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Design of a Vehicle Based System to Prevent Ozone Loss

by the

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ABSTRACT

Reduced quantities of ozone in the atmosphere allow greater levels of ultraviolet light (UV) radiation to reach the earth's surface. This is known to cause skin cancer and mutations. Chlorine liberated from Chlorofluorocarbons (CFCs) and natural sources initiate the destruction of stratospheric ozone through a free radical chain reaction.

The project goals are to 1) understand the processes which contribute to stratospheric ozone loss, 2) examine ways to prevent ozone loss, and 3) design a vehicle-based system to carry out the prevention scheme. The 1992/1993 design objectives were to accomplish the first two goals and define the requirements for an implementation vehicle to be designed in detail starting next year.

Many different ozone intervention schemes have been proposed though few have been researched and none have been tested. A scheme proposed by R.J. Cicerone, Scott Elliot and R.P.Turco late in 1991 was selected because of its research support and economic feasibility. This scheme uses hydrocarbon injected into the Antarctic ozone hole to form stable compounds with free chlorine, thus reducing ozone depletion. Because most polar ozone depletion takes place during a 3-4 week period each year, the hydrocarbon must be injected during this time window.

A study of the hydrocarbon injection requirements determined that 100 aircraft traveling Mach 2.4 at a maximum altitude of 66,000 ft. would provide the most economic approach to preventing ozone loss. Each aircraft would require an 8,000 nm. range and be able to carry 35,000 lbs. of propane. The propane would be stored in a three-tank high pressure system. Missions would be based from airport regions located in South America and Australia.

To best provide the requirements of mission analysis, an aircraft with L/Dcruise=10.5, SFC=0.65 (the faculty advisor suggested that this number is too low) and a 250,000 lb TOGW was selected as a baseline. Modularity and multi-role functionality were selected to be key design features. Modularity provides ease of turnaround for the down-time critical mission. Multi-role functionality allows the aircraft to be used beyond its design mission, perhaps as an HSCT or for high altitude research.

ACKNOWLEDGMENTS

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1. INTRODUCTION

The discovery of an Antarctic "ozone hole" in the late 1970s led to a flurry of scientific research to determine its causes and impacts. Despite a better understanding of the processes involved, ozone loss continues and, if current predictions are correct, will continue far into the future. The potential effects of continued ozone loss on the global biosphere are devastating and this threat has created the need for methods to replace lost ozone or interrupt the processes which cause ozone loss.

The overall goal of this three year NASA/USRA Advance Design Program project is to respond to the threat of ozone loss. This is accomplished by 1) defining the processes which contribute to stratospheric ozone loss, 2) examine possible prevention schemes and determining the best scheme to prevent ozone loss, and 3) designing a vehicle-based system to carry out the prevention scheme. The 1992/1993 design objectives were to accomplish the first two overall objectives and to define the implementation vehicle's baseline requirements.

Definition of the processes which contribute to stratospheric ozone loss is the purpose of the next chapter. The destruction of ozone by free-radical chain reactions with chlorine is described as well as the chlorine sources. In addition, the catalytic effects of the Polar Stratospheric Clouds (PSCs) and the importance of the polar vortex are described.

Chapter three describes the various intervention schemes which have been proposed to combat ozone loss. From analysis of the proposed schemes a hydrocarbon injection scheme proposed by R.J. Cicerone et al. is selected due to its research base and relative feasibility (Cicerone, 1991).

Using the requirements of the hydrocarbon injection scheme selected in Chapter 3, mission analysis defines the type of vehicle to be used, the required range, payload, general mission bases and the necessary injection plume area. These and other mission parameters are discussed in Chapter 4. Note that mission analysis does not select the vehicle flight regime (subsonic or supersonic), but the analysis is used later in the selection process of the flight speed.

Chapter 5 presents the analysis used to obtain the injection plume area. Turbulent mixing analysis is used to substantiate estimates of injection plume area made during the mission analysis. Not only does this discussion lend credence to the mission analysis, but it also defines the required mass flow and exit nozzle area for the hydrocarbon storage system.

Chapter 6 provides a refinement to mission analysis by discussing the needs and problems associated with finding a mission base. The number of vehicle sorties per day and per base is defined as well as the problems associated with hydrocarbon storage in existing bases.

Although a discussion of the ground storage of hydrocarbons is provided in Chapter 6, a discussion of the vehicle-based hydrocarbon storage system is needed. This is the subject of Chapter 7. The tank size and construction requirements are defined as well as details for safety. In addition, an inflight storage tank support structure is designed with emphasis on tank mobility and reducing torsional and bending moments on the tanks.

Using mission analysis parameters and vehicle storage system data as well as inputs from the computer design tool ACSYNT, Chapter 8 details the vehicle baseline requirements and defines key design features. Among the desired features of the vehicle system are modularity, for ease of repair, and multi-role functionality to provide multiple uses for the design vehicle.

Chapter 9 outlines vehicle propulsive concerns. The chapter discusses the possibility of using additional amounts of the injected hydrocarbon as a propulsive fuel, as well as the possibility and availability of providing for the design vehicle's propulsive needs. Finally, Chapter 10 summarizes the analysis and provides the vehicle baseline.

2 OZONE

Ozone (O_3) is found mainly in the stratosphere with 90% of its mass located between 10 and 50 km. (Henderson *et al.*, p. 1111) The highest concentration of ozone can be found between 15 and 35 km, and is called the 'ozone layer.' (Mims, p. 34) It is a thin layer which at sea level conditions would be 3 mm thick. This layer is very important to the living organisms on earth. It absorbs harmful ultraviolet (UV) radiation (of wavelength from 220 to 290 nm), which causes skin cancer, cataract, and possibly immune system deficiencies. (Mims, p. 34)

2.1 Ozone Chemistry

Ozone is formed by a natural photochemical process when UV radiation from the sun splits molecules of oxygen into the two oxygen atoms from which they are formed. The free O atoms soon react with O_2 molecules to form O_3 (Mims, p. 34):

$$O_2 \xrightarrow{h\nu} O + O \quad (129 < \lambda < 240 \text{ nm}) \tag{2.1}$$

$$O_2 + O + M \rightarrow O_3 + M \tag{2.2}$$

where M represents "bath" species such as O_2 or N_2.

Natural ozone loss occurs by absorbing harmful ultraviolet radiation of longer wave length ($220< \lambda <320$ nm) as in the below process:

$$O_3 \xrightarrow{h\nu} O_2 + O \qquad (220< \lambda <320 \text{ nm}) \tag{2.3}$$

Obviously, a natural ozone destruction process is required and is in the form of the following reaction:

$$O + O_3 \rightarrow 2O_2 \tag{2.4}$$

The reaction 2.3 is the most important process in which most of harmful UV is prevented from reaching the earth surface. Other important natural ozone destruction reactions are of the form:

$$X + O_3 \rightarrow XO + O_2 \tag{2.5}$$

$$XO + O \rightarrow X + O_2 \tag{2.6}$$

$$\text{net: } O + O_3 \rightarrow 2O_2$$

where X is a radical which is OH, NO, Cl, or Br (Fig. 2.1). (Henderson *et al.*, p. 1111)

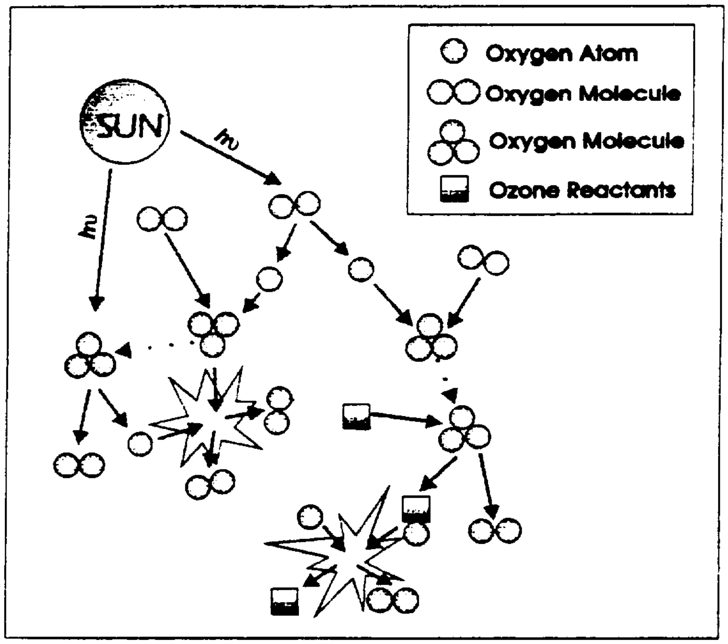


Figure 2.1 A simplified view of how ozone molecules can be formed and destroyed. Source: Forrest Mims, III, Ozone Layer, *Science PROBE!,* Nov. 1992.

From reactions (2.1) through (2.6) it seems that ozone is being destroyed catalytically by oxygen atoms and radicals (X). However, there are two reasons that the catalytic ozone destruction does not occur in natural stratospheric conditions. First, O is much less abundant than O_3 in the middle and lower stratosphere. Second, the reactive species HO_2, NO_2, ClO, and BrO, from which the radicals are formed, are generally not abundant. (Henderson *et al.*, 1991) This calls for additional explanations for the global ozone depletion, especially over Antarctica.

2.2 Polar Ozone Depletion

There have been many reports over the last decade that, during austral spring (autumn in North Hemisphere), the ozone concentration over Antarctica dropped to the point of almost total depletion. This decrease in stratospheric ozone is known as the "ozone hole." In 1987 and 1989, the ozone hole covered almost twice the area of Antarctic continent. (Ahrerns, p. 44) According to some researchers, severe ozone depletion may occur over the North Pole within the decade. (Vogel, p. 32)

Many scientists believe that man-made chlorofluorocarbons (CFCs) are the main cause of the depletion of the ozone layer. CFCs are used in refrigerators, air-conditioners and fast-food cartons as well as in solvents and cleaners. Two of the most commercially important CFCs are $CFCl_3$ (CFC-11) and CF_2Cl_2 (CFC-12). CFC-11 and CFC-12 together contribute to over 67% of the chlorine in the atmosphere accounting for the release rates and ozone depletion potentials. (Wayne) These chemicals are inert under most circumstances but when they slowly rise to the stratosphere and are exposed to large amount of UV radiation, they release chlorine atoms by the following processes (Fig. 2.2):

$$CF_2Cl_2 \xrightarrow{h\nu} CF_2Cl + Cl \quad (\lambda < 220\text{ nm}) \tag{2.7}$$

$$CFCl_3 \xrightarrow{h\nu} CFCl_2 + Cl \quad (\lambda < 265\text{ nm}) \tag{2.8}$$

The free chlorine then reacts with ozone by the following processes:

$$O_3 + Cl \rightarrow O_2 + ClO \tag{2.9}$$

$$O + ClO \rightarrow O_2 + Cl \tag{2.10}$$

In the above reactions Cl reacts as a catalyst, which can go on to destroy millions of ozone molecules before it combines into a stable "reservoir".

2.3 Chlorine Reservoirs

Most of the atomic chlorine (Cl) released from CFCs and brought to the South Pole by stratospheric winds eventually react with hydrogen, nitrogen, and oxygen molecules to remain as reservoirs. (Kawa, et al.) Reservoirs are "stable compounds in which ozone-depleting species can be tied-up." (*Stratospheric Ozone*, 1987) The formations of the primary chlorine reservoirs involve the following reactions (Fig. 2.2):

$$Cl + CH_4 \rightarrow CH_3 + HCl \tag{2.11}$$

$$ClO + NO_2 + M \rightarrow ClONO_2 + M \tag{2.12}$$

where M is a catalytic molecule.

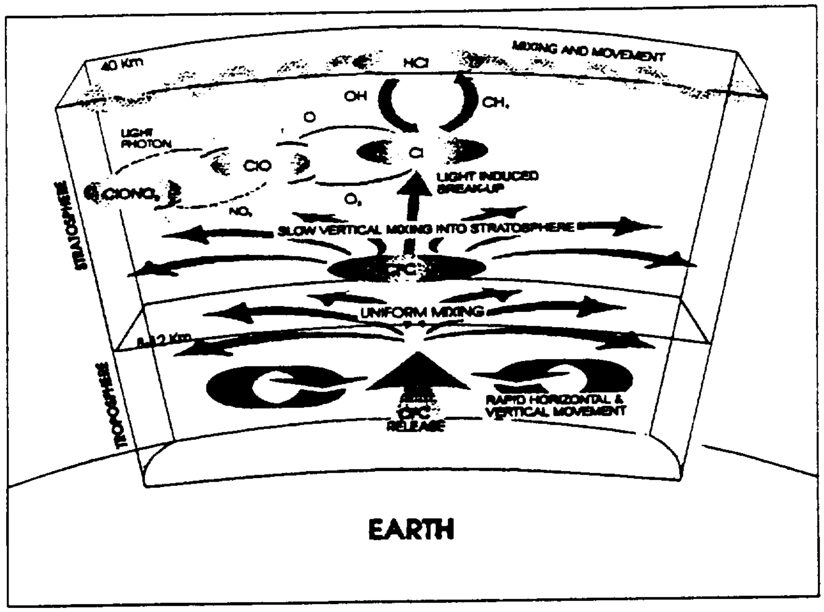


Figure 2.2 The Chlorine Connection. Source: Ashley Steven, "Ozone Drone", Popular Science, July 1992.

2.4 Polar Stratospheric Clouds (PSCs)

During September and October, lower latitude air with high concentration of CFCs is transported poleward by stratospheric winds, and is trapped inside a belt of stratospheric winds called the Polar Vortex (Fig. 2.3). This polar vortex isolates the cold Antarctic stratospheric air from the warmer air of the middle latitudes. The vortex starts to form during the polar night (around June) and lasts until late September. (Wayne, 1991) During the Antarctic winter, temperatures inside the vortex often fall well below 200 K which allows for the formation of polar stratospheric clouds (PSCs). There are two types of PSCs. Type I PSCs appear to form at about 197 K and consist of condensed HNO_3 and H_2O. The larger type II PSCs appear to form below 187 K and consist primarily of H_2O (Henderson *et al.*, p. 1113). Eventually, in the spring, the vortex breaks down due to enhanced planetary wave activity. (Henderson *et al.*, p. 1113)

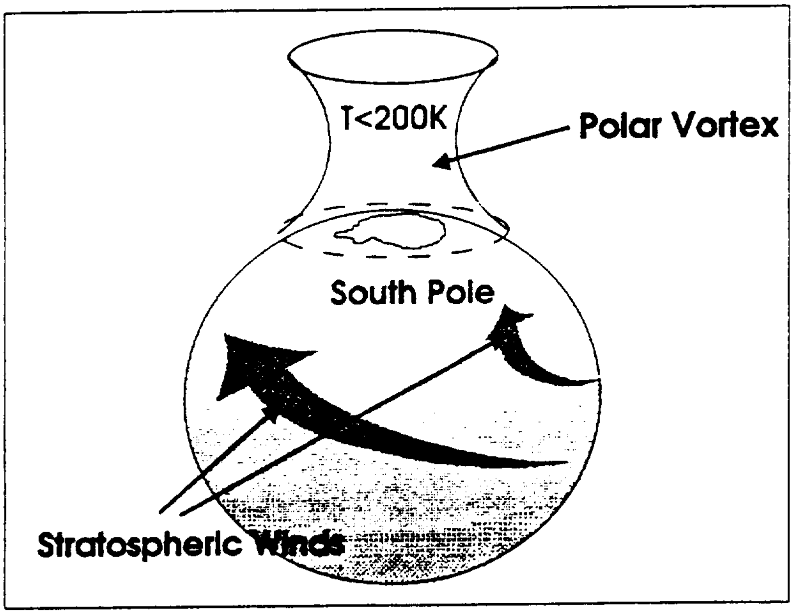


Figure 2.3. Model of the polar vortex.

2.5 Heterogeneous Reactions

It has been speculated that the heterogeneous reactions occurring on the surfaces of PSCs are the possible main chemistry of the catalytic ozone destruction. Some of the important heterogeneous reactions are:

$$HCl + ClONO_2 \rightarrow HNO_3 + Cl_2 \quad (2.13)$$

$$HCl + N_2O_5 \rightarrow ClNO_2 + HNO_3 \quad (2.14)$$

$$H_2O + N_2O_5 \rightarrow 2HNO_3 \quad (2.15)$$

$$H_2O + ClNO_3 \rightarrow HOCl + HNO_3 \quad (2.16)$$

As the temperature falls below about 197 K, reaction (2.13) and (2.14) convert the reservoir $ClNO_3$ and HCl species to Cl_2, $ClNO_2$, and HNO_3. These heterogeneous reactions are likely to first occur as type I PSCs form. With further cooling within the vortex, type II PSCs form and any remaining N_2O_5 and $ClNO_3$ react as in reaction (2.13), (2.14), (2.15) and (2.16). (Henderson *et al.*, p. 1114) These newly formed chemicals ($ClNO_2$, Cl_2, and HOCl) are rapidly converted to ClO via photolysis:

$$HOCl \xrightarrow{h\nu} OH + Cl \quad (2.17)$$

$$Cl_2 \xrightarrow{h\nu} 2Cl \tag{2.18}$$

$$ClNO_2 \xrightarrow{h\nu} ClO + NO \tag{2.19}$$

These Cls and ClOs destroy O_3 catalytically by reaction (2.9) and (2.10), and the ozone hole is formed.

2.6 The Future

The severe ozone depletion at the South Pole is not a local problem; it is a global problem. In 1989, in southern Australia, UV radiation levels climbed by 20% for a few days right after the breakup of the polar vortex, and the average increase for the month of December was 14%. (Flamsteed, p. 33) According to David A. Wirth, even if CFC and Halon production is phased out within 5-7 years, the environment could take up to a century to recover due to their long atmospheric lifetimes. Even then, the existing stock of refrigerators, air conditioners, insulation, and other repositories would still carry the potential to continue to destroy the ozone layer.

In 1987, many industrial nations met in Montreal and agreed on reducing their emission of CFCs 50% by 1998 and stopping the production of CFCs by the year 2000. But these proposals do not have much effect on reducing the atmospheric chlorine level. Fig. 2.4 is a good example of a future prediction on CFC levels. It shows that the global consumption of CFCs would have to be reduced by 85% immediately in order to level off at roughly twice the concentrations of the 1970s, 95% reduction over the next decade would reverse the chlorine buildup, and 95% immediate reduction would drop the curve in a shorter period of time. (Hively, p. 223)

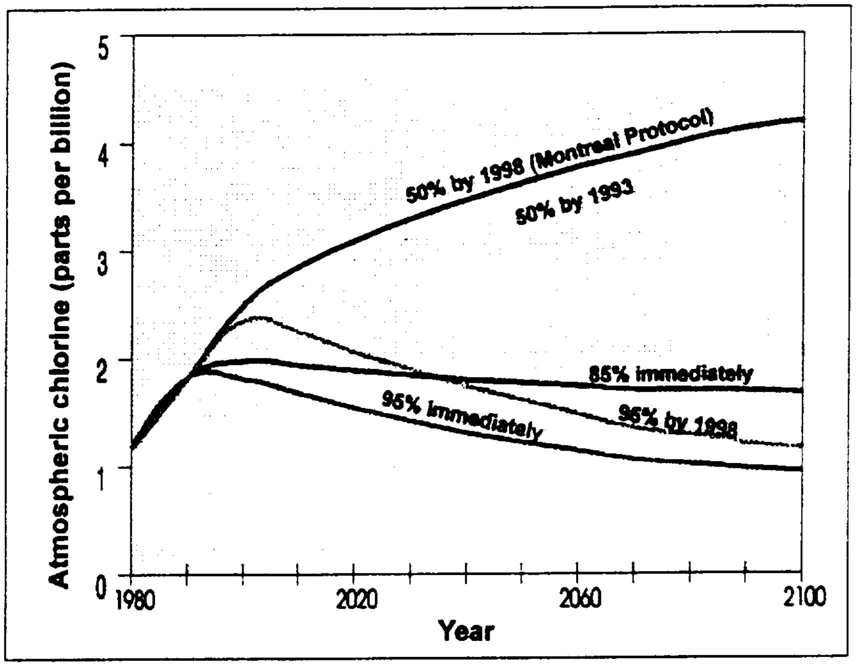


Figure 2.4 Proposed CFC reduction: Source: Hively, W., "Science Oberver," American Scientist, May 1989.

On November 26, 1992, representatives from 87 countries met in Copenhagen, Denmark, to move up the phase-out of CFC's from the year 2000 to Jan. 1, 1996 and halons to Jan. 1, 1994. HydroChloroFluoroCarbons (HCFC's - CFCs replacement) which still deplete ozone, but not as much as the chemicals they replace, are now to be eliminated in stages starting in the year 2004 and ending in 2030. (The New York Times, Nov. 26, 1992)

These proposals will help to reduce the ozone depletion, but they lack the inclusion of the ozone destroying methyl bromide and a more affirmative global coordination for these plans to be fully complied with.

3. ACTIVE INTERVENTION SCHEMES

Few suggestions have been proposed regarding active intervention schemes for preventing or reducing stratospheric ozone loss. This is due to the lack of understanding regarding the atmospheric processes and dynamics within the ozone hole. Computer simulations have been used to test various hypotheses, but the complexity and chaotic nature of the atmosphere makes it impossible to accurately model all aspects of the atmospheric dynamics. In spite of the difficulties involved several active intervention schemes have been presented.

3.1 *in situ* Ozone Generation

Ozone can be generated by photolysis or electrical discharge. Photolysis could emulate the solar dissociation of oxygen by irradiation with Ultra-Violet energy. Ten gigawatts of power must be provided continuously for one year to generate one percent of the total global ozone (Kay, 1992). A value higher than ten gigawatts would be needed due to energy absorbed by unintended non-ozone forming oxygen reactions.

A several kilovolt electrical discharge lasting about a nanosecond could also be used to generate ozone. The maximum yield for the discharge would be 150 grams of ozone per kilowatt-hour. Thirty gigawatts of power supplied continuously for one year would produce one percent of the total global ozone. The following are the general ozone generation reactions:

$$e^- + O_2 \rightarrow 2O + e^- \tag{3.1}$$

$$O + O_2 + M \rightarrow O_3 + M \tag{3.2}$$

where M is a catalyst.

A major advantage of this method is that there is no other substance injected into the atmosphere which may cause unintended reactions. The major disadvantage of this procedure is the enormous amount of power required. The average nuclear power plant only produces 3 gigawatts of power as compared to the required amounts of 10 and 30 gigawatts. This scheme is only temporary, and in fact pointless as long as chlorine is around.

3.2 PSC Removal

Polar Stratospheric Clouds (PSCs) provide the platform for the reactions to occur. Dissipation of the PSCs would eliminate the heterogeneous reaction platforms. High power microwave radiation could be used to dissipate the PSCs. According to a Texas A&M study, 5.5 gigawatts of microwave power at 22 GHz would be needed to evaporate the PSCs (Kay, 1992). This could be accomplished from a low earth orbit satellite in a period of 38.8 days. The advantages to this procedure are no chemical additions to the atmosphere and limited radiation exposure during PSC evaporation in the spring.

A much greater understanding of PSC formation and composition is needed before this method can be implemented. The microwave radiation may cause damage to other regions of the upper atmosphere. The troposphere and the surface of the earth may also be exposed to this high power microwave radiation. This would raise the temperature of the troposphere and thus upset the thermal balance of the environment. Radiation reaching the earth could mutate or kill existing life forms.

3.3 NOx Injection

Unstable chlorine atoms could be transformed into stable reservoirs by introducing active nitrogen elements. The following equation is an example of one of the NO_2 reactions:

$$ClO + NO_2 + M \rightarrow ClONO_2 + M \qquad (3.3)$$

This reaction represents the stabilization of chlorine by formation of an inert molecule. By stabilizing the chlorine reservoirs in the ozone hole, the cyclic destruction of ozone is stopped.

An advantage to this procedure is that high speed air-breathing aircraft could be used to generate the NOx. These planes could overcome the lack of mixing within the ozone hole during the short injection time frame window.

A major problem with this scheme is the unpredictability and lack of data regarding NOx injection. NOx destroys ozone, but it does not have the cyclic destruction characteristics of chlorine. Another disadvantage to this method is that excess NOx could spread and cause global damage. Because of the PSC process that frees stable chlorine reservoirs, this is only a temporary solution.

3.4 Stratus Cloud Formation

All of the other suggested methods attempt to protect or generate the UV shielding ozone layer in the stratosphere. Stratus cloud formation provides a temporary UV shield during periods of diminished ozone concentrations.

Water particles and energy are the only ingredients needed to form stratus clouds. Thus, no harmful chemical injection reactions can result. However, the major obstacle is generating the stratus clouds. There is no known stratus cloud generation scheme. Another drawback is that the clouds could dissipate quickly in the atmosphere.

3.5 Chemical Adsorbent

Zeolite sieves could be injected into the atmosphere to remove or filter ozone destroying chlorine. A zeolite sieve is a natural or man-made mineral with extensive internal molecular pores which act as a sponge to soak up target elements. There are existing industrial zeolites which filter Freon-12 and other chlorinated molecules. Sieves range in size from fine powders to large wafers depending on the application.

A major advantage is that this procedure is permanent - the chlorine is removed from the atmosphere. Another advantage is that a continuous world-wide filter process could be utilized.

The selectivity, or ability to absorb only specific molecules, of these sieves must be researched and fully understood before such a procedure could be implemented. The drawback is the unpredictable sieve-atmospheric reactions. The destruction of ozone and adsorption of needed elements may outweigh the chlorine scavenging results. Zeolite sieve recovery, disposal, and recycle procedures may be difficult and expensive. Another disadvantage is that UV radiation may transform the sieves into a useless destructive forms.

3.6 Alkali Salt Injection

Alkali metals/salts could be used to stabilize the chlorine reservoirs in the PSCs. This would neutralize the PSCs capability to attack ozone. The following is an example reaction of an alkali salt, NaOH, within the PSCs:

$$NaOH + HCl \rightarrow NaCl + H_2O \tag{3.4}$$

As a chlorine reservoir, HCl is unstable in PSCs and will contribute to ozone loss. The stability of NaCl determines the success of this method. If NaCl does not react to free the chlorine then it will eventually settle out of the atmosphere (Kay,1991).

This is the only other permanent method that removes chlorine from the atmosphere. Another advantage is that there is a three month injection period starting with vortex formation and ending with PSC evaporation. This would allow plenty of time to inject the estimated 60,000 tons of alkali salt (Kay, 1991).

A major disadvantage is the unpredictable adverse consequences resulting from the injection. There is little test data predicting the actual atmospheric results.

The lack of experimental test results make all of the proceeding intervention schemes too risky at this time. The unpredictable adverse consequences may cause great environmental damage. The two permanent chlorine removal schemes are most inviting but must be examined in depth before an implementation procedure can be suggested.

3.7 Cicerone Method

R.J. Cicerone, Scott Elliott and R.P. Turco proposed a hydrocarbon injection system for attacking the free chlorine in the polar vortex (Cicerone *et al.*, 1991). Though not tested in the atmosphere, they determine the plausibility of his method using a simplified computer model. The results of their simulations make the hydrocarbon injection scheme the most promising.

3.7.1 Concept

Cicerone's basic concept is to stabilize the active chlorine elements within the PSCs so that ozone will not be deteriorated. PSCs dehydrate and denitrificate the polar air so that stable chlorine reservoirs break up and attack the ozone. Injecting certain hydrocarbons in the ozone hole during PSC evaporation forms stable chlorine reservoirs. The following are some representative reactions:

$$Cl + CxHy \rightarrow HCl + C_xH_y \quad \text{(general reaction)} \qquad (3.5)$$

$$Cl + C_3H_8 \rightarrow HCl + C_3H_7 \quad \text{(propane reaction)} \qquad (3.6)$$

The ratio of hydrocarbon to active chlorine is 1 to 1 (it takes one hydrocarbon molecule to stabilize one active chlorine species). The stability of HCl determines the effectiveness of this scheme. The presence of PSCs will free chlorine to attack the ozone.

3.7.2 Model

The zero-D computer simulation of the hydrocarbon injection is based on current polar stratospheric data coupled with Antarctic ozone chemistry. The calculations consisted of 130 stratospheric gas-phase photochemistry equations, all of the heterogeneous PSC equations, gas phase reaction rates, and ice surface reaction efficiencies based on laboratory experiments. The baseline input parameters are:

Parameter	Baseline
Altitude	15 km
Latitude	80 S
Calculations begun	8 August
Sunrise	15 August
PSC evaporation	1 September

Ethane and propane were tested for the simulated hydrocarbon injection. Two simulation versions were utilized. The versions differed by the amount of trace gases in the Antarctic stratosphere remaining after the polar night. If fewer than 1.8 ppb of any injected hydrocarbon were used in either version, the ozone problem may increase.

3.7.3 Complications

Lacking extensive polar vortex data, many simplifying assumptions were incorporated in the simulation. These assumptions may cause many inaccuracies and may produce misleading results. However, this method produces the best available active intervention prediction. Furthermore, the Cicerone method is sensitive to an injection time frame, due to PSC dissipation which determines the stability of the chlorine reservoirs. The simulation is also sensitive to the concentration of hydrocarbon injected. If not enough ethane or propane is injected then the ozone loss may increase. It was recommended that a minimum of 3.6 ppb hydrocarbon be used to avert this problem (Cicerone).

3.7.4 Implementation Requirements

The Cicerone method is not a permanent solution because the stable chlorine reservoirs become unstable in the presence of PSCs. These PSCs form annually near the end of the polar night. Thus, an annual hydrocarbon injection must be performed until a safe permanent solution is found.

Propane was chosen over ethane as the injected hydrocarbon because of its advantageous chlorine scavenging abilities. Fifty thousand tons of propane is needed to obtain 3.6 ppb in the ozone hole. The delivery time constraint is a 3 week period beginning with PSC dissipation. There must be a uniform propane distribution throughout the volume of 1×10^8 km^3 for effective chlorine scavenging.

This will be difficult due to the lack of mixing within the polar vortex. The propane delivery system evolved is critical for a successful propane injection campaign.

4. MISSION ANALYSIS

4.1 Introduction

Several types of delivery systems were considered to implement the Cicerone method. These included missiles, lighter-than-air (LTA) vehicles and aircraft. An aircraft delivery system was selected as the most appropriate. The following highlights the criterion for that selection.

- Missiles have a low payload/gross weight ratio and debris problems. However, they could be used to fill up gaps left by aircraft.
- LTAs are very slow, are altitude restricted, have low payload capacity and are prone to weather damage.
- Aircraft are capable of high speeds, high payload capacity, are reusable but require more extensive ground support.

4.2 Mission Parameters

The driving factors for this mission are the sheer volume of the polar vortex, the delivery time frame, the mission range and the required hydrocarbon concentration. The polar vortex volume is recognized as a dynamic variable in this analysis, but assumed to be constant for simplicity. The delivery time frame is set to 3 weeks as specified by Cicerone, but may be raised to a month if need arises (Kay). The airport region of choice will determine the mission range.

One of the main parameters in the mission analysis is the concentration of the injected hydrocarbon. To realize the impact of this requirement, it is necessary to look at the delivery volume. It is 5 km in height and has a surface area of 20 million square km. That's a total of 100 million cubic km. Figure 4.2-1 depicts the polar vortex, with a cut-out section showing the proposed delivery scheme. This drawing is not to scale, as the vortex has a diameter one thousand times it's height. The delivery scheme must be able to inject the hydrocarbon achieving a concentration satisfying the Cicerone requirement.

The proposed delivery scheme is shown in Figure 1. It shows the layout of the injected hydrocarbon plumes, and how they overlap each other. The plumes are presented as having a finite radius with uniform mixing. This is only for demonstration purposes. The hydrocarbon has the greatest concentration at the center of a plume, with ever-decreasing concentration further away from the center. This is discussed further in the mixing analyzis section.

To achieve the overall concentration required by the Cicerone method, the plumes can be packed closer together or wider apart as desired. As proposed the plumes are arranged in three levels or stacks. A mixing analyzis showed that this pattern would satisfy the Cicerone scheme requirements. Arranging the plumes wider apart in two stacks was unattainable due to natural dispersion effects.

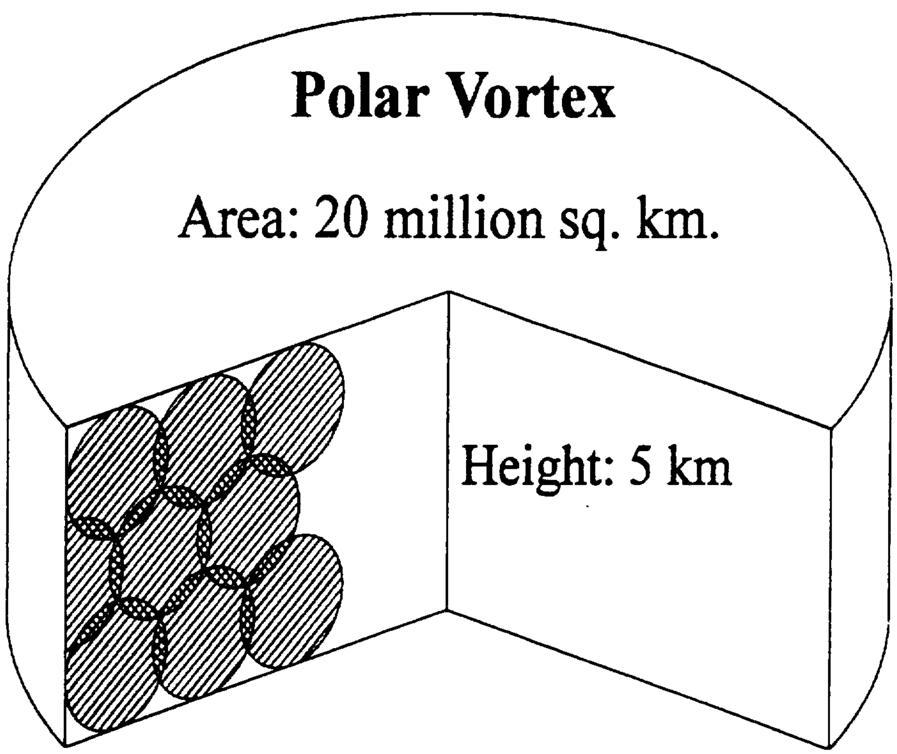


Figure 4.2-1. Depiction of the Polar Vortex geometry and the proposed delivery scheme.

4.3 Mission Model

To obtain the mission requirements, a mission model was established. This model splits the polar vortex volume into equal shares for each airport region. Figure 4.3-1 depicts the model representation for a single arbitrary mission when two airport regions are used. For three airports the vortex is divided into three parts.

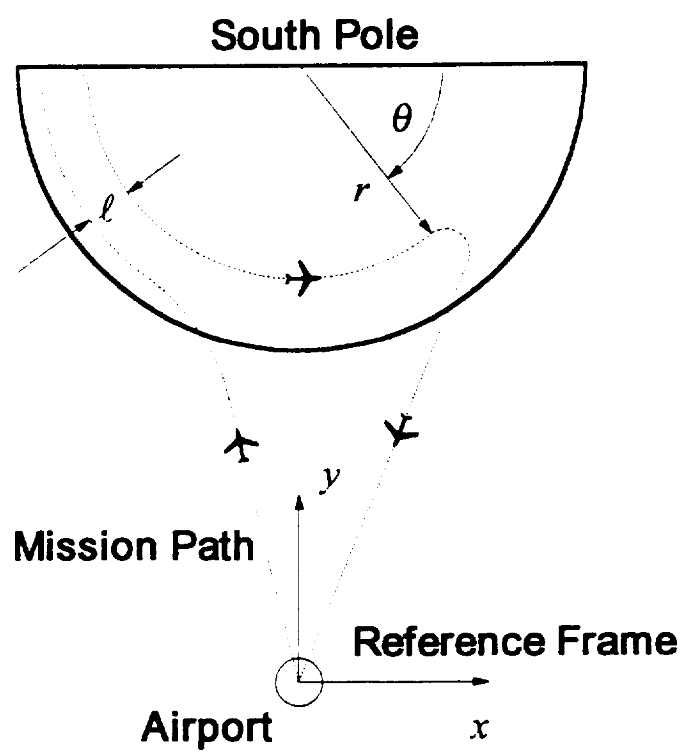


Figure 4.3-1. A schematic of the mission model.

The general mission flies out of the airport to the location where the last mission stopped, and starts to deliver payload flying in an arc circling the South Pole. It flies back when it is out of payload or if the distance to the airport equals the remaining range. Whenever a mission reaches the edge of the polar vortex, it turns around, reduces it's arc radius by ℓ and continues delivering payload. The location of a particular mission can be represented in either Cartesian or polar coordinates as indicated on the schematic. These locations can be calculated by using the following equations.

$$\theta = \tan^{-1}\left(\frac{APR - y}{x}\right)$$

$$r = PVR - k \cdot \ell$$

or

$$x = r\cos\theta$$

$$y = APR - r\sin\theta$$

where

APR	=	Distance between airport location and South Pole
PVR	=	The radius of the Polar Vortex
k	=	Counter for the number of arcs flown
ℓ	=	The distance between two flight paths

A FORTRAN program implementing this model was used. The location where a mission finished injecting it's payload is stored using Cartesian coordinates. The program then starts sweeping through the mission with a certain step size, keeping track of the Mission Range (*MR*) and Delivery Range (*DR*) being used. The equations used are:

$$\Delta\theta = Sign \cdot \frac{Step}{r}$$

$$DR = DR + \Delta\theta \cdot r$$

$$MR = \sqrt{y^2 + x^2} + DR + \sqrt{(r\cos\theta)^2 + (APR - r\sin\theta)^2}$$

Where

$Sign$	=	Positive when clockwise and negative when anti-clockwise
Step	=	A certain step size

The code sweeps through the whole volume and calculates the number of missions and the Total Mission Range (*TMR*). *TMR* is the total range of all the missions combined. It then calculates the required number of aircraft using,

$$A/C = \frac{TMR}{M \cdot a \cdot Time(\sec)} + \frac{GT(\sec) \cdot Missions}{Time(\sec)}$$

where *M* is Mach number, *a* is the speed of sound and *GT* is the ground time per mission. This combination of a detailed code model and a short equation provide for a relatively simple and cheap analysis tool.

This program required to perform this analysis is short, but runs for a long time on a PC. Note that there is no Mach number dependency. This requires some explaining. Since the code is expensive to run, it would take days (on a 30 MHz 486 PC) to get answers for a range of Mach numbers, using different values for the aircraft range ,etc. It is extremely economical to use a simple equation to calculate the required number of aircraft as a function of the code output and Mach number.

4.4 Model Assumptions

Some assumptions were necessary to simplify the model representation. One assumption was to make the temperature over the whole mission profile equal to the lowest temperature in the Polar Vortex, 190 °K. This is equivalent to reducing the range of the aircraft, and effectively increases the number of missions required.

Another assumption has to do with the polar vortex. Cicerone specifications treat it as a cylinder. The shape of the polar vortex has usually been that of an oval stretching from South America to Australia. This shape changes some from year to year, making it difficult to model.

One implicit assumption is that the mission can be achieved in the specified manner. For instance, the distance between two delivery paths is only 1,732 m. This may be hard to maintain, resulting in non-uniform delivery.

4.5 Mission Analysis

South America, Australia and Africa were considered as locations for airport bases, in order of increasing distance to the Polar Vortex. Combinations of the first two bases and all three were considered. Using the model previously established, the former combination resulted in 6006 missions versus 6066 missions for the latter combination. Not a big difference, but enough to conclude that Africa is not a desirable airport location. The reason being that the larger distance from Africa to the polar vortex requires more aircraft, as well as that operating from two airport regions simplifies ground operations. Additionally, the oval shape of the Polar Vortex as previously discussed, makes the range from Africa to the Polar Vortex longer than the model assumes, adding a third reason to dismiss Africa as a airport region.

Antarctica was also considered as an airport base, but problems associated with operating an airport in an environment such as Antarctica, particularly during the dead of winter, may nullify any payoffs. However, field locations in Antarctica may be desirable as emergency airfields.

Figure 4.5-1 depicts the number of aircraft required versus Mach number comparing different ranges. The minimum range here is 6000 nm. That is approximately the distance to the South Pole and back. As the range is increased from 6,000 to 7,000 nm. the payoffs are tremendous, and the reduction of aircraft is approximately 60%. But as the range is further increased, the reduction in the number of aircraft is less significant. As a result, the 8,000 nm. range was decided upon as the baseline for this mission. That is the current limit for commercial jets and the payoffs at the higher ranges are too small to make them desirable.

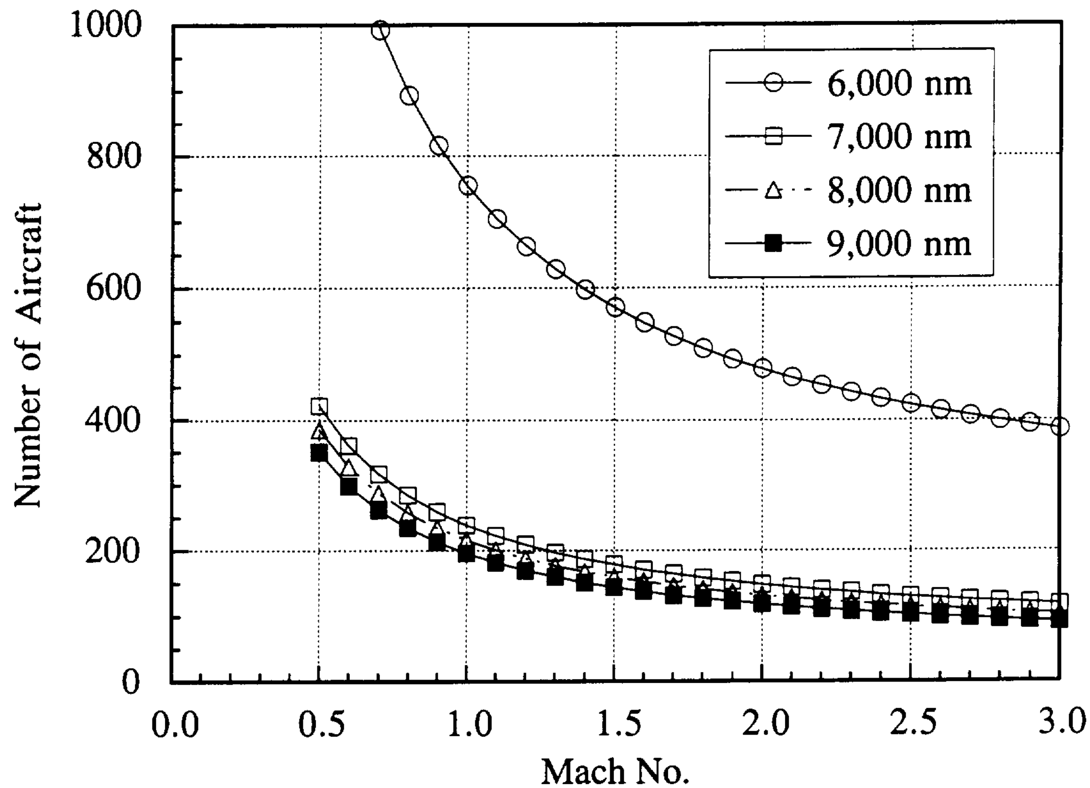


Figure 4.5-1. Comparing different aircraft ranges. Airport locations are in South America and Australia. Ground Time is 4 hours and the plume area is 3.142 $km.^2$

Figure 4.5-2 shows the number of aircraft required versus Mach number comparing different payloads. This is at the 8,000 nm. total range baseline. As the payload is increased the payoffs are large at first, in terms of number of aircraft, but the payoffs decrease rapidly, and the difference between carrying 35,000 lb or 40,000 lb payload is barely observable. As a result, 35,000 lb payload was determined to be the baseline.

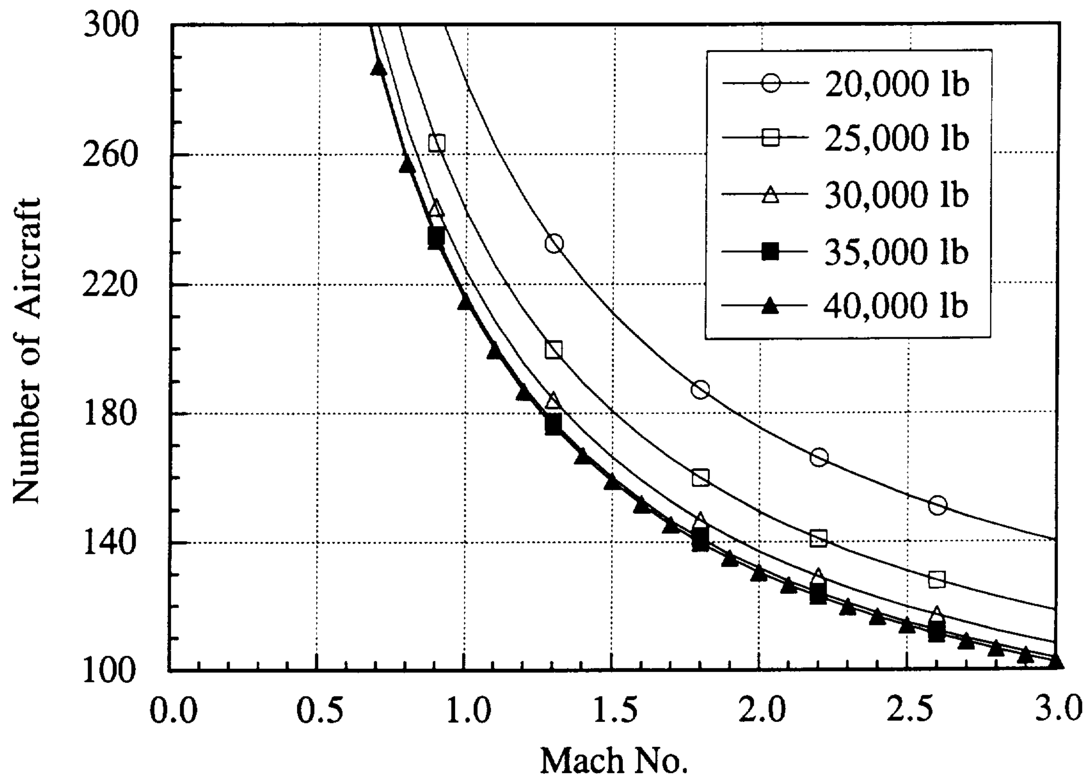


Figure 4.5-2. Payload Considerations. Airport locations are in South America and Australia. Ground time is 4 hours and the plume area is 3.142 $km.^2$ Range is 8,000 nm.

For a given range and plume area there exists a total number of missions. As you increase the Mach number you need fewer aircraft to accomplish these missions. In turn each aircraft needs to be employed more often. This relationship is depicted in Figure 4.5-3 for different ranges. This does not induce any limitations on the aircraft type. However, for fewer number of planes which are deployed more often, maintainability and reliability become more important and any misfortunes in regard to these will be more severe. For example, if one plane becomes inoperative, it will be more severe for a fleet of 50 planes than 500. This needs to be taken into consideration in the design phase. Figure 4.5-3 also shows indirectly the time of flight. For subsonic aircraft this is between 18 and 24 hours, requiring two crews to be present on each mission.

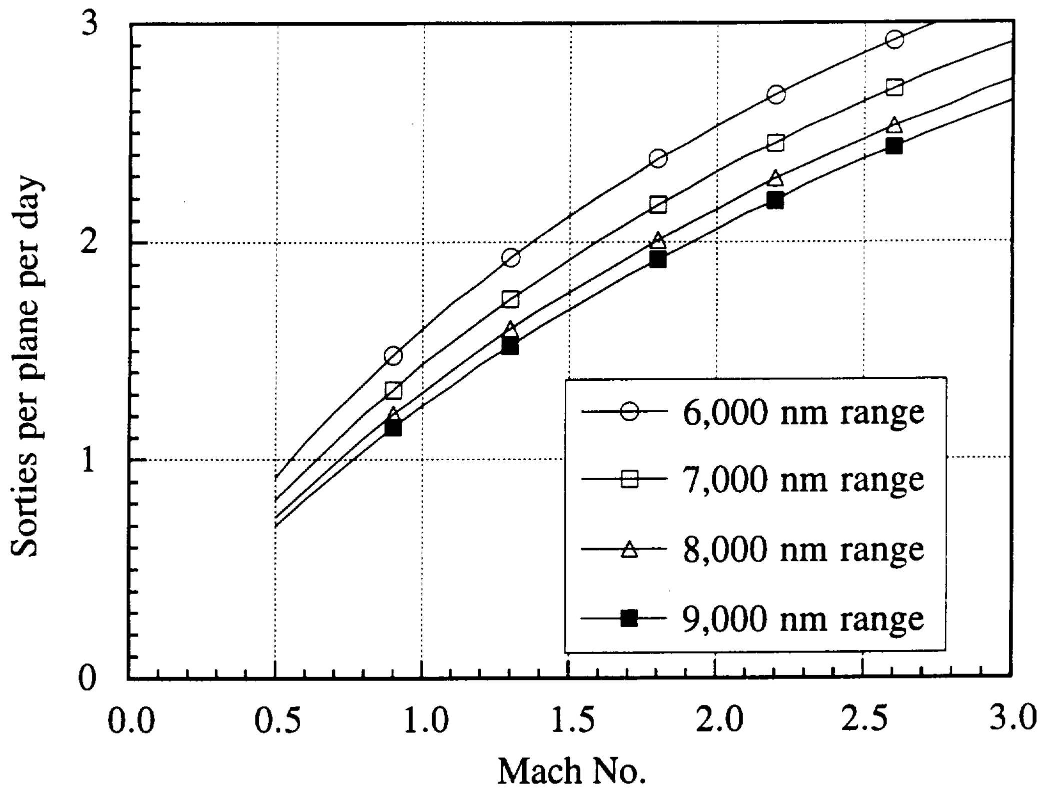


Figure 4.5-3. Comparison of Sorties per plane per day vs. Mach number, for different ranges. Airport locations are in South America and Australia. Ground time is 4 hours and the plume area is 3.142 $km.^2$

Reloading fuel and hydrocarbon and other activity on the ground was included in terms of ground time. It is assumed to be 4 hours. Figure 4.5-4 shows that the ground time has a large effect on the number of aircraft. This is more severe at high Mach numbers; there is a difference of 18% at M=0.8 compared to 40% at M=2.4. This shows that care must be exercised in the design process, minimizing the time it takes to reload payload, fuel etc.

Operations on the airports need to be taken into account. Figure 4.5-5 depicts how many landings and takeoffs per day are necessary at each airport region vs. range. Note that this relationship only depends on the number of missions, which in turn depend only on the range of the aircraft and the plume area. The curve falls off relatively sharply and levels off with increased range. At 8,000 nm. range the traffic is at 300 landings and takeoffs per day (24 hrs.) That might necessitate two airports per region, increasing the cost significantly. However, an alternate airfield may be necessary anyway, in case one airfield is closed because of weather or due to an accident. Airfields already in the area may be used, or rough fields constructed. Again, only 300 landings and takeoffs per day can be achieved per airport.

Some missions will consist mainly of injecting propane at the edge of the vortex, while others will be flying to the South Pole. Which means that the former missions will not have the payload capacity to effectively use its available delivery range, and the latter missions will not have the total range to effectively use its payload capacity. These two different mission profiles contradict each other in terms of requirements for the aircraft

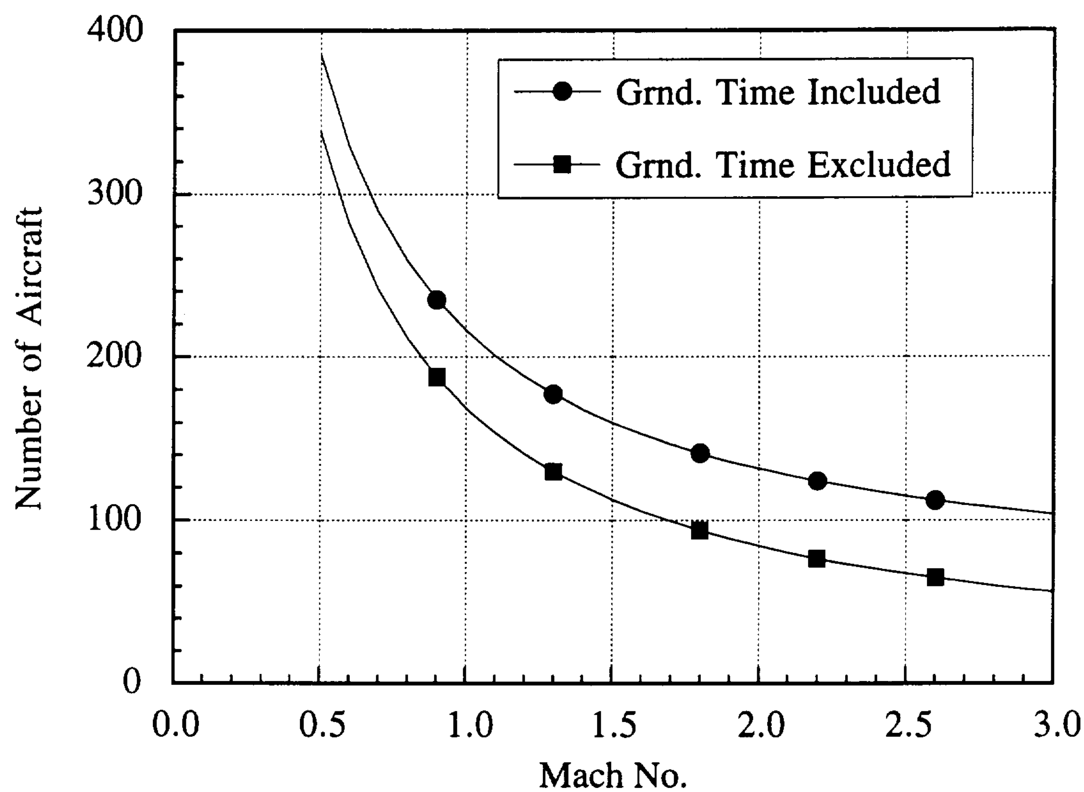


Figure 4.5-4. Effect of Ground Time on the number of aircraft. Airport locations are in South America and Australia. Ground time is 4 hours and the plume area is 3.142 km.2

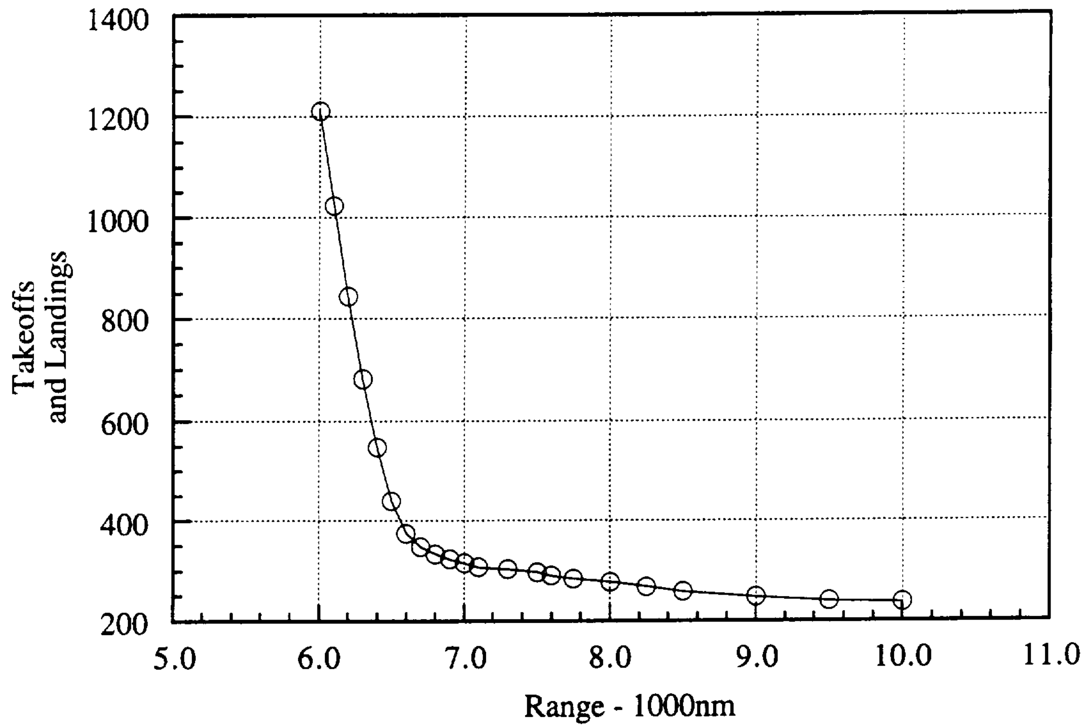


Figure 4.5-5. Comparison of landings and takeoffs vs. aircraft range per airport region. Airport locations are in South America and Australia. Ground time is 4 hours and the plume area is 3.142 $km.^2$

design, and both can not be satisfied simultaneously. On the other hand, the latter missions could load up with less than full capacity of propane, increasing fuel capacity and range, making those missions more efficient. That would require a design that builds into the aircraft more fuel capacity than required for the 8,000 nm. nominal range. Note that the aircraft would not have to be designed to carry more weight, only have increased fuel capacity. Since three tanks will be used for the propane storage, one of the tanks might possibly be used for this purpose.

The sizing analysis (Ch. 8) concludes that supersonic aircraft would be more advantageous than subsonic in fulfilling this mission, the main reason being the altitude limit of subsonic aircraft. A Mach number of 2.4 was chosen as a baseline, mainly because of material considerations. This is discussed in more detail in chapter 8. Assuming that maintenance requires 10% more aircraft, 130 airplanes are required at that Mach number, including a ground time of 4 hours per mission.

4.6 Summary

The results of this analysis can be summarized as follows:

- Airports in South America and Australia
- 8000 nm. Range
- 35,000 lb. Payload
- 130 airplanes at M = 2.4

5. HYDROCARBON INJECTION AND MIXING

5.1 Introduction

The mission analysis of Chapter 4 assumes each plane can uniformly mix the propane in a cylindrical plume behind the aircraft. The hydrocarbon injection scheme requires propane to be dispersed in an plume area of 0.916 square nautical miles (approximately 1,000 m radius circle). This chapter is devoted to determine the validity of the plume mixing estimate. A mathematical model was developed to analyze the plume area estimate. The model assumed the propane was injected into a constant velocity flow with turbulence. The turbulence arises from the chaotic flow caused by separation. Figure 5.1 shows the basic physical representation of the model. The Cicerone method requires a uniform distribution of 3.6 ppb propane in the ozone hole. Due to mixing phenomena and aircraft range and payload limits, it is determined that an average concentration of 3 ppb can be achieved. This is lower than suggested by Cicerone, but is still within acceptable limits and it would be a good starting point for actual physical testing.

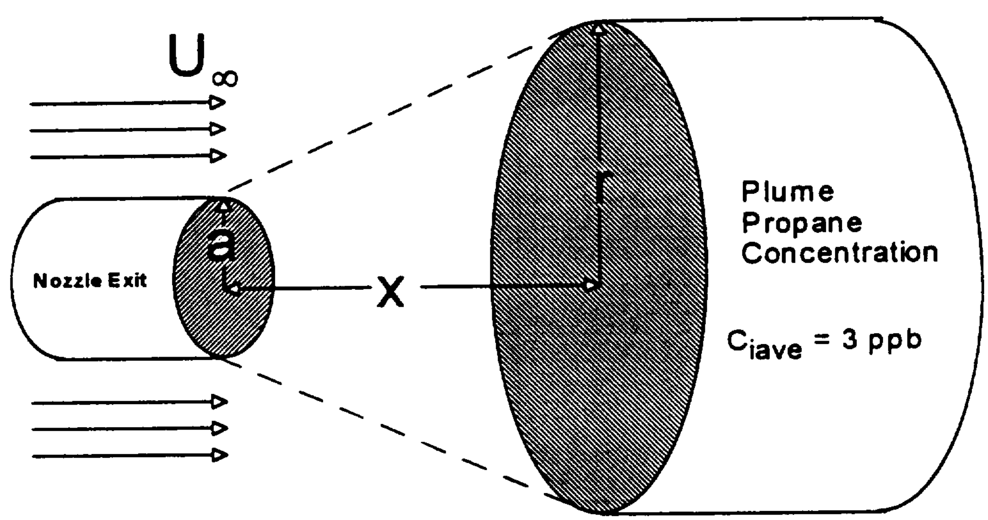


Figure 5.1 Hydrocarbon injection model.

5.2 Classical Diffusion Theory

Classical diffusion theory(Carslaw & Jaeger, p.260) can be used to predict the propane concentration level after its injection from the aircraft, including turbulent mixing.[9] The analysis was developed with the help of Dr. Joseph A. Schetz (Schetz, *Boundary Layer Analysis,* p. 378). A classical heat transfer integral was modified for species diffusion including the results of a turbulent mixing model:

$$C = \frac{e^{-\left(\frac{r^2}{4cx}\right)}}{2cx} \int_0^a e^{-\left(\frac{r'^2}{4cx}\right)} I_0\left(\frac{rr'}{2cx}\right) r'dr'$$

C = propane concentration
x = plume distance behind aircraft
r = plume radius,
a = nozzle exit radius
$I_0(rr'/2cx)$ = Bessel function,
$c = \frac{0.025}{0.8}\left|1 - \frac{\rho_j U_j}{\rho_\infty U_\infty}\right|$, a turbulent mass diffusion coef.
ρ_j = propane injection density
ρ_∞ = air density
U_j = propane injection velocity
U_∞ = aircraft velocity

This integral was adjusted for the propane injection mixing model with the help of Dr. Joseph A. Schetz (Schetz). The original integral is a heat transfer equation which was modified to predict the results of the mixing model.

5.3 Circular Coverage Function

The P-function is defined as the off-set circular probability function also known as the circular coverage function. The P-function simplifies the integration of certain diffusion problems in cylindrical coordinates (Masters, pp.1865-1871). The P-function provides the plume size and propane concentration for the integral defined above.

Swirl was utilized to increase the amount of propane injection mixing (Schetz, *Injection and Mixing in Turbulent flow*, pp.111-122). The exit nozzle design would include vanes to initiate the swirl effect (Figure 5.2). The swirl effect only adjusts the constant, "c", which is used in the P-function, thus the turbulent mixing and swirl results are combined to maximize propane mixing.

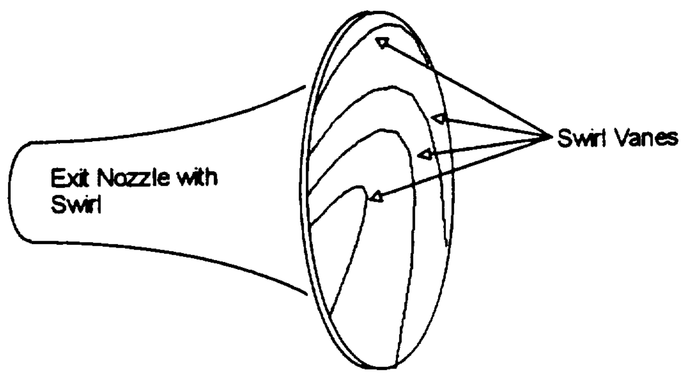


Figure 5.2 Nozzle with swirl vanes to mix propane at release.

5.4 Calculations

Several assumptions are used to define an injection mixing baseline. For ease of calculation, the temperature of the air and propane are assumed to be identical. The velocity of the propane was assumed to be 1/16th of the free stream velocity. This allows swirl to be effective and allows the supersonic and subsonic cases to be more easily compared. The exit nozzle radius is defined to be 0.1m. All of the assumptions are based on engineering judgment and test the validity of the mission analysis plume area estimate. Analysis shows that the mixing time may increase with parameter changes, however the required plume area and concentration can be achieved. Appendix A contains sample calculations for the Mach

2.4 aircraft. The calculations for the Mach 0.8 aircraft are identical; except for the free stream velocity value.

The P-function constant, "c", is determined from calculation 5.1. Calculation 5.2 shows how the swirl effect increases "c". The distance behind the aircraft for the required mixing is calculated in 5.3. The average propane concentration is assumed to be 3.6 ppb. The average propane concentration in the plume is assumed to 40% of the centerline concentration. The P-function is calculated and the mixing distance is obtained.

Calculation 5.4 contains the required mixing time based on aircraft velocity. The plume radius is calculated in Calculation 5.5. This utilizes the centerline value of the P-function to arrive at the resulting plume radius. Calculation 5.6 determines the actual average propane concentration. The average was calculated by integrating the plot in Figure 5.3.

Calculation 5.7 determines the total and aircraft propane payload requirements. This is based on the average concentration, plume radius, and aircraft injection range. Calculation 5.8 determines the required propane mass flow exiting the injection nozzle.

5.5 Results

Figure 5.3 shows the propane concentration as a function of radius for the Mach 2.4 case. The slight concentration increase near the plume edge is due to the plume overlap mentioned in the mission analysis flight pattern. Obtaining 3.6 ppb propane concentration would more than quadruple the required aircraft payload. Therefore, an average concentration of 3.0 ppb propane is dispersed in the ozone hole. This lower than optimum concentration still satisfies the requirements of the Cicerone method. The propane concentration in the inner third of the plume will satisfy the Cicerone requirements. The outer part of the plume has less than the required concentration, however natural diffusion will be utilized to mix the propane.

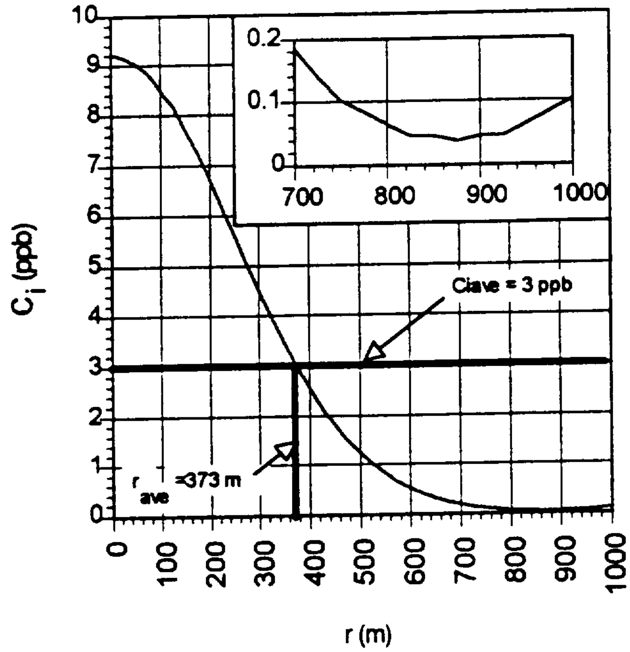


Figure 5.3 Propane concentration.

Table 5.1 compares the Mach 2.4 and Mach 0.8 aircraft. The results are similar due to identical dispersion range, payload, and propane injection velocity compared to free-stream velocity. The only difference is the mass flow and the mixing time. Changing input parameters will not drastically adjust the mixing plume size.

Table 5.1 Propane diffusion parameters for two Mach numbers
(note: U_j=1/16U for both cases)

	Mach # = 2.4	Mach # = 0.8
plume radius (m)	1000	1000
x (m)	$5.9x10^6$	$5.9x10^6$
propane (ppb)	3.0	3.0
diffusion time (hours)	2.5	7.4
aircraft payload (lbs)	34000	34000
propane required (tons)	100000	100000
mass flow @ nozzle exit (lbs/s)	0.3781	0.126

5.6 Conclusions

The plume radius of 1,000 meters required by mission analysis can be obtained by turbulent mixing with induced swirl. The average concentration of propane in the plume is 3.0 ppb. As Figure 5.3 shows, the concentration decreases as the radius increases. The slight propane concentration increase near the plume radius maximum is due to flight pattern over-lapping as discussed in mission analysis. The concentration 3.0 ppb is less than the predicted Cicerone requirement of 3.6 ppb. However, 3 ppb is greater than 2.8 ppb propane concentration which may damage the ozone.

Subsonic aircraft would be better if the plume area more than doubled. Plume area can not be obtained by adjusting the propane exit velocity (the only parameter which changes between the super- and subsonic cases). The plume radius may be doubled by increasing the amount of injected propane. This would require a mixing time approaching two days. The P-function injection analysis may not accurately predict the mixing beyond several hours. The inaccuracy would arise from the effect of natural dispersion in the Antarctic polar vortex. Also, the additional propane would tax the already limited range of the subsonic aircraft.

6. OPERATIONAL BASES

6.1 Introduction

Airports and ground support are vitally important to aircraft operation. Two choices for airports exist; either existing airports could be used or new airports could be constructed. However, before the selection can be made, some initial requirements must be established. The airports need to be as near the south polar vortex as possible. They will also need provisions for propane storage. Based on the mission analysis and mixing requirements it was determined that a total of 100,000 tons of propane is required for the overall mission. So, 50,000 tons of propane will need to be stored in each of the two airport regions as described in Chapter 4.

6.2 Existing Airports

The two areas of interest are the southern tip of South America including the Falkland Islands and the Australia/New Zealand region. Table 6.1 lists civilian airports with runway lengths of at least 8,000 ft. which are being considered for bases of operation.

Table 6.1 Airport Locations and proximity to Polar Vortex

Airport	Latitude (deg.)	Longitude (deg.)	Range to Polar Vortex (nm.)	Range to South Pole (nm.)
Porvenir, Chile	-53.2667	70.3333	844.53	2206.53
Mount Pleasant, Falkland Islands	-51.8167	58.4500	931.63	2293.63
Rio Gallegos, Argentina	-51.6167	69.2833	943.64	2305.64
Christchurch, New Zealand	-43.4833	-172.5330	1432.21	2794.21
Puerto Montt, Chile	-41.4330	73.1000	1555.37	2917.37
Avalon, NSW, Australia	-38.0333	-144.4670	1759.58	3121.58
Melbourne, Victoria, A-Tulamar	-37.6667	-144.8330	1781.60	3143.60
Auckland, New Zealand-INTL	-37.0167	-174.7830	1820.65	3182.65
Canberra, ACT, Australia	-35.3167	-149.2000	1922.77	3284.77
Buenos Aires, Argentina-Pistari	-34.8167	58.5333	1952.80	3314.80
Sydney, NSW, Australia-Kingsford	-33.9500	-151.1830	2004.86	3366.86
Santiago, Chile	-33.3833	70.7833	2038.90	3400.90
Rosaria, Argentina	-32.9167	60.7833	2066.93	3428.93
Mendoza, Argentina	-32.8333	68.7833	2071.94	3433.94
Perth, WA, Australia	-31.9333	-115.9670	2126.00	3488.00

Source: (Boeing Aircraft Company)

Figure 6.1 (Astapenko) shows a map of the south polar region including Africa, Australia, and South America. This map helps illustrate the relative locations of the airports. The airports are not indicated on the map due to the confusion it may cause.

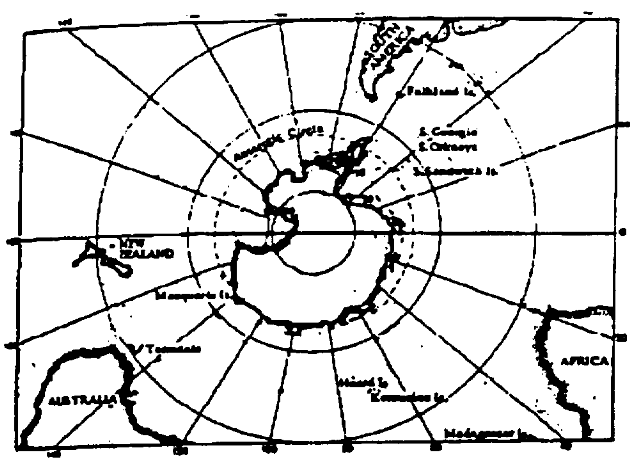


Figure 6.1 Map of South Polar Region (Astapenko)

Difficulties may arise at existing airports due to the large amount of propane storage necessary. The propane needs a large amount of storage space which may be limited, and the airports need to maintain certain safety levels. This may be resolved by building the propane storage depot in an area far enough from the airport that the danger is minimal. Trucks could then be used to transport the propane and fill the aircraft's onboard tanks. Other limitations to using existing airports are the availability of aircraft storage and actually gaining access to the airport. Aircraft storage may not be much of a problem because nearly all the aircraft will either be flying or preparing to takeoff. The most difficult problem to overcome will be gaining permission to use the airport and to build propane storage.

Another limitation for an airport is air traffic. For an average international airport the maximum capacity of the entire airport is approximately 300 operations per hour; an operation being either a landing or a takeoff (Lauth). The typical daily demand for an airport servicing larger aircraft is about 310 to 350 operations per day (Airport Capacity and Delay). As determined in the mission analysis 263 operations per day are required for each region. This could be handled by one airport in each region, even with other traffic using the airport. Using only one airport would be preferable in that only one propane depot need be constructed for each region.

6.3 New Airports

Constructing entirely new airports would solve many of the problems associated with existing airports. They could be located in areas that would minimize the risk of catastrophic occurrences. For example a major explosion of the propane storage area would cause little damage if located in an unpopulated, undeveloped area. A new airport would be designed to meet the minimum requirements for the mission. In doing so, the airport could also be suitable for general aviation and commuter type aircraft, or even emergency aircraft staging areas (i.e. medical, rescue, and fire fighting units). This could benefit the local community by providing jobs and encouraging business and industry to move into the area. Figure 6.2 illustrates a possible layout for a new airport configuration. The two runway configuration provides an operational base of 94 operations per hour under visual flight rule (VFR) conditions and 60 operations per hour under instrument flight rule (IFR) conditions. VFR conditions occur with a cloud ceiling over 1,000 ft and visibility of over three statute miles. IFR conditions occur when the cloud ceiling is between 500 and 1,000 ft and the visibility is between one and three statute miles (Airport Capacity and Delay).

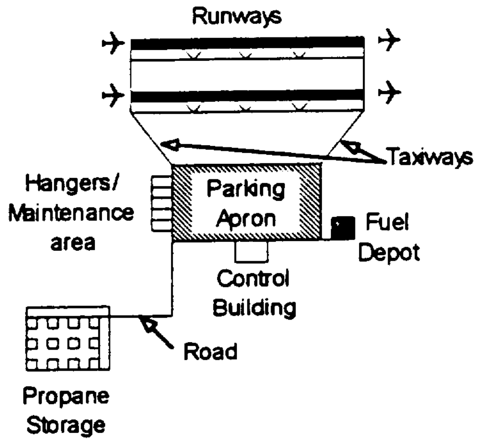


Figure 6.1 Possible airport configuration

6.4 Other Locations

Locations other than South America and Australia were considered for new airports. The locations considered were small islands between 55°S and Antarctica. Due to the inclement weather of the islands, the barren and often mountainous terrain, and the time of year of the mission, the islands were rejected as possible locations.

6.5 Summary

It would be preferable to use existing airports for bases of operations, since the only major construction necessary would be the propane storage areas. However, if the existing airports can not be used, then new airports would need to be constructed in locations as near the polar vortex as possible without being subjected to Antarctic winter storms.

7. PROPANE STORAGE

7.1 Propane Delivery Scheme

The aircraft propane storage system is an integral part of the delivery scheme. The propane will be stored in three identical, individual tanks. A multiple tank system is utilized to minimize center of gravity shift, reduce the in-flight stresses on the tanks, and provide safety in a redundant system. Each tank is to be furnished with a relief valve in case of an emergency. The tanks will be filled using check valves to prevent back flow in the lines when the pressure changes. The internal pressure of the three tanks will be utilized to disperse the propane. Each of the three tanks will be individually connected to settling chamber with a butterfly valve to maintain a constant stagnation pressure. The stagnation pressure will be such that the flow into the air stream, after traveling through piping and an appropriate exit area nozzle, will have the necessary velocity and mass flow rate to achieve the required plume area.

7.2 Types of Storage Systems

Two types of aircraft storage systems are feasible: a cryogenic temperature system and an atmospheric temperature system. A cryogenic temperature system would store the propane in 100% liquid form at a very low temperatures (200K) (Vargatile p.235). This system would consist of the propane tank and insulation to prevent heat loss. This system requires extensive ground support systems and a cryogenic plant on site to cool the propane before it is placed on the aircraft.

An atmospheric temperature system would store the propane at atmospheric temperature, but would use a pressure of about 10 atmospheres in order to keep the propane at approximately 85% liquid state (Southwestern Virginia Gas Service Corp.). The system would consist of a tank able to withstand the high pressure and a compressor to load it into the tanks.

7.3 Pressure Vessel Design

The pressure vessel was designed to be a cylindrical tank (length: l and radius: r) with spherical end caps (radius: r). This tank provides the best balance of maximum space utilization and minimum stresses. The dimensions of the tank were approximately optimized to provide maximum volume while maintaining minimum weight and not violate the constraints of supersonic flight.

Aluminum 2199 was used for the tanks because of its high tensile strength, low density, its ability to withstand low temperatures if the cryogenic system is used, and is readily available at a reasonable cost. The properties of this alloy are:

density, $\rho_{Al} = 503.29 \text{ kg/m}^3$

yield stress, $\sigma_Y = 200$ MPa (Barnier, p.10)

These tanks are designed so that they would not fail due to the internal pressure and weight of the propane at a 2.5 g load. Failure would occur along the center line on the bottom of the tank when the tank is full due to weight concentration (Johnson). A minimum thickness was determined to prevent the tank from failing according to Tresca's failure theory for 2-D stresses (Va. Tech Thin Walled Structures 1991)

$$\sigma_A^2 - \sigma_A \sigma_B + \sigma_B^2 < \sigma_Y^2 / \text{ F.S.}^2$$

where:

σ_A	= Stress A = Hoop stress = PR/t
σ_B	= Stress B = Longitudinal Stress = $PR/2t$
P	= Pressure which the tank has to stand
R	= Radius of the cylindrical tank
t	= wall thickness
F.S.	= Factor of safety = 2

The pressure value used was the gage pressure of the tanks plus the pressure caused by the propane weight along the tank bottom centerline times a load factor of 2.5, which is standard for air transports (Va. Tech Aircraft structures 1990). Pressure caused by the weight comes from the following equation:

$$P = \rho_p \; D \; g$$

where:

P = pressure caused by the contents weight
ρ_p = density of the propane
g = gravitational constant
D = diameter of the tank = $2R$ (Megyesy, p.29)

In addition to these requirements the tanks were set at a minimum thickness of 1/8" to comply with the standards set by American Society of Mechanical Engineers (ASME) for pressure vessels (Megyesy, p.23).

An appropriate tank size was designed by minimizing the tank mass as a function of tank volume. A FORTRAN program was used to iterate tank radius as a function of the tank volume. Tank wall thickness was determined using Tresca's failure theory as previously stated.

The inner temperature of the cryogenic temperature tanks is much lower than the external temperature in the aircraft. Therefore insulation is necessary in order for the propane to stay in a liquid form without a high pressure. Polyurethane foam was decided upon as the insulator to be used because of its low density, high thermal conductivity, and it is relatively inexpensive. A conservative estimate based upon previous research done in cryogenic storage systems gives 4" of insulation. Polyurethane foam has the following characteristics.

Density = 30 kg/m^3
Thermal conductivity = 210 microwatt/cm K

The mass of the insulation needed is equal to its density times the volume of insulation.

Knowing the dimensions, and the material used the weight of the tanks were determined from the equation:

$$Mass = \rho_{Al} \; Vol$$

where:

$Mass$ = mass of tank
ρ_{Al} = density of Aluminum 2199
Vol = Volume of tank structure
$= \{(r+t)^2 l + 4/3 \; \pi(r+t)^3\} - \{\pi r^2 l + 4/3 \pi r^3\}$

Based on a three tank system, Table 7.1 shows the final tank design.

Table 7.1 Comparison of storage systems

Dimensions	Atmospheric	Cryogenic
Radius	.5 m	.5 m
Length	12.964 m	11.596 m
Wall thickness	.004975 m	.003175 m
Mass	608 kg	348.8 kg

The cryogenic system takes up less volume and offers less tank weight. Even with additional insulation weight it would seem like the cryogenic system has the clear advantage. This is negated, operationally, due to the extensive ground support and loading systems required for this system. In addition, atmospheric tanks do not involve an unreasonable payload or cargo volume penalty and its flexible loading system would decrease turn around time. Therefore the atmospheric temperature system will be used.

7.4 Overall Dispersion System

Once the tank size has been set, the tank system will become an integral part of the aircraft. The propane tanks in the fuselage must be able to withstand the torsional and bending moments imparted by the aircraft as well as the weight of the propane. It is also necessary that the tanks can be removed from the aircraft with ease.

The tanks are placed in a support system whose outer shell is rigidly fixed to the airframe at one end while the other end is relatively free in order that the torsional and bending moments of the aircraft are reduced when transferred to the tanks. The three tanks are rigidly fixed to one other, and are then attached to the outer shell with a non-rigid support. The tank system is equipped with tracks which move along airframe mounted ball bearings so that it can be removed from the aircraft (Fig 7.1).

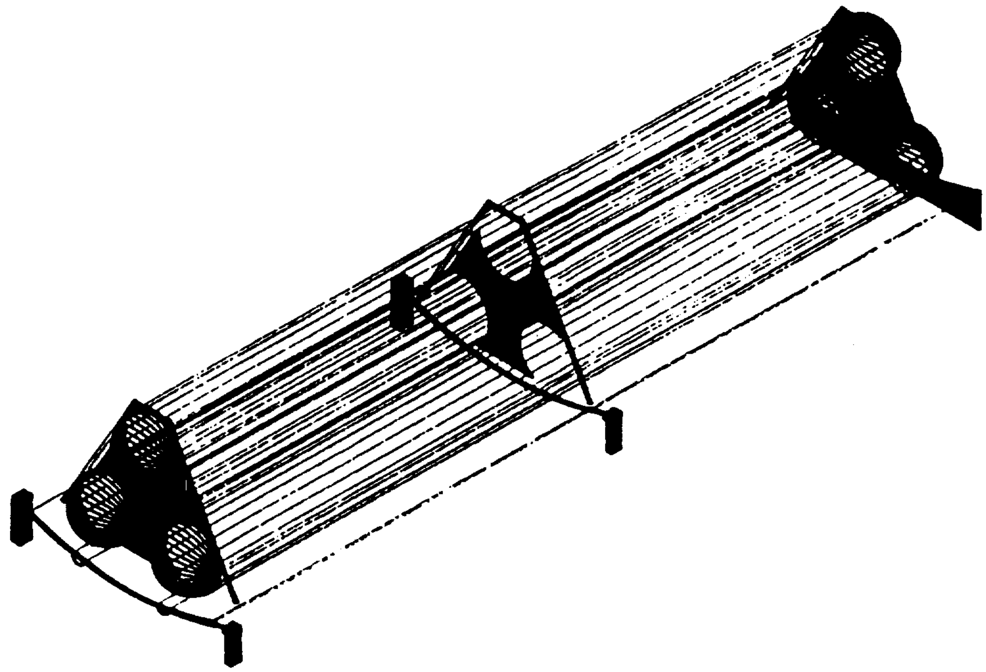


Fig. 7.1 Storage Tank Design

In flight, propane flow control is regulated with a series of pipes and valves. The pipes through which the propane is loaded and unloaded will be made of aluminum. The required minimum thickness of the pipes according to ASME standards must be 3.175 mm. The inlet valve loading the propane into the tanks is a stainless steel check valve; the outlet, a stainless steel butterfly valve. The check valve was selected because it prevents back flow in the lines. The butterfly valve throttles the release propane as needed and is also able to jettison the full propane load in case of emergency.

The coefficient of thermal expansion values for stainless steel ($9.6 \times 10^{-6}/°F$) and aluminum ($13.1 \times 10^{-6}/°F$) are close such that, with enough overlap in the joining flange, the propane will not leak because of the small difference. A neoprene (rubber chemical resistant with propane) gasket between the pipe and tanks will compensate for the small expansion differences. These basic requirements will provide a support and safety system for the tanks in the fuselage.

The dispersion system was designed with the nozzle area for the appropriate mass flow rate to provide the necessary 3 ppb concentration. This was determined using the back pressure in the settling chamber and compressible flow analysis. This is dependent upon the actual configuration of the aircraft and the location of the nozzle on the aircraft. In addition the nozzle is to be designed so that boundary layer separation is prevented.

8. FLIGHT REGIME AND SIZING

8.1 Subsonic or Supersonic Cruise

To further define aircraft requirements, it was necessary to select the flight regime (subsonic or supersonic) and to set the general size of the aircraft. Mission analysis provided the following requirements for an aircraft delivery system:

Range of 8000 nm.
Cruise Altitude of 47,000 to 66,000 ft
Payload of 35,000 lbs. of usable propane
Plume area of about 0.916 nm^2

This data was then used to make a comparison of supersonic cruise vs. subsonic cruise aircraft.

8.2 Supersonic vs. Subsonic Cruise

8.2.1 Advantages and Disadvantages of Supersonic Cruise

A supersonic cruise aircraft delivery system has distinct advantages over a subsonic one. These included :

1) A supersonic cruise mission requires fewer aircraft than a subsonic cruise mission. The subsonic mission flying at Mach (M) = 0.8, requires 230 aircraft, while, to accomplish the same number of sorties, the supersonic mission, cruising at M = 2.4, requires only 130 aircraft .

2) A supersonic cruise aircraft provides more mission flexibility than its subsonic counterpart. Examples of potential secondary uses include civil, commercial, and military transports as well as high altitude research, for which the aircraft can be leased or purchased. This would offset the cost of the aircraft among many diverse investors.

3) Supersonic cruise aircraft can more easily reach the required altitude than a subsonic aircraft. (Supersonic: Concorde - 60,000 ft cruise, SR-71 - 80,000 + ft cruise).

4) Current research is directed toward developing High Speed Civil Transports (HSCT), which are intended to supersonic cruise at Mach 2.4 and carry 50,000 lbs. payload with a similar mission profile. The cost of development can be reduced by building on the work of others.

5) A supersonic cruise aircraft might distribute the propane more effectively due to the larger amount of energy being dissipated in the atmosphere.

$$\text{Energy} = \text{Work Done} = \text{Force} \times \text{Distance} \tag{8.1}$$
$$\text{Energy} = \text{Work Done} = \text{Drag} \times \text{Range} \tag{8.2}$$

Drag can be calculated from:

$$\text{Drag} = \text{Lift} / L/D_{Cruise} \tag{8.3}$$

where L/D_{Cruise} is the lift to drag ratio during cruise, and since

$$\text{Lift} = \text{Aircraft Weight} \tag{8.4}$$

then

$$\text{Drag} = \text{Aircraft Weight} / L/D_{Cruise} \tag{8.5}$$

For equivalent sized aircraft, the supersonic cruise aircraft will have a $L/D_{Cruise} = 10.$ while the subsonic aircraft will have a $L/D_{Cruise} = 20$ to 30. Thus the supersonic aircraft will have much more drag (roughly 2-3 times as much) than the subsonic cruise aircraft. It would release more energy per sortie into the atmosphere in the form of shocks, friction and wingtip vortices and jet wake (exhaust). This would theoretically give better mixing for the supersonic cruise aircraft.

6) The supersonic cruise mission would require a smaller operational staff due to the shorter sortie flight times. The mission flight time for the supersonic flight is 6.1 hrs. at $M = 2.4$, compared to 18.2 hrs. at $M = 0.8$. The subsonic aircraft would need a relief crew(s) for every sortie due to the extended flight time. For this reason, the subsonic mission would double the number of flight personnel, ground facilities, and ground support personnel required for the supersonic mission.

7) Analysis and research was conducted into the requirements of high altitude flight. It was found that a supersonic cruise aircraft like the Concorde could be constructed in existing facilities and use existing runways.

Disadvantages of supersonic cruise aircraft:

1) The supersonic aircraft would have to withstand heating of the leading edges of the aircraft and the resulting thermal stresses.

2) The supersonic aircraft would be noisier than the subsonic aircraft. The noise would come from shocks while cruising and the engine noise. The altitude the aircraft is flying at would reduce the noise pollution at the surface, however, it will be difficult to meet existing noise regulations at take-off.

8.2.2 Advantages and Disadvantages of Subsonic Aircraft

Subsonic cruise aircraft have limited advantages over supersonic cruise aircraft:

1) A subsonic aircraft is likely to be more reliable, but the larger number of planes required for this mission may nullify this advantage of subsonic cruise over supersonic cruise.

The disadvantages of the subsonic aircraft are:

1) Larger number of planes.

2) More flight crew and ground support.

3) Difficulty reaching altitude. No subsonic aircraft has ever been built that would take 40,000+ lb. of payload (propane plus tank weight) to 66,000 feet altitude and fly 8000 nm., and not much research has been done in that area. For example, the service ceiling for civil transports is around 42-45,000 ft., while for the B-52 it is 55,000 ft. The service ceiling of an aircraft is defined as the altitude where the climb rate = 100 ft./min.

4) The subsonic aircraft would require a large wingspan of 270 ft, thus requiring the development of costly manufacturing techniques and facilities. A rough estimate of the size of the aircraft required for this mission is discussed in Section 8.3.

5) There is no existing subsonic aircraft on which to build.

6) The subsonic aircraft would require extremely advanced structural development. With an AR = 12 and with a sweep of approximately 30°, the structural span will be 312 feet.

8.3 Sizing

Once an aircraft delivery system was decided upon, preliminary sizing was performed. A general mission profile was defined to be the same for either flight regime. This mission profile was:

1) Engine start and warm-up.
2) Taxi and take-off.
3) Climb to between 47,000 and 66,000 ft.
4) Cruise for 8,000 nm.
5) Loiter 20 minutes.
6) Descend
7) Land, Taxi, and shut-down.

For comparison, sizing of aircraft for each flight regime was performed using Nicolai's sizing algorithm. The supersonic cruise aircraft was also sized on ACSYNT. The ACSYNT model also explored the possible use of current High Speed Civil Transport aircraft (HSCT) for the propane injection mission.

8.3.1 Nicolai's Sizing Method

The aircraft was defined to carry a 50,000 lb. payload for the entire mission. This was to ensure that the aircraft would be flexible enough for the advantage of other uses and to allow growth in the payload.

Nicolai's sizing method was implemented in a BASIC program with the following input parameters:

Reserve Fuel	= 5 % of TOGW
Trapped Fuel	= 1 % of TOGW
atmospheric press.	= 116.5 lb./ft.^{2}(at 66,000 ft.)
	= 5 % of sea level pressure
Speed of Sound	= 926.6 ft./sec.
Structural Technology Factor = 0.8	

The speed of sound for this mission was estimated by an average of the speed of sound in the ozone hole (911.0 ft./sec.) and the speed of sound outside the ozone hole (942.2 ft./sec.) at altitude range. The difference was due to the lower temperature in the ozone hole.

Weight fractions for engine start, warm-up, taxi, takeoff (W2/W1), climb and initial acceleration (W3/W2), and acceleration to cruise Mach number (W5/W4) were taken from Roskam, Vol. I, p. 12, for supersonic cruise aircraft and for transports for the subsonic aircraft.

8.3.1.1 Supersonic Cruise

The focus of the sizing was the supersonic cruise mission because of the advantages it has over the subsonic mission, primarily the multirole adaptability and the fewer number of aircraft required.

The supersonic cruise Mach number was set to 2.4 to minimize heating effects and to utilize existing research and technology. Supersonic cruise L/D (Lift to Drag) was varied from 9.0 to 10.5. Cruise SFC (specific fuel consumption) was varied from 0.65 to 0.85 this is representative of non-afterburning, low-bypass turbo-fan engines.

The results of the entire run are shown in Fig. 8.1. This gives some idea as to the TOGW and weight of fuel required per plane (sortie) and thus to some of the costs. The best case supersonic aircraft (lightest) had:

cruise L/D = 10.5
cruise SFC = 0.65 lbs./sec./lb.
total fuel = 104,900 lbs.
TOGW = 248,700 lbs.

Accounting for trapped and reserve fuel, the usable fuel/sortie is 90,000 lbs. This aircraft is almost half the size of the Concorde.

(Note: The faculty advisors do not concur with the use of this value of sfc. A value twice this large is more appropriate, thus the aircraft size results obtained by the design team are not realistic)

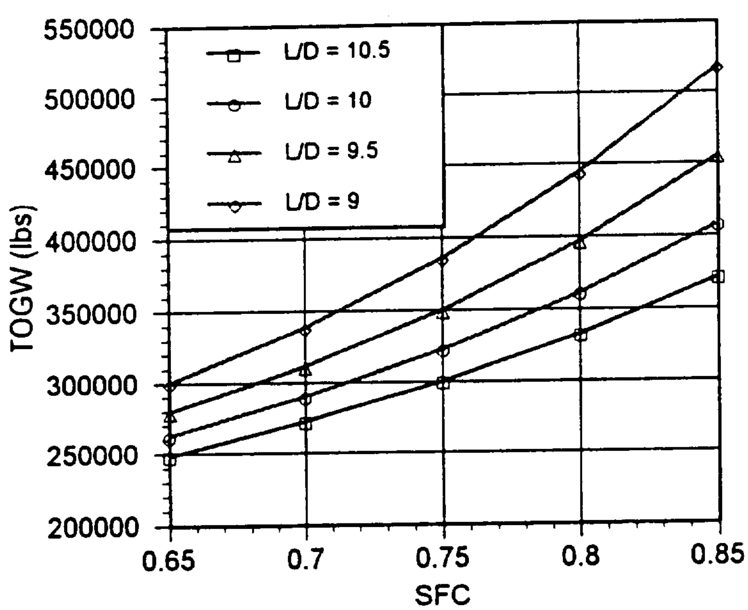


Figure 8.1 TOGW sensitivity

Since cost correlates to TOGW and weight of fuel per mission, this best-case aircraft would give the cheapest alternative. However, the values of L/D and SFC are only estimates. This is the reasoning for the varying of SFC and L/D. Figure 8.1 shows how TOGW varies with SFC and L/D. Those results were for aircraft optimized for those values of SFC and L/D. As can be seen from the graphs, an aircraft using existing technology would be Concorde sized. It's characteristics were:

cruise L/D	= 9.5
cruise SFC	= 0.80 lbs./sec./lb.
total fuel	= 201,200 lbs.
TOGW	= 397,400 lbs.

A more detailed analysis would be required to ascertain the best values for L/D and SFC. In addition, the weight fractions for climb, acceleration, and reserve fuel and loiter time could be further analyzed for optimum design.

8.3.1.2 Subsonic Cruise

For comparison, a subsonic aircraft satisfying the mission profile was sized. Subsonic cruise, M = 0.8 was selected to optimize L/D and range. This represents current designs of civil and military long range transports. The mission and input parameters used earlier for a supersonic cruise mission were also used for the subsonic case. Weight fractions were again taken from Roskam, Vol. I, p. 12, however, the fraction for subsonic transports were used.

Using an AR = 12, which is slightly better than modern civil transports, and other input based on Lockheed U-2 data, a reasonable weight was obtained for the subsonic aircraft. The results were:

L/D cruise	= 23.185
Cruise SFC	= 0.48 lbs./sec./lb.
Total Fuel	= 106,800 lbs.
TOGW	= 283,500 lbs.
CL cruise	= 0.894

This aircraft is almost identical in weight to the best-case supersonic cruise aircraft. However, it requires:

Wing Area = 6074. ft.^{2}
Span = 270. ft.

These values are larger than equivalent weight civil transports. For example, the 767-200 has a TOGW = 300,000 lbs., and a span of 156 ft (Roskam, Vol. I, p. 40, Vol. II, p. 197). Figure 8.3 contains sketches comparing the two supersonic size aircraft, the subsonic aircraft, and the 767-200.

Based on the substantial advantages that a supersonic cruise aircraft had over a subsonic cruise aircraft, it was decided that the supersonic cruise mission would be used.

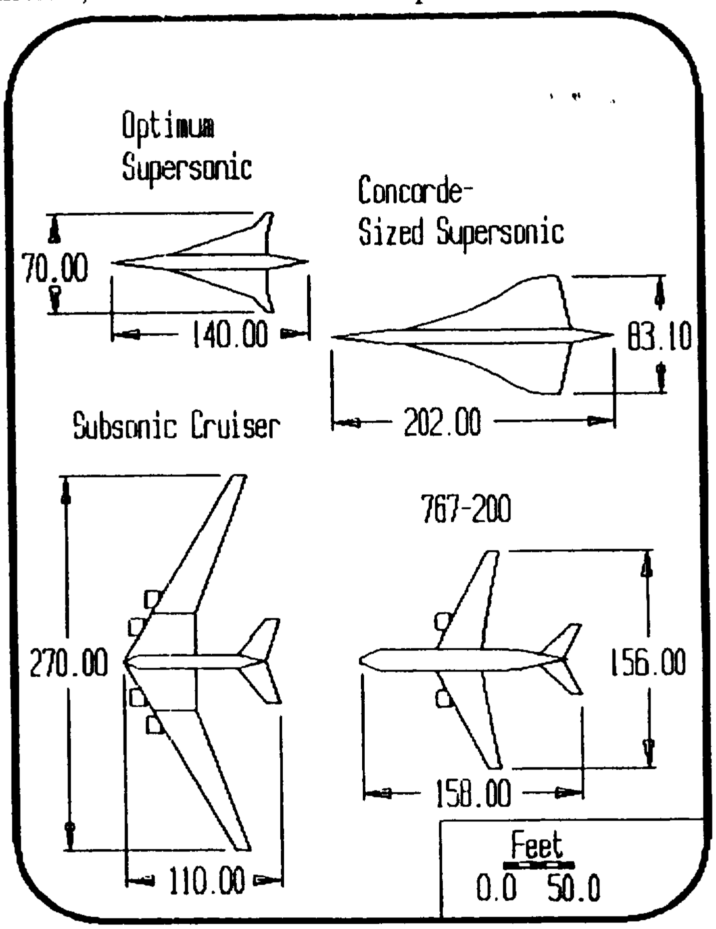


Figure 8.3 Comparison of notional concepts

8.3.2 ACSYNT

In order to verify the Nicolai sizing method as well as compare the mission profile with that of the High Speed Civil Transport, the mission profile data was parameterized in ACSYNT. ACSYNT is a software package on the IBM RISC System for Aircraft Synthesis currently under development at the Virginia Tech Mechanical Engineering CAD/CAM laboratory.

Current design parameters for ACSYNT's HSCT model include a payload of 50,000 lbs and a maximum speed of Mach 2.4. The standard mission of the HSCT calls or the use of afterburners for the first three legs of the mission profile. The range of the HSCT is set at 4,000 nm.

Modification of the HSCT input file was done targeting the areas of range, L/D and SFC. These were selected as the governing parameters in determining if the HSCT could be further developed to meet the mission profile for propane injection. As stated in the sizing section, for economical flight, the goals of L/D of 10 or greater for the wing-body configuration and an SFC of 0.80 or less must be achieved at cruise. The payload of the HSCT was left at 50,000 lbs.

Of the three parameters, range was the simplest to successfully modify. This involved a change in the cruise distance and the total range variables in the input file such that the total flight range went from 5,500 nm to 8,000 nm. The L/D parameter is a function of the actual aircraft geometry as well as an array of variables in the input file. First the geometry file was modified by increasing the wing area and strake area. Second, the variable array was altered until the L/D was in the necessary range of 10.5, which is an increase from 6.1 Modification of the SFC proved to be more difficult. Many variables govern the calculation of the SFC in the propulsion module as well as the weights module of ACSYNT. Unfortunately, not all of the necessary changes were made to drive the SFC of the HSCT down to 0.80. Modification, however, was made in the trajectory module to exclude afterburners, in accordance with the stated mission profile.

Results of the ACSYNT analysis yielded an aircraft weighing 770,962 lbs, which is approximately 16% increase in weight from the baseline HSCT. However, the increase in weight is a result of a 45% increase in range. The overall fuel weight increased 38%. Further weight savings would have been achieved if the SFC had been correctly modified from the default of 0.98.

The results from ACSYNT, though not in total agreement with the results of the Nicolai sizing method, demonstrate that the HSCT as envisioned today can be modified to be an ozone fighter. The reason for the discrepancy in weights is the difference in SFC between what was run with the Nicolai method and ACSYNT. The Nicolai model was run with the design SFC of 0.65, while ACSYNT was run with an SFC of 0.98. Modification of the ACSYNT input file to adjust the SFC proved unsuccessful.

8.4 Aircraft Summary

Based on the mission profile a supersonic aircraft was selected with the following baseline parameters:

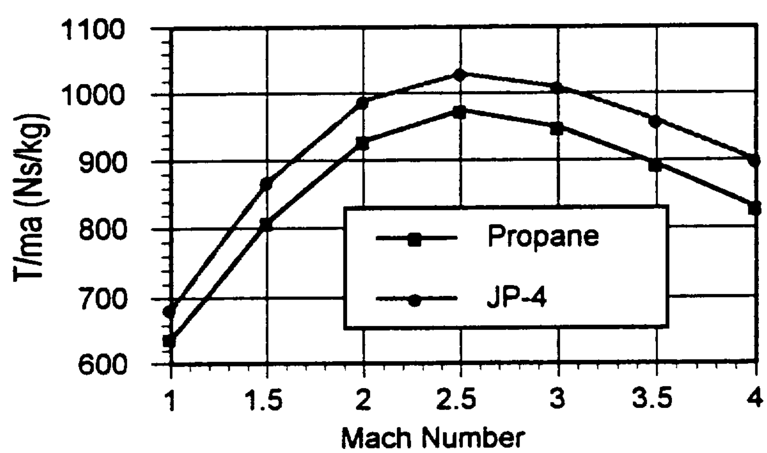
Range	= 8000 nm.
Altitude	= 66,000 ft. (max. cruise altitude)
Mach	= 2.4
L/D	= 10.5
SFC	= 0.65
K	= 0.8

Other factors considered in the comparison were a modular design as well as multiple mission capability. Analysis of the HSCT by ACSYNT, as modified to fit the above parameters, showed that the current HSCT design should not be excluded from consideration. The most important modifications that need to be made to the current HSCT are to the L/D and SFC parameters.

9. PROPULSION

9.1 Propulsive Fuel Analysis

Initially in the analysis of the design problem, much attention was focused on the propulsive fuel to be used in the aircraft. Several liquid hydrocarbon candidates were considered. After analysis of thermodynamic properties, ethane and methane were eliminated and propane was determined to be the most likely candidate. Its thermodynamic qualities nearly matched that of conventional jet fuels in several important categories. Its heating value actually exceeded that of JP-4 providing a higher combustion chamber temperature under the same fuel flow conditions. The graph below illustrates the relative specific thrusts of the two fuels over a range of Mach numbers (Data compiled using NOTS -- Naval Ordinance Test Station Combustor Analysis Software).



JP-4 Qr = 45,000 KJ/kg
Propane Qr = 50,343 KJ/kg

Figure 9.1 Specific thrust for candidate hydrocarbon fuels

Using propane as the propulsive fuel did pose problems with the tank design. As stated earlier, the propane to be injected into the stratosphere is to be stored in a cylindrical, standard temperature tank, mounted lengthwise in the aircraft fuselage. For the range and airspeed requirements from the mission analysis, the tank would have to be greatly modified to accommodate for the extra fuel to burn for propulsion. The volume would have to be increased by a factor of 14.7 and its radius would have to increase by a factor of 3.8. A tank wall thickness increase of approximately 1/4" would also be necessary to accommodate the extra weight of the fuel. This option would also leave the wings empty as no fuel could be stored due to the choice of pressurized propane storage tanks. One possible solution would be the use of cryogenic storage tanks to be placed in the wings for more efficient use of aircraft volume. This option would require expensive, bulky equipment, as well as increased ground support, and with current design requirements, its use is precluded.

9.2 Engine Cycle Analysis

With a range dominated mission profile, engine efficiency is of paramount importance. A supersonic cruising aircraft has been selected to accomplish the mission. Preliminary sizing analysis yielded an aircraft with a total gross weight (TOGW) about 250,000 lbs cruising without afterburner at Mach 2.4 at an altitude in excess of 60,000 ft. A supersonic cruise L/D of 10.5 was assumed. A supersonic cruising aircraft requires engines that can produce enough thrust to break through the transonic drag rise and propel the craft into the supersonic regime. An engine that could provide enough thrust while burning dry (no afterburning) requires extremely efficient components as well as very high combustor and turbine inlet temperatures. A compressor with higher compression per stage than currently obtainable is required to generate increased thrust without a significant weight penalty. Segmented burning in the combustor and bleed air for cooling would allow for higher chamber temperatures. High technology ceramic turbine blades and state of the art Thermal Barrier Coating (TBC) would allow for much higher turbine inlet temperatures. Given the proposed flight conditions, successive iterations of a cycle analysis for a turbojet engine yielded a cruise SFC of approximately 0.65 with a thrust output of approximately 20,000 lb. The table below summarizes the engine parameter requirements.

Table 9.1 Engine parameter requirements

Mach Number	2.4
Ambient Temperature	220K
Stagnation Pressure Ratio Across Diffuser	0.9136
Specific Heat Ratio of Compressor	1.2
Stagnation Temperature Ratio Across Compressor	1.682
Specific Heat Ratio of Turbine	1.1
Stagnation Temperature Ratio Across Turbine	0.786
Turbine Inlet Temperature	1900K
Stagnation Pressure Ratio Across Nozzle	0.97
Resulting Specific Fuel Consumption	18.407 mg/Ns (0.65 lb/hr/lbf)

These parameters were determined using algorithms proposed by Gordon C. Oats. With Mach number, ambient temperature, and efficiency ranges as input, the algorithms were successively iterated until the specific fuel consumption was minimized. The value of 0.65 lb/h/lbf is the lowest realistic SFC thermodynamically obtainable. The diffuser design is a significant phase of the engine design as its efficiency must be very high for this aircraft (0.9136). The very high stagnation pressure ratio across the nozzle also requires that its design be very efficient. These two factors have a significant effect on the specific fuel consumption of the engine in supersonic cruise.

9.3 Summary

The propulsion system will have to be powerful yet very efficient. These two requirements together appear "ideal" at first glance but with incorporation of the advancing technologies mentioned above, they are believed obtainable.

10. CONCLUSION

The objectives of the 1992/1993 NASA/USRA Advance Design Program group were 1) to define the processes which contribute to stratospheric ozone loss, 2) determine the best scheme to prevent ozone loss, and 3) to establish the baseline requirements of a vehicle-based system to implement the selected prevention scheme.

Objective 1 was obtained by analyzing current research and predictions. Analysis revealed that the destruction of ozone is caused by the presence of free chlorine. Ozone destruction is assisted by the presence of Polar Stratospheric Clouds (PSCs) and by the presence of a polar vortex. The sources of free chlorine were found to be man-made Chlorofluorocarbons (CFCs) as well as natural sources and it was discovered that ozone loss will continue for over 20 year even if CFCs are eliminated immediately.

Objective 2 was obtained by studying several schemes proposed to prevent ozone loss. Few of these methods were researched and none were physically tested. Many were not feasible due to economic and/or energy constraints. However, a hydrocarbon injection scheme proposed by R.J. Cicerone et al. was found to be reasonably feasible as well as having reasonably extensive theoretical background research. For these reasons the hydrocarbon injection scheme was selected for the project. Propane was selected for the injected hydrocarbon due to the ease of obtaining it and its greater effectiveness.

A greater body of research and design was undertaken to accomplish Objective 3. With the requirements of the hydrocarbon injection scheme as a base, mission analysis determined that an aircraft would be best suited to the task. Mission analysis also provided the aircraft range, total number of sorties, aircraft payload, number of aircraft, suitable geographic regions for mission bases, and numerous other parameters. Mission analysis along with turbulent mixing theory provided estimates of injection plume area.

Parameters from mission analysis were used to design the vehicle storage tanks. An aluminum three-tank system was selected, each tank a cylinder with spherical endcaps. Each tank was designed to be 0.5m in diameter and 12.97m in length. The total weight of all three tanks was calculated at 608 kg. In addition, a tank support system was designed to minimize airframe-imparted torsion and bending moments on the tanks and to ensure tank portability from the aircraft. Finally, an injection nozzle was theorized although unknowns associated with the actual aircraft flowfield prevented conclusive design.

Airports and facilities were studied, based on mission analysis. Analysis concluded that only two airports were required to support the mission, one in each airport region. In addition, the cost of building new airports was discussed as well as problems associated with using existing airports. Though analysis was inconclusive, a new, primary airfield with use of existing airports for emergencies is perhaps the best option available.

A large part of the overall analysis went into defining the baseline requirements of the vehicle used for injection of propane. The results are listed below:

1) 130 aircraft required
2) Cruise at Mach = 2.4
3) Range of 8,000 nm
4) Payload Weight = 40,000 lbs
5) Takeoff Gross Weight = 250,000 lbs
6) Engine SFC = 0.65
7) Current HSCT designs and technology are applicable
8) Mission flexibility
9) Modular design

The possibility of acquiring the necessary SFC, and exploration of the possible use of propane as an alternative fuel was considered. Conclusions were that predicted SFC values were possible, but dependent on engine nozzle design. Although propane provides energy comparable with typical JP-4 fuel, its use would increase the necessary propane volume by approximately 150%. This increase in volume would extend beyond aircraft limits, making propane unsuitable as a fuel for this design.

Finally, it must be noted that the above aircraft specifications provide the baseline for an "ideal" case. Though an aircraft of these specifications is ultimately desired, an aircraft of roughly Concorde size could be more easily constructed to fit the mission profile.

APPENDIX: Details of the Dispersion Calculation

Calculation 5.1 P-Function Constant c Calculation

$$\frac{0.025}{0.8} U_\infty \frac{D}{2} \left| 1 - \frac{\rho_j U_j}{\rho_\infty U_\infty} \right| = c U_\infty$$

Variable Definitions:

(1) U_∞ = free stream velocity = 663m/s

(2) D = diameter of injection nozzle exit = .2m

(3) ρ_j = propane density = $.1609 kg/m^3$

(4) ρ_∞ = air density = $.1609 kg/m^3$

(5) c = P - Function constant(*below*)

(6) a = radius of injection nozzle exit = .1 m

(7) MW_{prop} $\cong 44$

(8) MW_{air} $\cong 29$

Assumptions:

(1) temperature of propane = temperature of air

(2) D $\equiv 2a$

(3) U_j $= \frac{1}{16} U_\infty$, (to maximize c, greater c is then the greater the mixing radius is)

$$\Rightarrow c = \frac{0.025}{0.8} a \left| 1 - \frac{\rho_j U_j}{\rho_\infty U_\infty} \right| = \frac{0.025}{0.8} \bullet 0.1 \bullet \left| 1 - \frac{44}{29 \bullet 16} \right|$$

$\therefore \quad c = 0.00283$

(Schetz, *Boundary Layer Analysis*, p. 378)

Calculation 5.2 Swirl Effect on P-function Constant c

$$\frac{\nu_{Tswirl}}{\nu_{Tno\text{-}swirl}} = (1 + \lambda_s S_x)^2$$

Variable Definitions:

(1) ν_{Tswirl} = turbulent kinematic viscosity,

(2) Sx ="swirl" number evaluated locally along the flow direction,

(3) λ_s = function of swirl rate through Sx(utilized to obtain a "best fit" with experiment,

(4) Φ = initial swirl level

Assumptions:

(1) Swirl effective if Uj < mach 1

(2) $U_j = \frac{1}{16} U_\infty$

(3) Uj < mach 1

(4) $\Phi_{max} \le 0.6$ {if greater the flow will re-circulate}

(5) $\lambda_s = S_x \le \Phi_{max} \le 0.6$

$$\frac{\nu_{Tswirl}}{\nu_{Tno\text{-}swirl}} = [1 + (.6)^2]^2 = 1.85$$

$$\frac{c_{swirl}}{c_{no\text{-}swirl}} = \frac{\nu_{Tswirl}}{\nu_{Tno\text{-}swirl}} = 1.85$$

$$\Rightarrow c_{swirl} = 1.85 c_{no\text{-}swirl} = 1.85 \bullet 0.00283$$

$\therefore c = c_{swirl} = 0.00524$ (note: same for all Uj < mach 1 in this analysis)

(Schetz, *Injection and Mixing in Turbulent Flow,* pp. 111-122)

Calculation 5.3 Plume distance behind Aircraft

Variable Definitions:

(1) Ci = propane concentration

(2) r = plume radius

(3) a = nozzle radius

(4) c = P - Function constant,

(5) Ciave = average concentration

(6) Ccl = centerline concentration

(7) P* = normalized P - Function

Assumptions:

(1) Ciave $= 5.5 \bullet 10^{-9}\{3.6 \text{ ppb}\}$

(2) r = 340m

(3) a = 0.1m

(4) c = 0.00524

(5) Ciave = 0.4Ccl

$$\text{Ccl} = 0.4\text{Ccl} = 1.4 \bullet 10^{-8}\{9 \text{ ppb}\}$$

Calculate the normalized P - function

$$P^* \equiv \frac{\text{Ciave}}{\text{Ccl}} = 0.39$$

Read the input parameter from P - function table

$$\Rightarrow \frac{r}{\sqrt{2cx}} = 0.1 \bullet \frac{0.43 - 0.39}{0.43 - 0.375} + 1.3 = 1.37$$

Calculate plume distance behind aircraft

$$\Rightarrow x = \frac{r^2}{1.37^2} \bullet \frac{1}{2c} = \frac{340^2}{1.37^2} \bullet \frac{1}{2 \bullet 0.00524}$$

$\therefore x = 5.9 \bullet 10^6 \text{m}$

(Masters, pp. 1865-1871)

Calculation 5.4 Required Propane Diffusion Time

Time required to obtain propane concentration:

$$\text{Mach No.} = \frac{U\infty}{\text{speed of sound}}$$

$$\text{speed of sound} = \sqrt{\gamma RT}$$

$$\gamma = 1.4,\ \ R = 287\frac{J}{Kg \bullet K},\ \ T = 190\ K$$

$$\Rightarrow \text{speed of sound} = 276\text{m/s}$$

$$U\infty = 663\text{m/s (M 2.4)}$$

$$U\infty = 221 m/s\ \text{(M 0.8)}$$

$$\text{time} = \frac{x = 5.9 \bullet 10^6 m}{U\infty}$$

$$\therefore \text{Mixing Time} = 2.6\ hours\ (@M\ 2.4)$$

$$= 7.4\ \text{hours (@M 0.8)}$$

Calculation 5.5 Plume Radius of Average Propane Concentration

Assumptions:

(1) Ci $= 0$

(2) $Ciave$ $= 5.5 \bullet 10^{-9}$

(3) a $= 0.1$

(4) c $= 0.00524$

(5) x $= 5.9 \bullet 10^6 m$

$$\frac{a}{\sqrt{2cx}} = \frac{0.1}{\sqrt{2 \bullet 0.00524 \bullet 5.9 \bullet 10^6}} = 0.0004$$

$$P^* \equiv \frac{Ci}{Ciave} = 0$$

Radius determined from P - function chart assuming zero concentraion @ plume edge

$$\Rightarrow \frac{r}{\sqrt{2cx}} = 4.0$$

$$\therefore r = 4 \bullet \sqrt{2cx} = 995m \approx 1000m$$

(Masters, pp. 1865-1871)

Calculation 5.6 Actual Average Propane Concentration

Plot 5.1 contains the propane concentration vs. the plume radius. The actual average propane concentration was calculated by integrating the plot.

$$C_{iave} = 4.55 \times 10^{-9} \text{ (3 ppb)}$$

Calculation 5.7 Aircraft Propane Payload & Total Propane Required

Assumptions:

(1) $r_{air} = 0.1609 \frac{\text{kg}}{\text{m}^3}$ {worst case scenario/heaviest aircraft payload} (Anderson)

(2) propane temperature = air temperature

$$r_{plume} = 1{,}000\text{m}$$

$$C_{iave} \equiv \frac{r_{prop}}{r_{air}} = 4.55 \bullet 10^{-9} \ \{3\text{ppb}\}$$

$$\Rightarrow r_{prop} = 4.55 \bullet 10^{-9} \bullet 0.1609 \frac{\text{kg}}{\text{m}^3} = 7.32 \bullet 10^{-10} \frac{\text{kg}}{\text{m}^3}$$

$$\frac{\text{propane mass in plume}}{\text{unit length}} = r_{prop} \bullet AREA_{plume} = 7.32 \bullet 10^{-10} \frac{\text{kg}}{\text{m}^3} \bullet 1{,}000^2 \text{m}^2 \bullet P = 0.0023 \frac{\text{kg}}{\text{m}}$$

$$\text{Aircraft}_{propane-payload} = \frac{\text{propane mass in plume}}{\text{unit length}} \bullet \text{range}_{injection} = 0.0023 \frac{\text{kg}}{\text{m}} \bullet 6.718 \bullet 10^6 \text{m}$$

$$\therefore \text{Aircraft}_{propane-payload} = 15{,}272 \text{ kg} = 34{,}000 \text{ lbs}$$

Calculation 5.8 Exit Nozzle Mass Flow rate

$$\rho_{propane} = .13157 \frac{kg}{m^3}$$

$$U = \frac{1}{16} U_\infty : \ U_{2.4} = 41.4 \frac{m}{s} : \ U_{0.8} = 13.8 \frac{m}{s}$$

$$\dot{m} = \rho_{propane} U A_{exit\ nozzle}$$

$$\therefore U_{2.4} \Rightarrow \dot{m} = 0.1711 \frac{kg}{s}$$

$$\therefore U_{0.8} \Rightarrow \dot{m} = 0.057 \frac{kg}{s}$$

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