approach that pushes the limits of existing technology is inherent in the nature of this project, the techniques used have been deemed reasonable within the limits of 1990 technology. The performance of each configuration is analyzed to investigate the feasibility of the project. In the event that a requirement cannot be obtained within the given constraints, recommendations for proposal modifications are given.

## INTRODUCTION

The recent discovery of the ozone hole above the North Pole has prompted the scientific community to accelerate its efforts in investigating man's impact on his environment. The existence of the ozone hole has brought about concern that the predictions of stratospheric scientists may come true. In 1974, two chemists from the University of California, F. Sherwood and Mario Molina, theorized that the ozone layer was being destroyed by chlorofluorocarbons. Unless the ozone depletion in the Earth's atmosphere is controlled, radiation levels at the surface may increase to harmful levels. At the tropics, the ozone layer ranges from 65,000 ft to 130,000+ ft which is beyond the capabilities of the ER-2, NASA's current high altitude reconnaissance aircraft. Therefore, to effectively investigate the ozone layer, NASA needs to develop a high-altitude aircraft that will reach altitudes of 130,000+ ft. To hasten the development of the technology and methodology required to develop an aircraft that can reach these altitudes, the NASA/USRA program has been working closely with industry and universities. Perhaps, with the data retrieved from this aircraft, scientists and politicians will be able to formulate an emissions control plan that will diminish the rate of degeneration of the ozone layer.

## DESIGN PROCESS

The 1989-1990 school year was the second in a three-year, ongoing design project geared to the design of a high-altitude reconnaissance aircraft. California State Polytechnic University, Pomona, has its yearly design sequence separated into three consecutive quarters. Basically, the assignment at the beginning of each year is to do a preliminary design analysis to determine the aircraft that best fits the Request for Proposal requirements. If such an aircraft is not deemed feasible, the aircraft must still be designed, with those aspects which are not approachable indicated in the concluding comments. Suggestions for making the Request for Proposal feasible are also requested. During the fall quarter, three groups were formed, aerodynamics, propulsion, and structures. After a short break, the groups reconvened during the winter quarter to decide on the best possible configurations and commence their design. The final design iteration was completed, and the final report was compiled in the spring quarter.

The three fall quarter groups were given the Request for Proposal and instructed to identify the potential problems in their area of expertise. Once the problem areas were identified, possible solutions were considered and analyzed in detail. From this analysis, the design process was established, and possible configurations were determined. The pros and cons of each configuration as it pertained to the specialty groups of aerodynamics, propulsion, and structures were collected.

At the beginning of winter quarter, the three most plausible designs were chosen based upon the analysis of the previous quarter. For reasons discussed later, the three configurations selected were a conventional twin-boom monoplane design, a joined-wing design, and a biplane design. The three groups were then reassembled into three new teams based upon the configuration of each individual's preference. At the same time, the team leaders from the original groups were assigned to consult the groups on any problem areas that were investigated the previous quarter. If all the groups suffered from the same difficulty during the quarter, the consultant was authorized to temporarily reassemble his original team in an attempt to solve the most common and pressing problems quickly and efficiently. In this manner, the students were given the experience of working with a matrix management system on a small scale.

As the spring quarter commenced, the final configurations were set. Each group wrote a 100-page report on their preliminary design findings. These were assembled into three volumes and made available through the USRA program.

## DESIGN SPECIFICATIONS

Ideally, the scientific community would like an aircraft that meets the following specifications:

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**RFP Specifications**

1. The cruise altitude is130,000 ft.
2. The payload capacity is 3,000 lb.
3. The design cruise Mach number is 0.7.
4. The cruise is a minimum of 6 hr.
5. The range is a minimum of 6,000 miles.
6. There is a minimum of one pilot.
7. The aircraft is to be designed with present technology.

These specifications meet the most optimistic demands of the stratospheric scientists. The results of previous studies have shown that flight at 100,000 ft with a range of 3250 n.m. is possible. Unfortunately, a mission at the lower altitudes would not give an accurate estimate of the chemical activity within the ozone layer at the equator. The ozone layer at the tropics is in the range of 65,000 to 130,000+ ft, as opposed to 50,000 to 100,000 ft at the mid-latitudes and 35,000 to 95,000 ft at the poles. The largest perturbations of the ozone are expected to be at 130,000 ft at the mid-latitudes. This fact coupled with an airplane's ability to follow an experimenter chosen path makes an airplane meeting the above specifications an ideal ozone testing platform[1].

Some of the constraints on the Request for Proposal are imposed by the sampling equipment, which is a modification of that in current use on the ER-2[2]. The increase in air temperature and the dissociation in the flow cause air samples to loose accuracy as compressibility effects become significant; therefore, the Mach number must be below the transonic regime. At the same time, the low air density (0.00003211 slugs/ft$^3$) at altitude implies low wing loadings and high wing planform areas. Figure 1 illustrates the variation of air density with altitude. All of these adverse effects become more

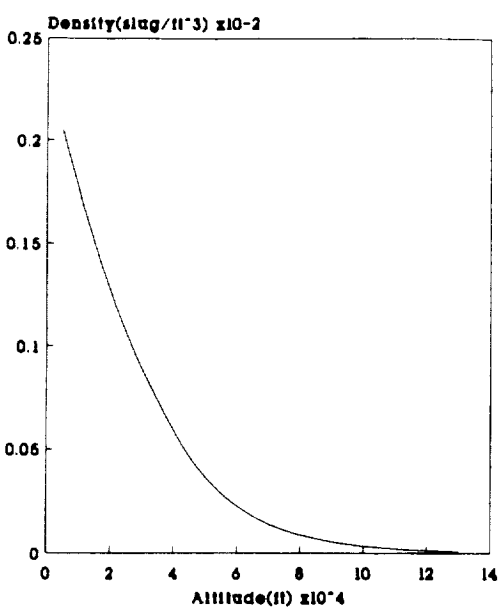


Fig. 1. Air Density vs. Altitude

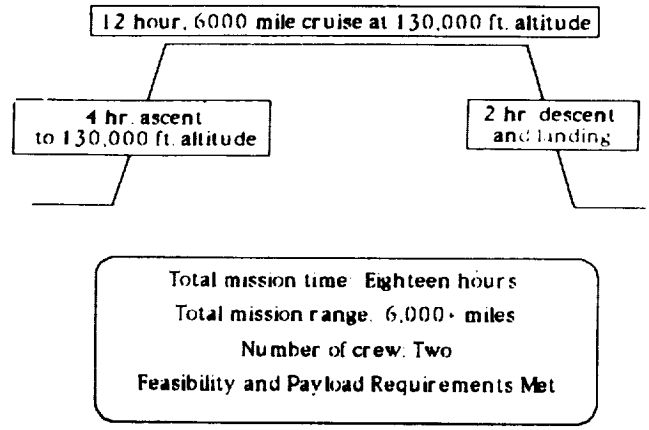


Fig. 2. Mission Profile

significant with decreasing Mach number. A Mach number of 0.7 was chosen to balance the contradicting effects of compressibility and air density. The air sampling equipment also dictates the cruise time and range. Stratospheric scientists are unable to obtain an accurate mapping of the ozone layer without extensive measurements that span a large area. The 6,000-mile range is easily accomplished within the specified minimum time constraints. As shown in Fig. 2, the total mission time is in fact on the order of 18 hours with a 12-hour cruise. The long mission time prompted the groups to design for two pilots in order to diminish fatigue.

The present technology requirement is desirable in order to acquire the maximum utility from this vehicle. In mid-1993, the Cryogenic Limb Array Etalon Spectrometer (CLAES), an instrument designed to monitor the ozone layer on the Upper Atmosphere Research Satellite scheduled for launch in 1991, will become inoperational. The first Earth Observing System (EOS) sensors are scheduled to become operational in 1996, at the earliest. It is during this testing gap that the results from a high altitude aircraft will be most crucial. After the EOS comes on-line, the aircraft will be used to cross-calibrate the measurements from the EOS and ground-based sensing instruments[1].

## CONFIGURATIONS

The configurations considered for this aircraft are (1) Flying wing, (2) Monoplane-conventional, (3) Monoplane-twin-boom, (4) Canard, (5) Joined wing, and (6) Biplane-twin-boom.

The flying wing has a high aerodynamic efficiency due to the lack of a horizontal tail. However, it has the disadvantage of stability problems coupled with poor takeoff rotation. These factors rendered this design undesirable.

The monoplane with the conventional fuselage tends to be stable and predictable. The large wingspan required would produce excessive bending moments that a single fuselage could not counteract. On the other hand, a twin boom fuselage structure would relieve the structural loads while maintaining

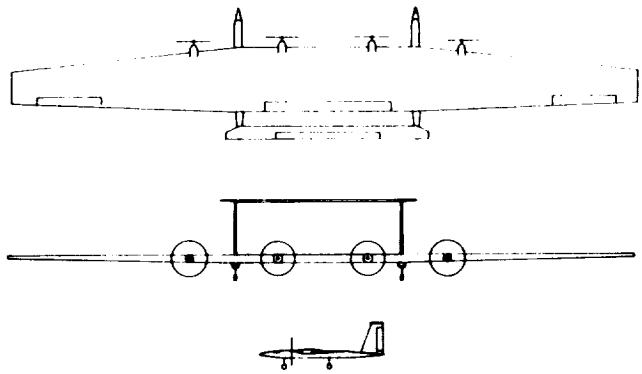

Fig. 3. *Global Sentry*

the advantages of stability and ease of analysis. The final design for the twin-boom monoplane is shown in Fig. 3[3].

A canard configuration is similar to a flying wing in that it has many of the same advantages and disadvantages. No justification for using a canard configuration could be found.

The joined wing aircraft at first seems ideal with its high aerodynamic efficiency and high structural strength. Unfortunately, a joined wing aircraft is not a proven design. Therefore, the extra testing may render it not cost effective. Despite this possible failure, the aircraft appears to be worth analyzing. The three-view for this aircraft is shown in Fig. 4[4].

A twin-boom biplane is structurally sound, minimizes the span, has good propeller clearance, and has a large frontal area. Its only apparent disadvantage is the interference from the wing struts. Considering the possibility that the strut interference may not be sufficient to undermine the advantages of the design, this aircraft is being considered further. Figure 5 shows a three-view.

In summary, the three designs chosen for further investigation were the twin-boom monoplane, the joined wing, and the twin-boom biplane. The three projects are called *Global Sentry*, *Icarus*, and *Hi-Bi*, respectively.

## AERODYNAMICS

The two design drivers in the area of aerodynamics are airfoil selection and propeller design.

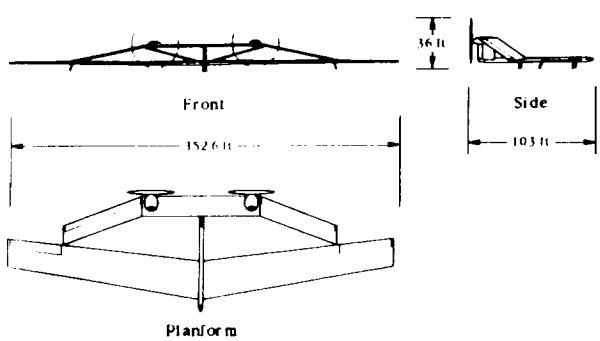


Fig. 4. *Icarus* High Altitude Aircraft

### Airfoil Design

The airfoil design criteria are high lift and low drag at cruise conditions. In addition, the rarefied flow at the cruise altitude introduces low Reynolds number aerodynamic phenomena. For this reason, the airfoil has a tendency toward laminar separation bubbles and compressibility effects, which must be avoided. For the conventional configuration, a low pitching moment is required, but for the joined wing configuration it is not so crucial, since the moment can be balanced with the other wing. To accommodate fuel storage requirements, a maximum thickness ratio at 10% of the chord is preferred.

In general, supercritical airfoils conform to these criteria. A modification of Richard Eppler and Dan M. Somers' ES-989 was found to best suit the needs of all three configurations. A computer code authored by Mark Drela called XFOIL was used to modify and analyze the airfoil. The code was able to tailor the pressure distribution to reduce shocks and flow separation. The resulting pressure distribution is shown in Fig. 6. XFOIL is prone to errors in integration. This manifests itself in excessive peaks in the pressure distribution at the leading edge and a slightly higher Mach number distribution as compared to test data for similar airfoils. However, despite these potential problems the performance characteristics of the final modification compare well with published data for similar airfoils designed for low Reynolds number flight[6].

### Propeller Design

Initially, XFOIL's counterpart, XROTOR, was considered for the propeller design. Unfortunately, it was found that XROTOR's tendency to optimize the propeller blade loading produced excessive propeller root chords on the order of 50 ft. As a result, the propellers were hand designed. They were optimized to produce the lowest section drag coefficients. There were two main criteria for designing the propellers. The first and foremost was that the tip velocities can not exceed

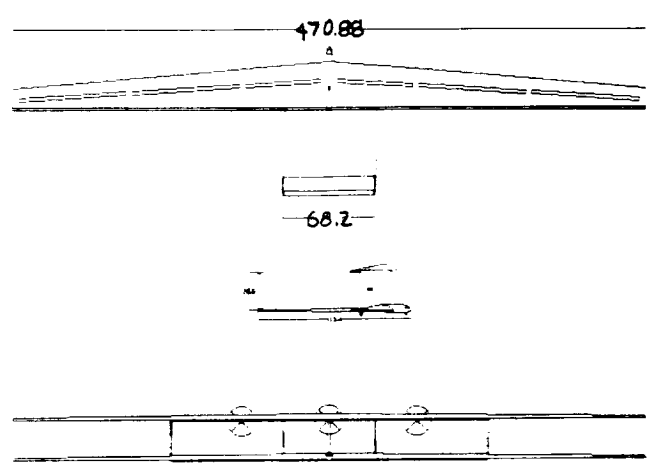


Fig. 5. *Hi-Bi*

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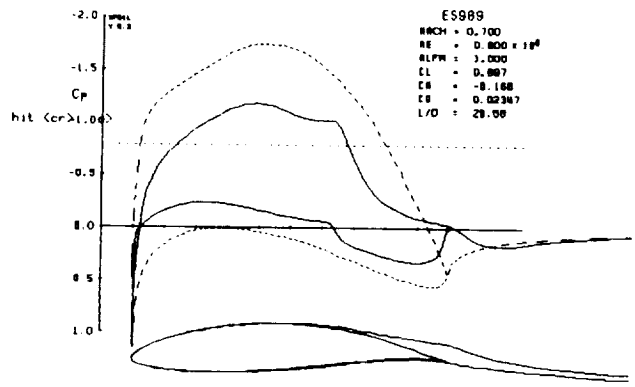



Fig. 6. XFoil

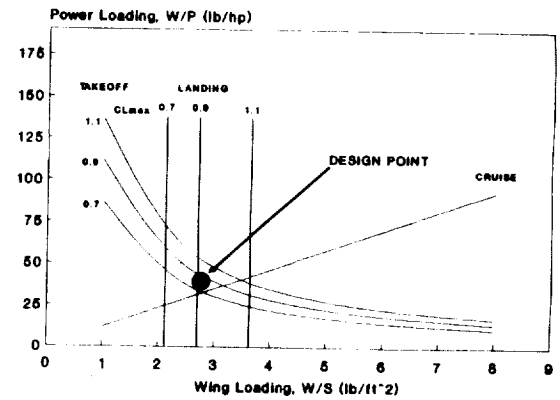



Fig. 8. Preliminary Sizing

the drag divergence Mach number. Since the air density is so low, the rpm and diameter need to be high.

Because of differences in ground-tip clearance, each configuration has a slightly different propeller design. The data for the six-blade, single-rotating propeller system that the *Icarus* chose, shown in Fig. 7, is a typical example.

## Performance

From the sizing chart shown in Fig. 8, it is evident that in order to meet the constraints imposed by the Request for Proposal, the wing loading is limited to a range of approximately 2.8 to 3.2 psf[7]. With these wing loadings, takeoff is not a problem. The takeoff distances are rather short, and high lift devices in the form of flaps and slats are generally considered unnecessary. Figure 9 shows a typical take-off analysis chart.

The best rate of climb is chosen from the rate of climb versus velocity graph shown in Fig. 10. For the *Icarus* project, the time to climb was chosen to be 3.87 hr. This is shown in Fig. 11. With this knowledge the fuel weight for climb is estimated to be 1330.75 lb. The time to climb for each configuration varied from 1.5 hr to 3.87 hr depending upon what parameter was optimized.

The flight envelope for all three aircraft is similar. The aircraft are constrained by the laminar stall velocity at lower speeds and by maximum power at higher speeds. Typically, high altitude aircraft have a very narrow flight envelope. These three designs are no exception as shown in Fig. 12.

Figure 13 shows the power required curve as a function of altitude. Figure 14 emphasizes the cruise condition. It is clear that the aircraft is flying within its power requirements at all times.

The landing characteristics are summarized in Fig. 15. The total landing distance is approximately 640 ft.

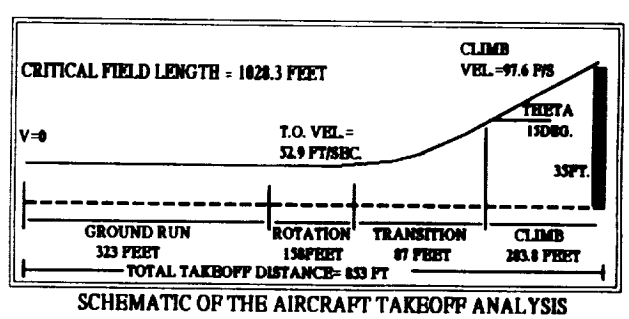
## PROPULSION SYSTEM

The mission profile for this aircraft sets very stringent requirements for the propulsion system. The powerplant for

| | |
|---|---|
| • Number of Blades | 6 |
| • Diameter | 30 feet |
| • Revolutions | 572.96 rpm |
| • Advance Ratio (J) | 2.59 |
| • Phi $_{.7R}$ (helix angle) | 61 degrees |
| • Activity factor | 1000 (166.67 per blade) |
| • Mach tip | .85 |
| • Propeller section | Naca 16-series |
| • Cruise Overall Efficiency | 75.7 % |

Fig. 7. *Icarus* Propeller Data

STALL VELOCITY = 44 ft./sec.
C-L max. = 1.3
WING LOADING = 3.1
THRUST LOADING =.25
ROLLING COEFFICIENT = .03




SCHEMATIC OF THE AIRCRAFT TAKEOFF ANALYSIS

Fig. 9. Takeoff Analysis

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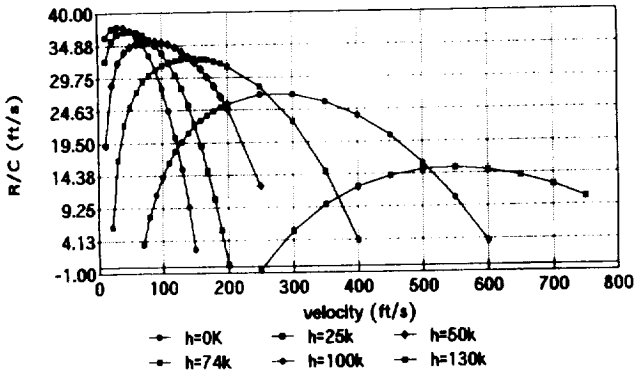



Fig. 10. Rate of Climb as A Function of Airspeed and Altitude

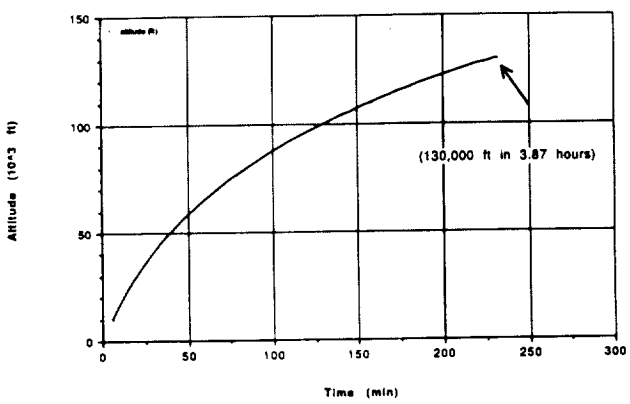



Fig. 11. Time to Climb

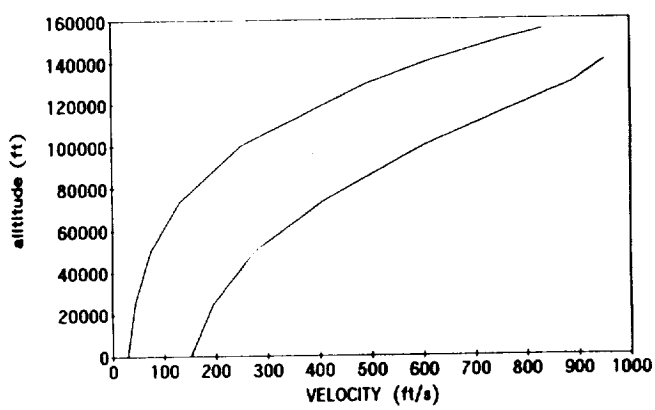



Fig. 12. *Hi-Bi* Flight Envelope

this aircraft must be able to operate with a low specific air consumption. The 6,000-mile range requirement necessitates that the powerplants have a low specific fuel consumption to reduce the amount and weight of fuel needed to complete the mission. Since the aircraft operates at subsonic velocities and very high altitudes, the aircraft's wings are large and heavy. This requires an engine that is capable of producing large amounts

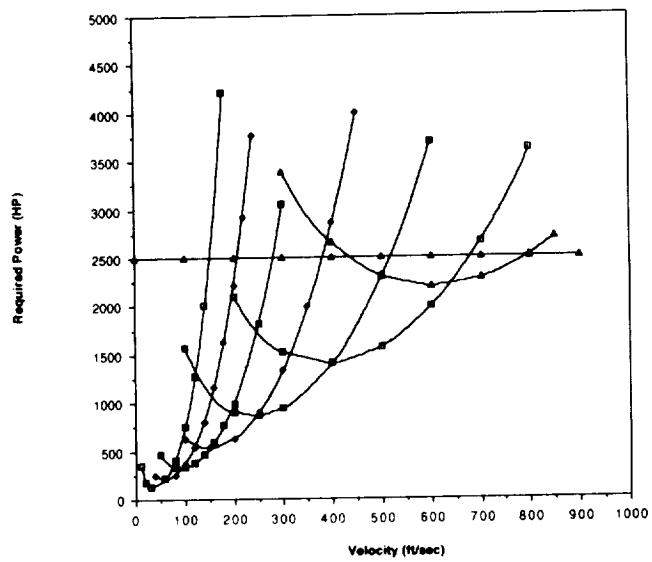



Fig. 13. Power Required Curve

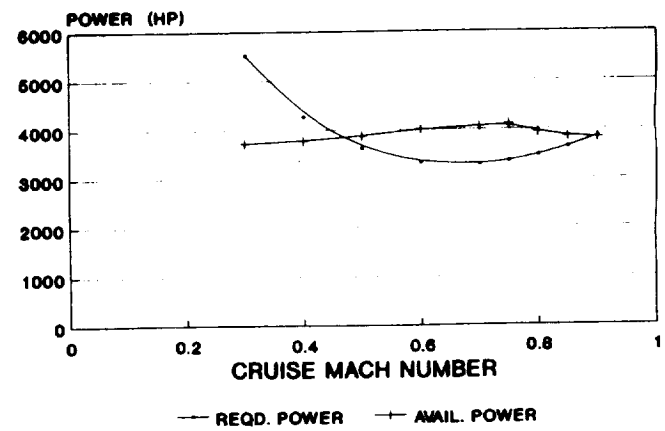
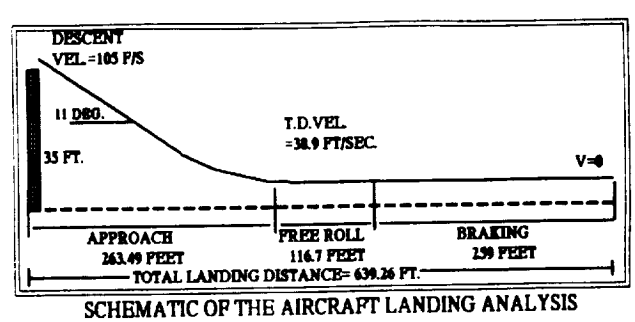



Fig. 14. Power Optimization

STALL VELOCITY = 33.85 ft./sec.
C-L max. = 1.3
WING LOADING =1.92
THRUST LOADING = .2
BRAKING COEFFICIENT = .1




SCHEMATIC OF THE AIRCRAFT LANDING ANALYSIS

Fig. 15. Landing Analysis

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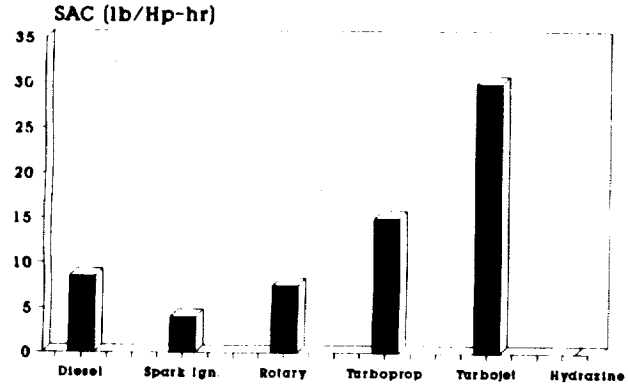


Fig. 16. Specific Air Consumption for Various Engine Types

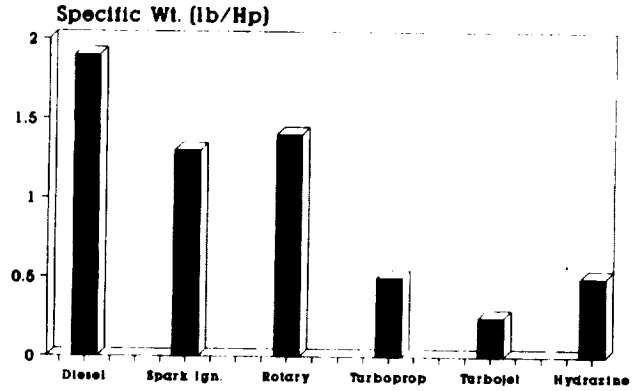


Fig. 18. Specific Weight for Various Engine Types

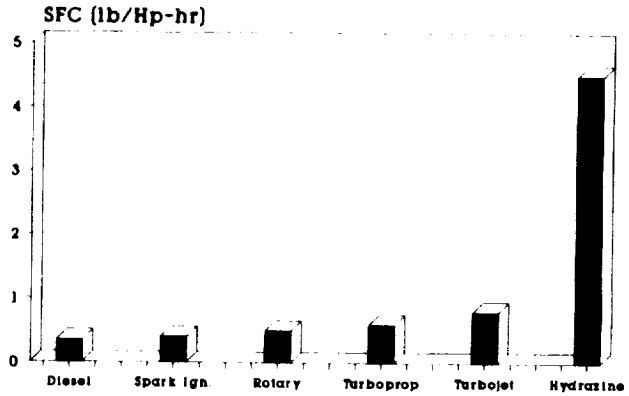


Fig. 17. Specific Fuel Consumption for Various Engine Types

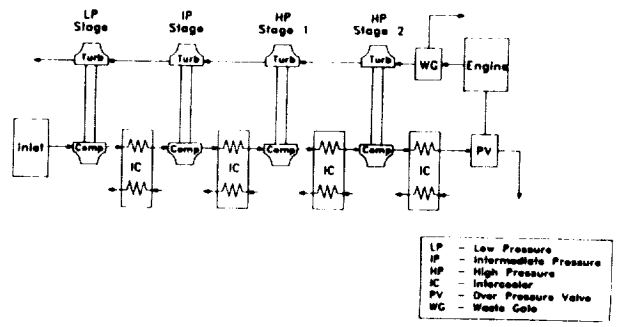


Fig. 19. Schematic of the Four Stage Turbocharging System

of power at altitude. The final requirements are to keep the engine and its systems as light as possible and to develop this system with current technology.

## Powerplant Selection

The driving constraint in the engine selection process is the air consumption of the engine at altitude. The air consumption has to be low for the engine to produce power at altitude. Figure 16 shows typical specific air consumption values for the engines examined. The second constraint is the propulsion system weight, which has to be kept as low as possible. Figures 17 and 18 show typical specific fuel consumption and specific weight values for the engines examined.

The low density of air at altitude and subsonic cruise velocity combined with the engine's high specific air consumption make it impossible for any turbojet or turbofan engine to produce any meaningful thrust. Turboprops follow the same trend as the turbojet producing little power at altitude. The hydrazine engine is also an unlikely candidate since it has an extremely high specific fuel consumption and is extremely toxic.

Internal combustion engines have a relatively low specific air and fuel consumption. Nonetheless, they are unable to produce enough power at altitude without some type of turbocharging. The Lockheed HAARP Project designed a turbocharging system to operate with an internal combustion engine at an altitude of 100,000 ft. Of the three internal combustion engines examined, diesel, rotary, and spark ignition, the spark ignition engine had the best mix of s.a.c., s.f.c., and specific weight.

Other engine technologies such as microwave propulsion, laser propulsion, nuclear propulsion, and electrical propulsion were examined. Practical versions of these engines are not feasible with present day technology; therefore, there is no merit in further investigation. Thus, the spark ignition engine was selected as the best choice for the high altitude propulsion system.

## Engine Configuration

The concept is based on an engine designed by Continental Teledyne Motors. It is a 500 hp engine designed to cruise at 100,000 ft with three stages of turbocharging.

| | |
|---|---|
| Turbocharger Type | Radial |
| Over All Pressure Ratio | 432:1 |
| 1st Stage Pressure Ratio | 3:1 |
| 2nd Stage Pressure Ratio | 4:1 |
| 3rd Stage Pressure Ratio | 6:1 |
| 4th Stage Pressure Ratio | 6:1 |
| Maximum Mass Flow Rate | 120.5 (lb/min) |
| Maximum Pressure Obtained at 130,000 ft. | 1788 (psia) |
| Inlet Size | 8.7 (ft^2) |
| System Weight | 900 (lb) |

Fig. 20. Specifications of the Four Stage Turbocharger System

| | |
|---|---|
| Engine Type | IC Spark Ignition |
| Number of Cylinders | 8 |
| Cylinder Arrangement | Horizontal Opposed |
| Bore and Stroke | 5.25 in and 6.5 in |
| Displacement | 1125 cu in |
| Compression Ratio | 10:1 |
| Width and Height, Engine | 38 in and 29.25 in |
| Width and Height, Installed | 41 in and 59.8 in |
| Length and Frontal Area, Engine | 33.6 in and 7.7 sq ft |
| Length and Frontal Area, Inst. | 69.6 in and 16.4 sq ft |
| Engine Weight | 11177 lb |
| Total Weight, Installed | 2077 lb |
| Weight/Horsepower | 1.89 lb/Hp |
| Fuel Grade | 100 LL |
| SFC, Cruise and Max Power | 0.357 and 0.383 lb/Hp-hr |
| SAC, Cruise and Max Power | 5.684 and 5.45 lb/Hp-hr |
| Cruise Power | 962 Hp/3900 RPM @ 130k ft. |
| Max Power | 1194.9 Hp/4250RPM @ S.L.<br>1100 Hp/4250 RPM @ 130k ft. |

Fig. 21. Performance Specifications, 960 hp Engine

The engine designed for this project uses four stages of turbocharging to allow it to operate at a higher altitude. Turbocharging was selected over supercharging so that the minimum engine power is required to run the engine. Figure 19 shows a schematic of the turbocharging system. Figure 20 tabulates the specifications of the system. The turbochargers are each composed of a radial compressor and a radial turbine. Each of the four turbocharger stages are intercooled with a crossflow air to air heat exchanger.

The high altitude engine is arranged in a horizontal opposed configuration to reduce frontal area and allow an aerodynamic cowling to be fitted around the engine. The block is composed of two forged aluminum alloy pieces bolted together vertically. The crank shaft is a forged steel, eight-throw, one-piece design and is supported by five journal bearings. The engine has eight, 10:1 compression ratio, aluminum alloy pistons displacing 1125 cu in.

The powerplant is modeled on a modified engine program[8]. Figure 21 shows the specifications and performance for the engine. Figure 22 gives the cycle information.

## WEIGHTS AND STRUCTURES

Of the three configurations, the joined wing was found to be the most structurally sound. It was modeled on NASTRAN with the joint at 70% of the semi-span and graphite/epoxy honeycomb sandwich spars. The composite fiber orientation is 0, 45, 90, -45, 0, -45, 90, 45, 0. The wing configuration, wing box, shear force and bending moment diagrams are shown in Fig. 23-25. The maximum deflection is 10.4 feet at the tip of the front wing. The total gross takeoff weight is 41,200 lb.

## RELIABILITY

Figure 26 shows the results of a reliability analysis for the Global Sentry. All three aircraft yield comparable results. The graph indicates the probability of a component failing as a function of mission time. Generally, the mission would have to be aborted 43% of the time[10].

## CONCLUSIONS

The mission would be more likely to succeed if the Request for Proposal is modified. There is some doubt as to whether the aircraft necessary to meet the constraints can be built using present technology. This conclusion concurs with parallel analyses being conducted by NASA and Lockheed[1,9].

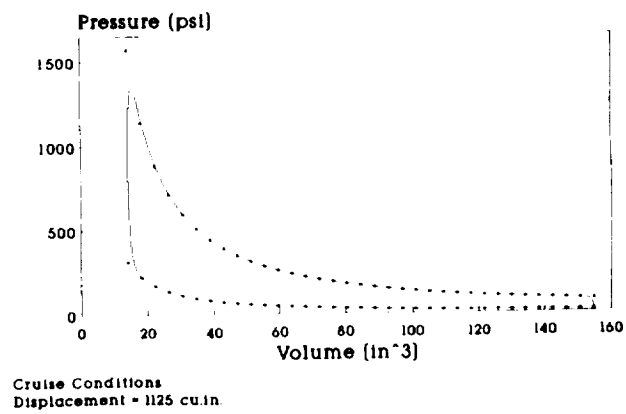


Fig. 22. Pressure vs. Volume Diagram, 960 hp Engine

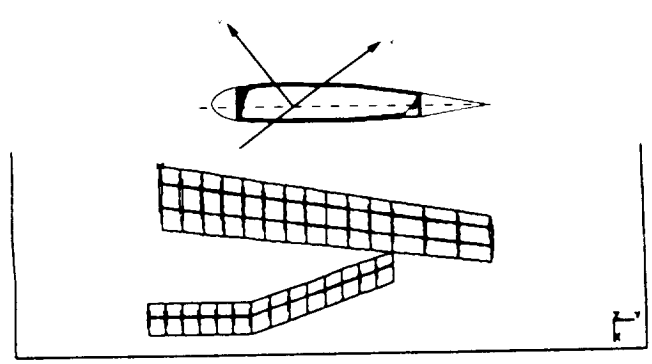

Fig. 23. Wing Box for Joined Wing

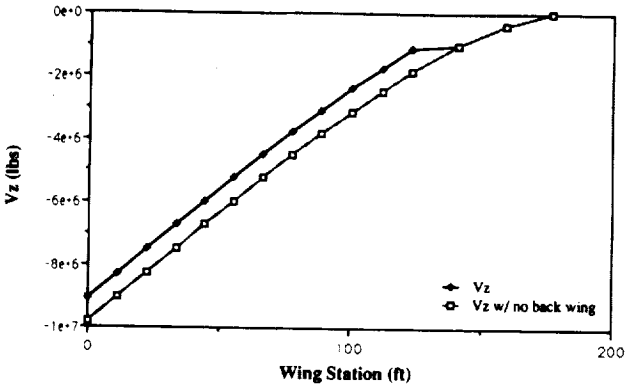


Fig. 24. Shear Force Diagram

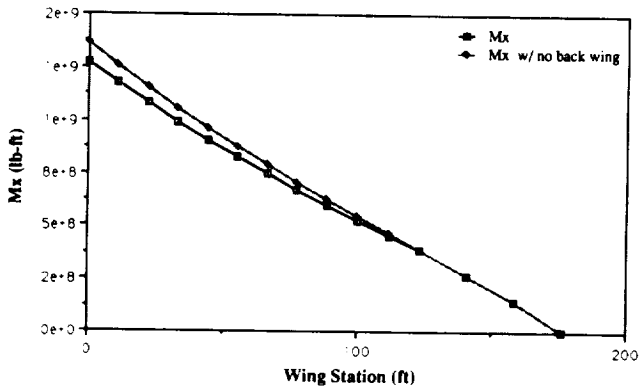


Fig. 25. Bending Moment Diagram

Suggested modifications to the Request for Proposal are as follows:

1. Decrease the cruise altitude to 100,000 ft with possible zooms to 130,000 ft.
2. Split the mission into a 6,000-mile unmanned mission and a 6-hour manned mission.
3. Decrease the cruise Mach number to 0.6.

These modifications should act to decrease the span which in turn makes the aircraft manufacturable and increases structural integrity. The present spans, which range from 400 to 450 feet, render it impossible to land at most airports. It would be more reasonable to design for a 150-ft-wide runway with four foot-high obstacles located 20 ft off the runway. Furthermore, the reliability will increase with the decrease in mission time. The four-stage, turbocharged propulsion system could be brought down to three stages, which are generally considered possible. Some work has been done on a three-stage engine in recent years but none on the four-stage.

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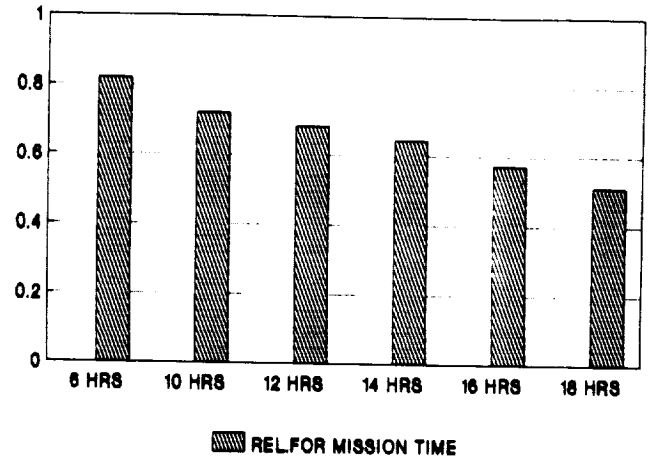


Fig. 26. Reliability Analysis of the *Global Sentry*

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