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HIGH-ALTITUDE RECONNAISSANCE AIRCRAFT

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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. This project is geared to designing an aircraft that can study the ozone layer. The aircraft must be able to satisfy four mission profiles. The first is a polar mission that ranges from Chile to the South Pole and back to Chile, a total range of 6000 n.m. at 100,000 ft with a 2500-lb payload. The second mission is also a polar mission with a decreased altitude and an increased payload. For the third mission, the aircraft will take off at NASA Ames, cruise at 100,000 ft, and land in Chile. The final mission requires the aircraft to make an excursion to 120,000 ft. All four missions require that a subsonic Mach number be maintained because of constraints imposed by the air sampling equipment. Three aircraft configurations have been determined to be the most suitable for meeting the requirements. The performance of each is analyzed to investigate the feasibility of the mission requirements.

INTRODUCTION

The recent discovery of the ozone hole above the North Pole has prompted the scientific community to accelerate investigations of humans' impact on the environment. The existence of the ozone hole has raised concern that the predictions of stratospheric scientists may come true. In 1974 two chemists from the University of California, F. Sherwood and M. Molina, theorized that the ozone layer was being destroyed by chlorofluorocarbons. Unless ozone depletion in the Earth's atmosphere is controlled, radiation levels at the surface may increase to harmful levels. To effectively investigate the ozone layer, NASA needs to develop a high-altitude aircraft that will reach altitudes of 130,000+ ft. To hasten development of the technology and methodology required to design an aircraft that can reach these altitudes, the NASA/USRA program has been working closely with industry and universities. With the data retrieved from this aircraft, scientists and politicians perhaps will be able to formulate an emissions control plan that will diminish the rate of degeneration of the ozone layer.

DESIGN PROCESS

The 1990-1991 school year was the third in a three-year, ongoing design project on a high-altitude reconnaissance aircraft. The assignment at the beginning of each year is to perform a preliminary design analysis to determine the aircraft that best fits the requirements. If such an aircraft is not deemed feasible, the aircraft must still be designed, with those requirements that 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. Each group investigated design drivers for the aircraft and did preliminary research for configuration, propulsion system, and airfoil selection. The groups reconvened during the winter quarter to commence their design. The final design iteration was completed, and the final report was compiled in the spring quarter. These were assembled into three volumes and made available through the USRA program.

DESIGN SPECIFICATIONS

The objective is to develop three possible designs for an aircraft that can cruise at 100,000 ft and sample the ozone layer at this altitude. Ideally, the scientific community would like the aircraft to meet the four mission profiles depicted in Figs. 1-4[(1)]. The requirements and constraints are listed as follows:

1. The aircraft must fly subsonically at all stages of the mission.
2. If the aircraft is manned, it must have redundant life-support systems and be pilot friendly. If an unmanned mission is chosen, proof that it is a better alternative must be provided.
3. The crosswind capability must be a minimum of 15 knots with moderate to severe turbulence.
4. Spoilers or alternative lift dump devices are to be provided for low wing loading landing.
5. For safety and flexibility, two engines are a minimum.
6. The hangar is 110×70 ft.
7. The aircraft enters production before the year 2000.

These specifications meet most of the 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[(2)]. Unfortunately, the current ER-2 missions at altitudes of 60,000 to 70,000 ft do not give an accurate estimate of the chemical activity within the ozone layer at the equator. The ozone layer at the tropics is within 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. 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 are imposed by the sampling equipment, which is a modification of that in current use on the ER-2[(3)]. Increase in air temperature and the dissociation of flow cause problems with air sampling as compressibility effects become significant; therefore, the Mach number must be below the transonic regime. At the same time, low air density (0.00003211 slugs/ft^3) at altitude implies low wing loadings

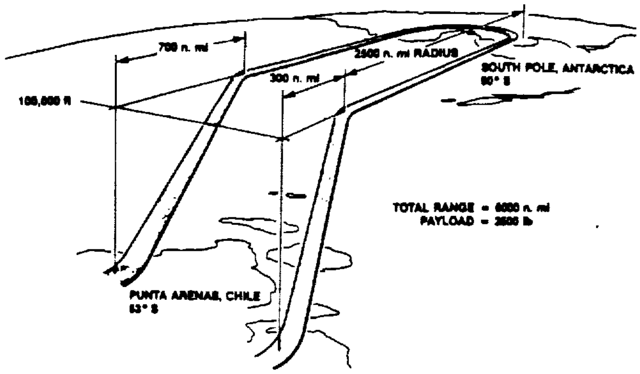

Fig. 1. Chile to South Pole to Chile, 5000 n.m. at 30 km (100,000 ft).

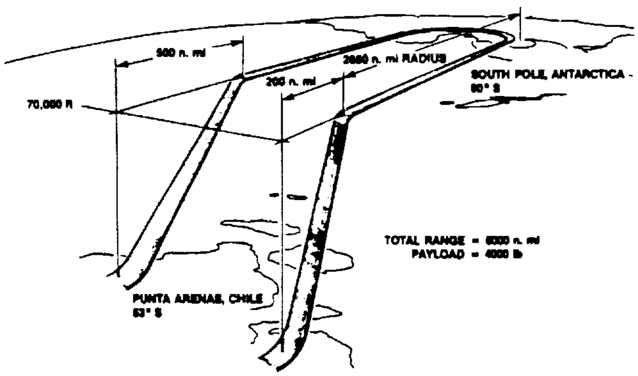

Fig. 2. Chile to South Pole to Chile, 5000 n.m. at 21 km (70,000 ft).

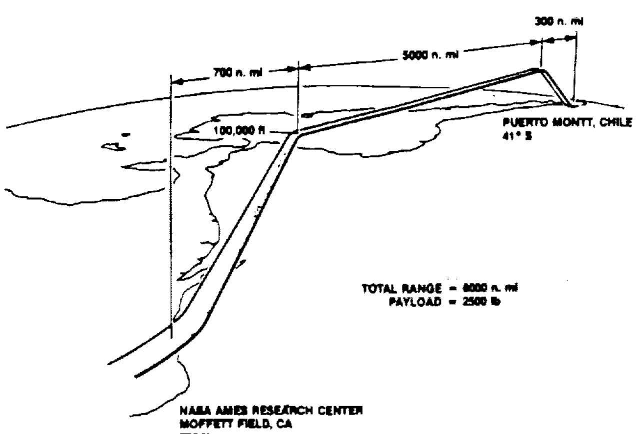

Fig. 3. NASA Ames to Chile, 5000 n.m. at 30 km (100,000 ft).

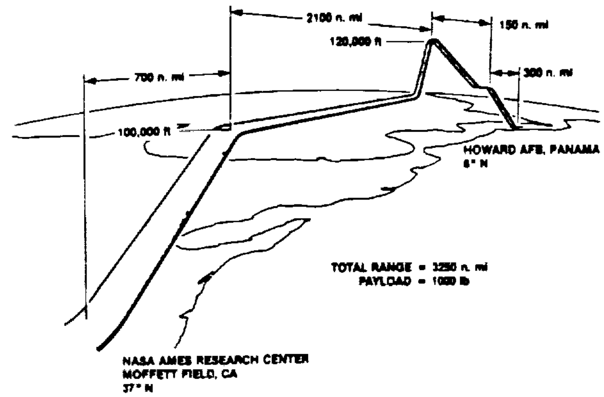

Fig. 4. NASA Ames to Panama, 3250 n.m. at 30 km (100,000 ft) with excursion to 37 km (120,000 ft).

and high-wing planform areas. Figure 5 illustrates the variation of air density with altitude. All these adverse effects become more significant with decreasing Mach number. The Mach number chosen must balance the contradictory 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 over a large area.

The aircraft must be operational by the year 2000 in order for 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 cease operation. The first Earth Observing System (EOS) sensors are scheduled to become operational in 1996 at the earliest. After EOS comes online, the aircraft will be used to cross-calibrate measurements from EOS and ground-based sensing instruments[1].

CONFIGURATIONS

Configurations for the three high-altitude research aircraft were selected according to the following criteria: (1) large wing area—minimum span, maximum aspect ratio, (2) maximum aerodynamic efficiency, (3) low wing-tip bending or twisting, (4) minimal structural weight, (5) ample ground clearance, and (6) minimum cost.

The configurations considered for these aircraft are (1) flying wing, (2) monoplane, (3) canard, (4) joined wing, (5) biplane, and (6) tandem wing.

The flying wing has a high aerodynamic efficiency because it has no horizontal tail. However, it has the disadvantage of stability problems coupled with poor takeoff rotation attributed to the lack of propeller ground clearance. The controllability of a flying wing can be increased by sweeping the wing, but this yields a decrease in laminar flow and reduced aerodynamic efficiency. These factors rendered this design undesirable.

The monoplane has the advantage of being a simple, proven configuration. The disadvantage is that the large wing span required would produce excessive bending moments that would

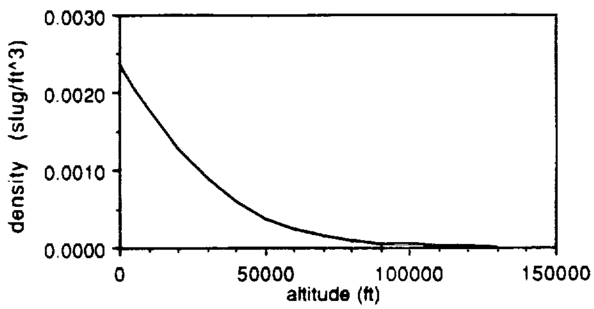

Fig. 5. Density variation.

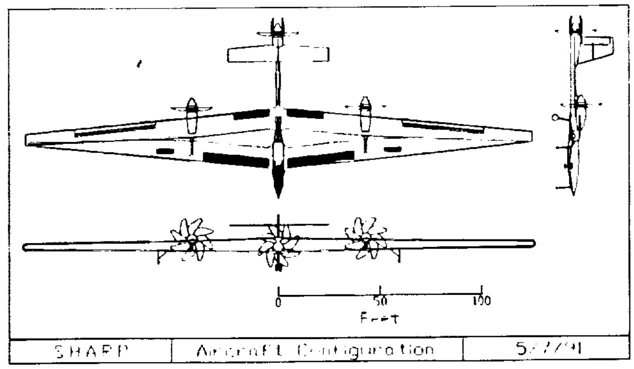

Fig. 6. SHARP.

be difficult for a single fuselage to counteract. On the other hand, a twin-boom fuselage would relieve the structural loads while maintaining ease of analysis. However, the structural loads would still be greater than those for a tandem wing with a twin-boom fuselage, so this design was eliminated.

A canard configuration has advantages and disadvantages similar to a flying wing. 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, like the flying wing, the joined wing incurs some aerodynamic penalties by using swept wings. In addition, the joined-wing aircraft is not a proven design. Therefore, the extra testing required may render it cost ineffective.

A twin-boom biplane is structurally sound, minimizes the span, and has good propeller clearance. Its most apparent disadvantage is interference from the wing struts. Allowing for the possibility that the strut interference may not be sufficient to undermine the advantages of the design, this aircraft is being considered further. Figure 6 shows a three-view. The struts that join the tips are to prevent the tips from touching when the wing is deflected. There are also reinforcing struts located under the engines. Classifying this aircraft as a biplane is controversial; some have referred to it as a joined-wing hybrid. However, it was analyzed as a biplane. Wind tunnel testing should be done in the future to verify the analysis procedure.

The tandem wing, like the joined-wing and the biplane configurations, minimizes the span by using two wings instead of one. At the same time, the wing bending and structural weight are better than the monoplane configuration as shown with the Rutan *Voyager* aircraft. The resulting structural weight for the *Voyager* was 9.7% of the total weight compared to 25% for most conventional monoplane configurations. The tandem-wing configuration also provides a lower induced drag. If interference effects between the two wings are not considered, the induced drag is half that of a monoplane. Interference effects can be reduced significantly by employing a negative stagger, which places the rear wing away from the downwash of the front wing. It should be noted that the drag reduction is not always realized, as indicated from the *Voyager* data, which suggested an increase in aerodynamic drag over conventional sailplane designs[(4)]. Ground clearance is not a problem if the engines are mounted on the higher wing. Two slightly different tandem-wing designs were studied, as shown in Figs. 7 and 8. Wind tunnel tests to verify the drag calculations would be needed at a later date in lieu of the *Voyager* results.

Two of the non-airplane considerations for this project were balloons and sounding rockets. Balloons are currently being used with some effectiveness, but their range varies from 100 to 1000 n.m. and they are not controllable. Sounding rockets have also proven effective in the past, but they too have range restrictions compounded by an endurance of only minutes.

Several launch methods were considered. Conventional runway, balloon ascent, and towed takeoff were the three alternatives considered most feasible for an aircraft with spans in excess of 200 ft. The conventional runway proved to be the simplest alternative, requiring the least amount of ground support personnel and equipment. A balloon ascent and towed takeoff added performance to the aircraft by decreasing the amount of fuel needed at takeoff. The added performance did not outweigh the complexity as calculations progressed, so a conventional runway was used.

In summary, the three designs chosen for further investigation were a biplane configuration and two tandem-wing configurations. The three projects are called SHARP, Gryphon, and H.A.M.M.E.R., respectively.

AERODYNAMICS

From the sizing chart shown in Fig. 9, it is evident that in order to meet the constraints, the wing loading was limited to a range of 6 to 7 psf. In order to achieve wing loadings of this magnitude at altitude, the aerodynamic parameter M^2C_L had to be maximized[(1)]. To avoid Mach buffet, the upper limit on Mach number was approximately 0.7. With the upper limit for the cruise speed known, the maximum tolerable Reynolds number per unit chord was determined. The lower limit for the Reynolds number was set at approximately 300,000, where the drag rise with increasing Reynolds number increases sharply. Even before a Reynolds number of 300,000 is reached, it is clear from Fig. 10 that a decrease in Reynolds number results in an increase in drag. In order to balance the contradicting

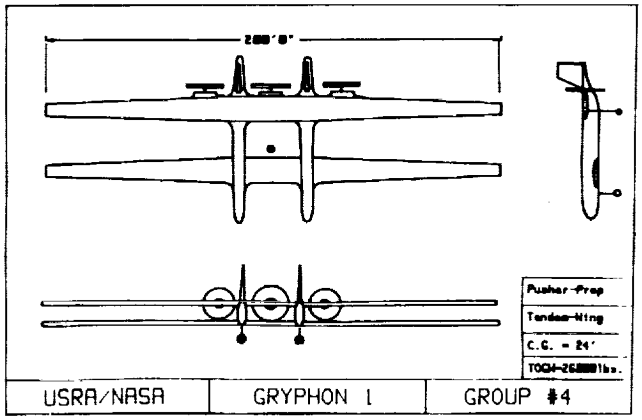

Fig. 7. Gryphon.

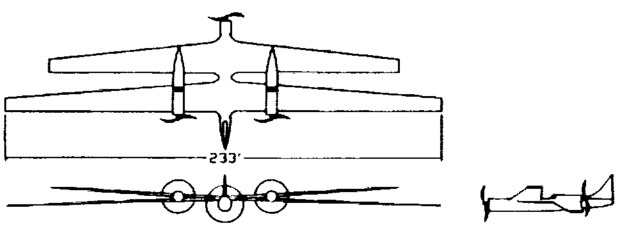

Fig. 8. H.A.M.M.E.R.

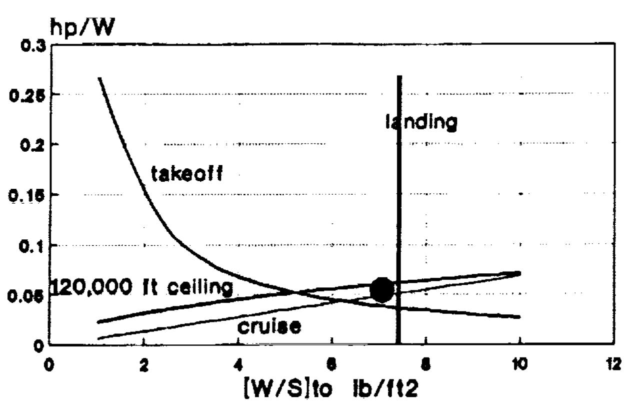

Fig. 9. Constraint diagram.

effects of Mach buffet and low-speed, low-Reynolds-number flight, a cruise Reynolds number of between 500,000 and 600,000, and a cruise Mach number of between 0.6 and 0.65 were selected by each group. The cruise Reynolds number was the design driver in the airfoil design.

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 airfoils tended to experience laminar separation bubbles and compressibility effects, which must be avoided.

Each group analyzed several different airfoils and modified them to best suit their needs. A computer code authored by Mark Drela called XFOIL was used to modify and analyze the airfoils[5]. The code was able to tailor the pressure distribution to reduce shocks and flow separation. 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].

The airfoils were modifications of the LA203A, the Eppler 1230, and the Lissaman 7769, which were chosen by the Gryphon, the SHARP, and the H.A.M.M.E.R. groups, respectively. Drag polars are displayed in Figs. 10 and 11. One disadvantage of the LA203A modification was an excessive pitching moment of -0.17/radian. The Gryphon group decided to compensate for the resulting trim drag by delaying the onset of separation with submerged vortex generators.

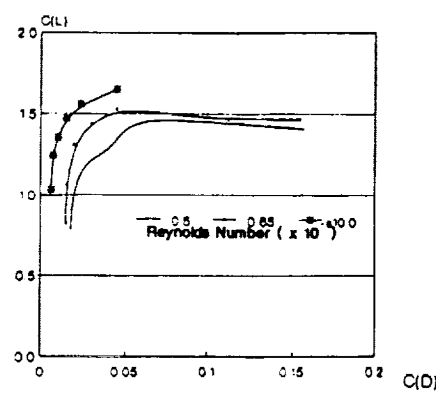

Fig. 10. LA203A drag polar.

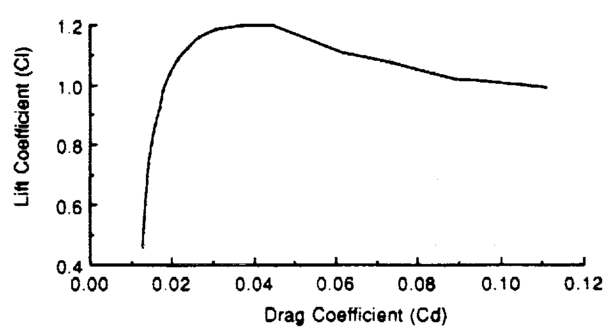

Fig. 11. Eppler 1230 drag polar.

Propeller Design

There were two main criteria for designing the propellers. The first and foremost was that the tip velocities cannot exceed the drag divergence Mach number. Since the air density is so low, the propellers lose their ability to transfer power at higher altitudes. This results in an increased diameter and RPM. For a conventional propeller configuration the necessary diameters are on the order of 25 to 35 ft. If a propfan propeller configuration is used, employing a blade sweep of 38°, the propeller diameters range from 16 to 20 ft with an increase in cruise efficiency of 3%.

All three groups opted for a pusher configuration to minimize flow disturbance over the wing. The disadvantage to a pusher configuration is that each time a blade passes through the wake of the wing it experiences a different loading than the freestream condition, which results in blade fatigue. This problem was eliminated by constructing the blades from composite materials. The SHARP group chose to fold the rearmost propeller on takeoff to increase ground clearance. The propeller will open during flight as a result of the centrifugal force produced by the engine rotation.

Performance

With low wing loadings, takeoff is not a problem. The takeoff distances were on the order of 1300 ft, and high lift devices in the form of flaps and slats were generally considered unnecessary. During landing, spoilers and other lift dump devices were employed. The landing distance for the SHARP project was 3537 ft. It should be noted that the SHARP's rearmost engine will be shut down for takeoff and landing. Because of this, the Gryphon and H.A.M.M.E.R. configurations take off and land in less than 75% of the SHARP value.

Using energy-state approximation methods, a minimum fuel-to-climb trajectory was found to minimize the weight of the aircraft without significantly increasing the time to climb. An ideal fuel-to-climb curve results in the aircraft climbing at stall speed, so the climb profile was designed to have a 10% margin over stall speed. The fuel consumed to reach 100,000 ft was 1200 lb. The climb velocity profile is shown in Fig. 12.

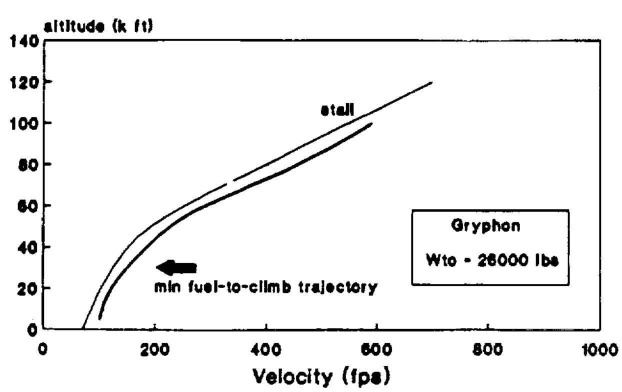

Fig. 12. Climb velocity profile.

The flight envelope for all three aircraft is similar. The aircraft are constrained by the 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. 13.

Figure 14 shows the variation of the power required as fuel is consumed for the cruise condition. It is clear that the aircraft is flying within its power requirements at all times.

PROPULSION SYSTEM

The powerplant for this aircraft must be able to operate with a low specific air consumption. The 6000-mile range requirement necessitates that the powerplant have a low specific fuel consumption. 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 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.

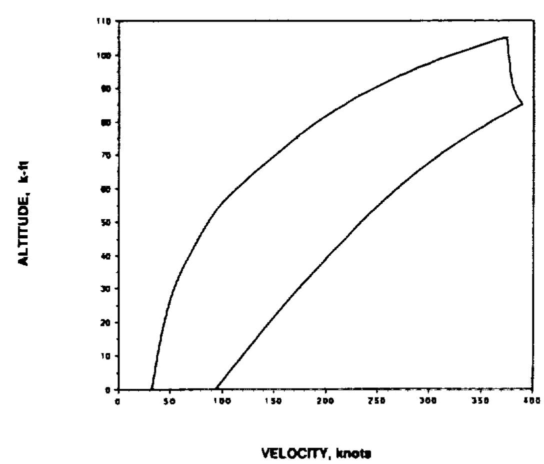

Fig. 13. Flight envelope.

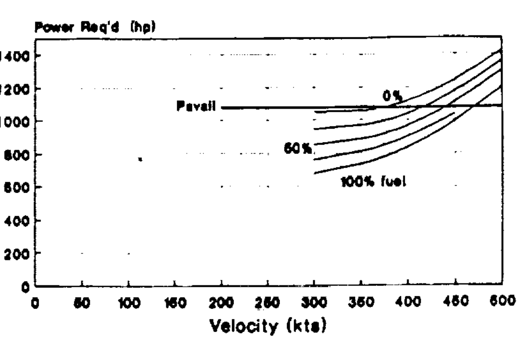

Fig. 14. Variation of power required.

Powerplant Selection

The driving constraint for engine selection is air consumption at altitude. The air consumption must be low for the engine to produce power at altitude. Figure 15 shows typical specific air consumption values for the engines examined. The second constraint is propulsion system weight, which must be kept as low as possible. Figures 16 and 17 show typical specific fuel consumption and specific weight values for the engines examined.

The subsonic cruise velocity combined with the high specific air consumption of turbojet and turbofan engines make it impossible for them to produce any meaningful thrust at altitude. 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 specific air consumption, specific fuel consumption, and specific weight.

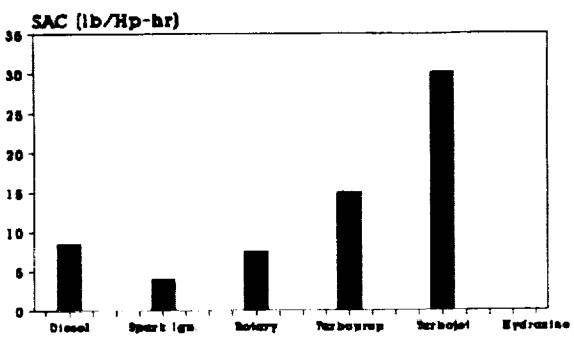

Fig. 15. Specific air consumption.

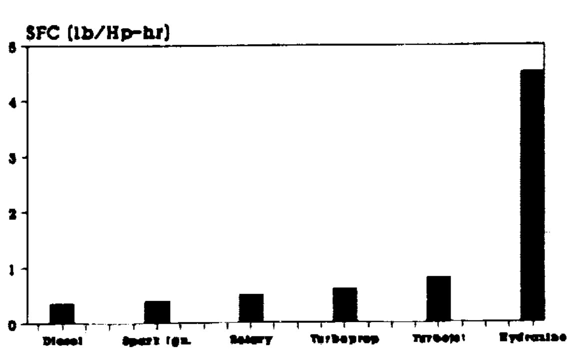

Fig. 16. Specific fuel consumption.

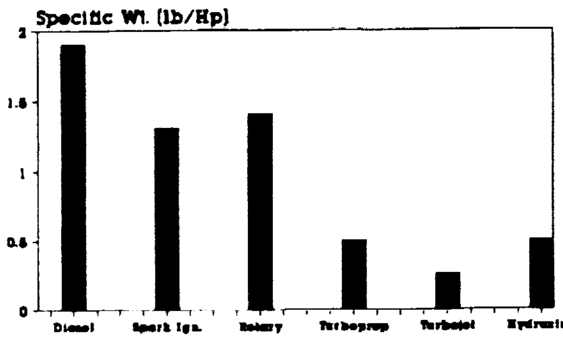

Fig. 17. Specific weight.

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 Teledyne Continental GTSIOL 550 engine with three stages of turbocharging now in the preliminary design stages was chosen for this project. At altitude, the engine produces 400 hp with a specific fuel consumption of 0.45 lb/hp/hr. The overall dimensions are 33.5 inches high by 42.5 inches wide by 42.64 inches long. The total weight with the turbocharged system is 1900 lb[7].

Engine Cooling

Cooling the engine at altitudes above 80,000 ft is a design challenge because of the low air density. The cooling techniques investigated are (1) using the fuel as a coolant with the wings acting as radiators; (2) liquid cooling with conventional radiators; (3) using oil coolant radiators; (4) use the fuel tanks as heat sinks; and (5) recycling heat in a steam turbine. In reality, none of these techniques were able to reject enough heat and still maintain a reasonable volume. A combination of two or more of the techniques is necessary.

WEIGHTS AND STRUCTURES

A typical weight breakdown is shown in Fig. 18. The takeoff gross weight varied from 25,000 lb to 30,000 lb depending on the mission and configuration. The aircraft structural analysis was constrained by the gust loading as shown in Fig. 19. Typical wing deflections are shown in Fig. 20.

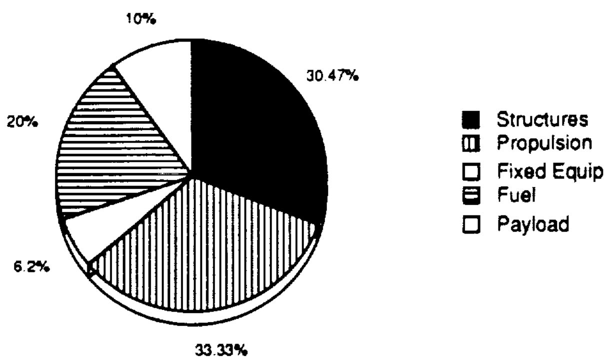

Fig. 18. Weight breakdown at takeoff = 30,000 lb.

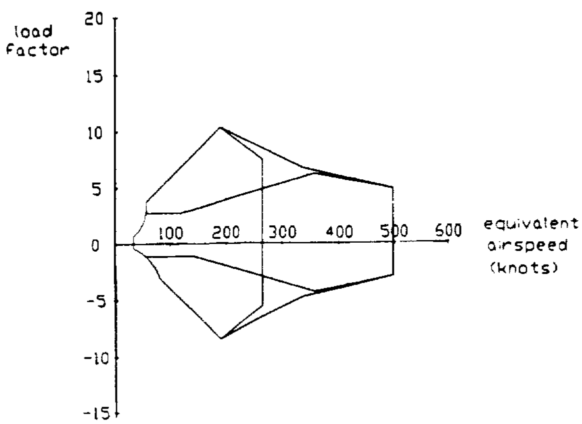

Fig. 19. Combined V-n diagram.

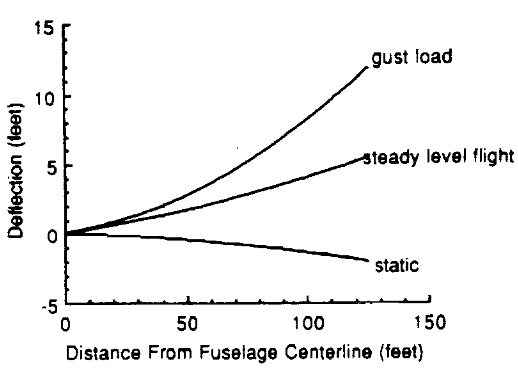

Fig. 20. Wing deflections.

Material Selection

To meet the wing loading requirements dictated earlier and still maintain the necessary strength, the wings were designed with composite materials and averaged a wing weight per unit area of 1.2 lb/ft^2. This is an attainable goal since both the *Daedulus* and the *Voyager* aircraft had lower wing weight per unit area. The material selection criteria are (1) high strength, (2) corrosion resistance, and (3) low density. Some of the materials considered and their properties are listed in Table 1.

TABLE 1. Material Properties Comparison.

Material	Young's Modulus (psi)	Tensile Strength (psi)	Density (lb/in^3)
Steel	$30(10)^6$	$110(10)^3$	0.278
Aluminum	$10(10)^6$	$61(10)^3$	0.101
Titanium	$16(10)^6$	$141(10)^3$	0.160
Gr/PAI	$10(10)^6$	*	0.056
Gr/epoxy	$8.0(10)^6$	$70(10)^3$	0.053
B/epoxy	$9.6(10)^6$	$85(10)^3$	0.068
Kevlar	$18(10)^6$	$525(10)^3$	0.052
Spectra	$25(10)^6$	$435(10)^3$	0.035

*Compressive strength = $95(10)^3$

Kevlar has the highest strength, but Spectra has the highest strength-to-weight ratio. Graphite Polyamide-imide acts best in compression. On the other hand, Graphite/Epoxy costs less than all three. H.A.M.M.E.R. was primarily designed with Graphite/Epoxy. Gryphon chose a combination of Graphite/Epoxy and Graphite Polyamide-imide, and SHARP chose Kevlar and Spectra 1000. Manufacturing with Spectra 1000 will prove to be expensive since it has not been used extensively.

MANNED VS. UNMANNED

Manned flight would be preferred by the stratospheric scientists since the pilot could monitor the aircraft rather than relying on data links for every desired action. Also, many countries may not allow an unmanned aircraft of this magnitude within their airspace. To put a man in the cockpit greatly increases the cost, complexity, and weight of the aircraft. The longer missions are on the order of 18 hours. It may not be reasonable to expect a pilot to remain in a space suit under cramped conditions for such a long period of time. For these reasons, all three projects chose to design an unmanned aircraft with an optional manned module that could be used for shorter flights and flights over populated areas.

COST

The total life cycle cost for these aircraft is $181 million. This number includes RDT&E, acquisition, operations, and disposal. Figure 21 shows the percent breakdown of life cycle cost. If two aircraft are built, each aircraft will cost $89.4 million over the course of its life. From the figure, it is apparent that the highest percentage of cost is attributed to RDT&E. The only way this number can be reduced is to postpone building this aircraft.

These figures were checked using a program produced by David Hall Consulting, under contract with NASA[8]. The results indicated a RDT&E cost of $191 million and an acquisition cost of $51 million.

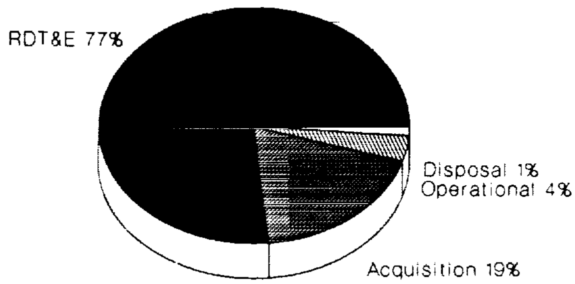

Fig. 21. Life cycle costs.

SUMMARY OF RESULTS

Table 2 contains a selective summary of the results obtained for each of the three configurations.

TABLE 2. Summary of Results

	Gryphon	SHARP	H.A.M.M.E.R.
Span - fore	202	250	186
(ft) - aft	202	250	233
MAC - fore	8.4	8.4	12.4
(ft) - aft	8.4	8.4	11.6
Aspect Ratio - fore	14	31.25	15.1
- aft	(eff)	31.25	20.1
Wing Area (ft^2)	3400	4000	5000
Empty Weight (lb)	17200	21000	16800
C.g. at take-off (ft)*	25.7	28.3	34.2
Gross Take-off Weight (lb)	26000	30000	26000
Time to Climb (hrs)†	1.7	4.1	2.5

*measured from nose
†mission one

1. All the design requirements were met except for the excursion to 120,000 ft in the fourth mission. Since the absolute ceilings of these aircraft were between 110,000 and 115,000 ft, it was determined that it would be unreasonable to expect a first-generation aircraft to zoom to altitudes of 120,000 ft. Perhaps after sufficient flight testing, a rocket-assisted zoom could be achieved.

2. The hangar requirements could be met by folding or removing the wings. A study should be done to determine if it would be cheaper over the life of the aircraft to build new hangars for housing.

3. Wind tunnel testing must be done on all three configurations. Interference effects caused by joining the wings at the tips for the SHARP configuration are still uncertain. The H.A.M.M.E.R. and Gryphon configurations positioned their wings for minimum drag. With wind tunnel testing, it can be determined if the drag is less than that for a conventional monoplane.

4. The exact combination of cooling methods to achieve the desired heat rejection deserves further research.

5. With RDT&E costs on the order of 140 to 190 million dollars it is difficult to determine who could finance this aircraft.

6. A high-altitude, ozone sampling platform configured with one or more three-stage turbocharged internal combustion engines is feasible.

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