NASA Contractor Report 186021

A Preliminary Design and Analysis of an Advanced Heat-Rejection System for an Extreme Altitude Advanced Variable Cycle Diesel Engine Installed in a High-Altitude Advanced Research Platform

Richard P. Johnston

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A Preliminary Design and Analysis of an Advanced Heat-Rejection System for an Extreme Altitude Advanced Variable Cycle Diesel Engine Installed in a High-Altitude Advanced Research Platform

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1.0 Summary

This is a study of an advanced heat rejection system for an Advanced Variable Cycle Diesel (AVCD) engine installed in an extreme altitude High Altitude Advanced Research Platform (HAARP). It was conducted by the DieselDyne Corporation under the sponsorship of the Dryden Flight Research Facility of the NASA Ames Research Center. Results of the study show that it is possible to reject waste heat from an AVCD engine propulsion system at the maximum sampling altitude (120,000 feet) selected for the study vehicle.

Use of a wing radiating element as part of the high temperature engine charge air heat rejection system was highly weight and drag efficient. An innovative arrangement of plate-fin convective elements and radiating elements allowed the achievement of mission altitude and cruise range goals.

Four major engine heat rejection systems were evaluated in the study. Two were low temperature engine coolant and oil cooling systems, one was an intermediate temperature system for supercharger aftercooling and the fourth was a high temperature intercooler. Since a very high pressure ratio turbocompressor was used as the initial charge air compression unit, high intercooler entry temperatures were developed at upper altitudes.

The lightest heat rejection system resulted from the use of aluminum plate-fin convectors for working fluid temperatures below 600 Deg. F. and a wing mounted nickel alloy radiating element for temperatures above 600 Degrees. The complete heat rejection system weighed more than 2000 pounds (lbs) for the study vehicle and was heavier than the installed engine.

Use of the AVCD engine and advanced heat rejection system with the study flight vehicle permitted all mission goals to be achieved or exceeded. These included a range of 6000 nautical miles (n. miles) and cruising subsonically at 100,000 feet altitude or above for more than 3 hours. The vehicle also carried an instrumentation package weighing 1000 lbs. The propulsion system and vehicle allowed the envisioned ozone sampling mission to be completed from a South American base to the region of the "ozone hole" over Antarctica.

Figure 1-1 is a plan view of the final vehicle along with system configuration information. To achieve the final sampling altitude of 120,000 feet required a .7 Mach Number (Mn) cruise capability. The cruise altitude and extended range resulted in an extremely large vehicle with a Take Off Gross Weight (TOGW) of more than 16,000 lbs and fuel load of 2800 lbs.

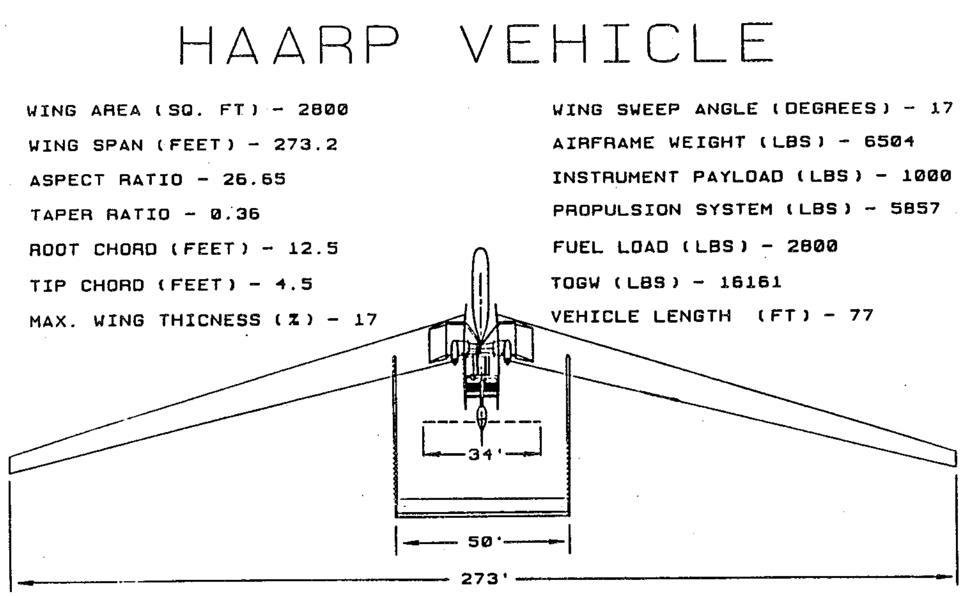

Figure 1-1 Description of final HAARP vehicle.

Based on projected AVCD engine performance with the advanced heat rejection system and estimated vehicle drag polars, a mission analysis was performed. Figure 1-2 shows the final mission profile. A flight duration of 19.6 hours (hrs.) was required to complete the 6000 n. miles mission. During the flight, 14.3 hrs. was spent at 100,000 feet or above with 1.2 hrs. spent at 120,000 feet. At the important 100,000 foot endurance cruise condition, engine system operation was altered to enhance fuel economy. A peak net shaft thermal efficiency of more than 60 per cent (%) was reached at the 100,000 foot cruise level.

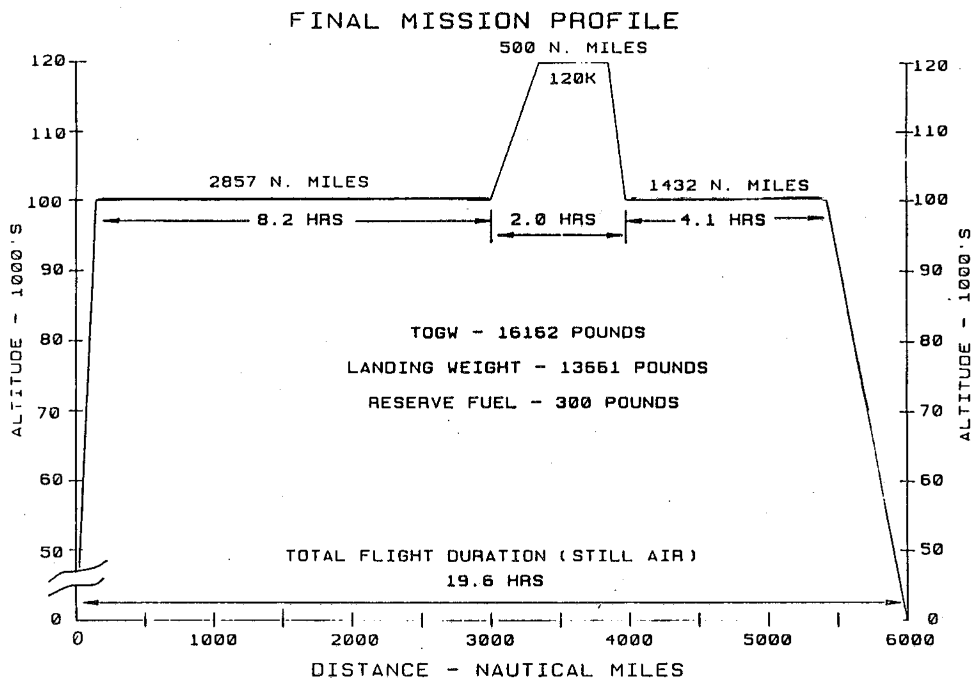

Figure 1-2 Final HAARP mission profile.

2.0 Introduction

Concern with perceived changes in the upper atmosphere ozone layer has prompted atmospheric scientists to propose high altitude sampling of the regions of greatest change. One of these areas that is well known to the public is the "ozone hole" over the Antarctic. Satellite surveillance has shown that the ozone concentration in this region changes from time to time with the size of the ozone hole growing and shrinking.

Since the ozone concentration affects the amount of ultra-violet radiation reaching the ground, significant reductions in atmospheric ozone could allow harmful (to humans) levels of ultra-violet radiation. There are many theories as to why ozone levels change including the incidental release of fluorinated hydrocarbons (freon) into the air and its interaction with ozone in the upper atmosphere. Some proposed actions to stop the reduction of ozone might be useless, or worse, have adverse economic effects. Therefore, there is reluctance to carry out some of the proposed actions without more proof of the actual causes of ozone depletion.

Obtaining useful atmospheric data would require a flight vehicle that could reach altitudes of at least 100,000 feet. The vehicle would also have to fly subsonically during the sampling portion of the mission. (The subsonic requirement is to prevent shocks from occurring as the sample passes into the vehicle's instrumentation) Such a vehicle would also need long endurance since the vehicle's operational base would be far from the sampling region of most interest.

Due to the lack of prior experience with HAARP type vehicles, the Dryden Flight Research Facility asked for studies dealing with various barrier technologies for such a vehicle.

Based on a High Altitude Long Endurance (HALE) study for the Defense Advanced Research Projects Agency (DARPA)[1], one major problem for flight at 100,000 feet and above is rejecting engine and charge air pre-compression heat. Therefore, the DieselDyne Corporation proposed a study of an innovative radiator/convector system to reject engine charge air and engine waste heat.

Figure 2-1 is a schematic of the extreme altitude AVCD engine propulsion system configuration for the HAARP. A high pressure ratio (55:1 design) exhaust driven multi-stage turbocompressor is used to perform initial engine charge air compression. An intercooler is needed to reduce the turbocompressor exhaust charge air temperature before it enters the second pre-compression element, an engine driven supercharger. The supercharger pressure ratio (5:1 design) requires additional cooling of the charge air in an aftercooler before the air enters the AVCD engine.

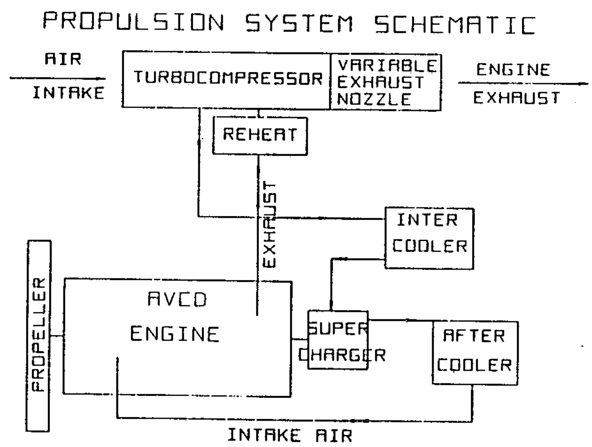

Figure 2-1 Schematic of AVCD extreme altitude propulsion system.

Since the heat exchanger portion of the study had a significant effect on AVCD propulsion system performance, a consultant was retained to perform the detailed analysis of the heat exchangers. Mr. James Bourne, of Lytron Incorporated, analyzed the behavior and effectiveness of the various heat exchanger systems. Cycle data from the proprietary DieselDyne Diesel Analysis Program (DAP) was used for heat exchanger boundary conditions. Fifteen performance points, reflecting the behavior of the engine as the vehicle proceeded through its mission, were used to characterize heat exchanger operation. A separate Lytron report on this portion of the study is included as an Appendix.

As exhaust backpressure decreases during climb, compressor work increases with the pressure ratio across the turbine. Therefore, as the ability to reject heat from convective charge air coolers diminishes with the decreasing density of available cooling air, the total heat of compression to be rejected multiplies.

During the current study, turbocompressor discharge temperatures reached a peak of 1362 Degrees Fahrenheit (Deg. F.) at the 120,000 foot cruise condition. Rejection of significant amounts of charge air heat by radiation at operating temperatures such as this are possible through relatively small radiating areas. (See page 45 in the Appendix.) A judicious choice of heat rejection between radiating and light alloy (lower temperature) convectors offered the possibility of a practical light weight system.

Determining the best arrangement of components and propulsion system performance characteristics required the use of a study vehicle capable of performing the mission. Once the vehicle and its performance were defined, it was then possible to do system trade-offs to arrive at favorable system configurations. Literature surveys did not reveal any existing study vehicles capable of performing the HAARP mission.

However, some work[2] done by the Teledyne Ryan Aeronautical Company (TRA) for the Flight Dynamics Laboratory of the Air Force Systems Command featured an Advanced Long Endurance Vehicle (ALEV) designed for a 90,000 foot cruise. This very advanced vehicle utilized the latest in lightweight structural design and low drag aerodynamics. The ALEV characteristics were scaled up appropriately to have the necessary performance (primarily climb rate and load carrying ability) to accomplish the HAARP mission goals. Drag polar estimates for the ALEV vehicle were used to estimate the thrust requirements of the scaled up study vehicle.

Table 2-1 illustrates the steps of going from the original TRA vehicle (left column) to the initial DieselDyne study proposal vehicle (far right column). The initial DieselDyne vehicle had a TOGW of 12366 lbs and carried a fuel load of 1712 lbs for the original 2500 n. mile endurance mission. A weight allowance was made for a pilot, a pressurized cabin and adequate engine bleed air to create a "shirt sleeve" cabin environment. However, as the range increased and mission endurance eventually reached almost 20 hours, these items were replaced by additional avionics, tanked fuel and an increasingly heavy propulsion system.

Table 2-1 Scaling of original ALEV to DieselDyne study vehicle.

WING AREA(sq ft)	1816		2000	2400	2800	2800
ENGINE SCALE			1	1	1	.8
WEIGHT (lbs)	Base Wt					
WING	2301.4	Base*S2/S1	2534.581	3041.498	3548.414	3548.414
BODY	458.3	"	504.7357	605.6828	706.6300	706.63
TAILS	201	"	221.3656	265.6388	309.9119	309.9119
TAIL BOOMS	356.5	"	392.6211	471.1454	549.6696	549.6696
CONTROLS	90.3		90.3	90.3	90.3	90.3
PROPULSION	1277.2	Ref R.P. Johnston	3500	3500	3500	2900
ELECT, AVIONIC, ETC	657.2	(1)	657.2	657.2	657.2	657.2
MISC.	40		40	40	40	40
OIL	25	Base*2	50	50	50	50
UNUSABLE FUEL	9.8		9.8	9.8	9.8	9.8
PAYLOAD	1300	Defined by proposal	1000	1000	1000	1000
PILOT & PERSONAL EQUIP.	0	(2)	250	250	250	250
OEW - LG	6716.7		9250.604	9981.265	10711.93	10111.93
LANDING GEAR	360.2	Base*(OEW-LG)2/(OEW-LG)1	496.0870	535.2705	574.4541	542.2775
OEW	7076.9		9746.691	10516.54	11286.38	10654.20
MISSION FUEL	1948.3	Determined from mission	1866.84	1849.15	1881.69	1711.8
TOGW	9025.2		11613.53	12365.69	13168.07	12366.00

(1) 365 lbs of RPV guidance and control avionics has been converted to pressurized cockpit and pilot support equipment.
(2) 250 lbs have been added for the pilot and carry on support equipment.

By the time the original solicitation was in print, the 3 hour endurance requirement had been increased to 2500 n. miles. This increased range was incorporated as part of the study proposal. As siting requirements were better appreciated, it became necessary to increase the mission endurance even more. When the current study began, the program manager requested that the mission endurance be extended to 6000 n. miles. Fortunately, the scaled vehicle (and engine size) had adequate wing surface area (and power) to carry the increased fuel loads for the longer endurance. It was not possible to reach 120,000 feet on the initial climb with the heavier vehicle. Only after a large fuel burn-off (see Figure 1-2) was it possible to reach the final 120,000 foot sampling altitude.

The AVCD engine used for the study is based on a very successful series of high speed two-cycle diesel aircraft engines developed by the Junkers Company of Germany between the World Wars. The opposed piston, twin crankshaft layout of these earlier Junkers engines has been retained by the DieselDyne AVCD engine designs. Figure 2-2 is a typical cross section of the AVCD and illustrates the compact arrangement of the ported two stroke diesel. One outstanding feature of these engines is the excellent air handling provisions of the intake/exhaust manifolding.

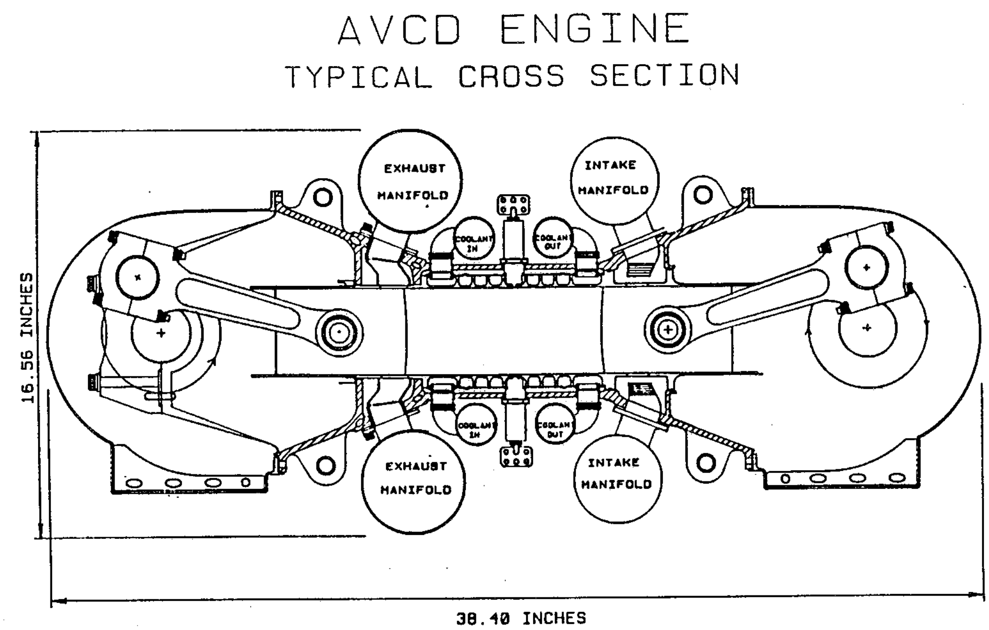

Figure 2-2 AVCD engine cross section.

Figure 2-3 is a top and side view of the AVCD engine block used for the study. The top view shows the arrangement of intake/exhaust manifold porting and injector/water jacket provisions. The engine driven supercharger is shown along with the necessary gear ratio needed to provide the required compressor pressure ratio and flow. The engine is extremely compact measuring only 39 inches across with a maximum overall length of 45.1 inches. The block itself is approximately 12 inches thick with the maximum dimension over the installed manifold piping just over 16 inches.

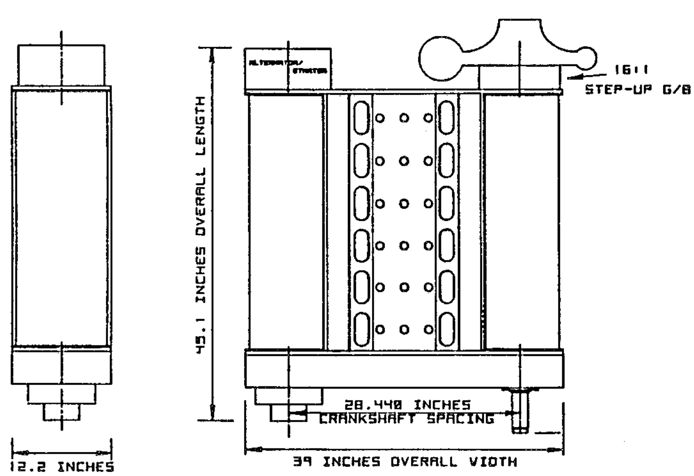

Figure 2-3 AVCD engine block dimensions.

Design features to allow cycle variability have been added to the AVCD engine to enable the engine to adjust to extreme changes in operating and ambient conditions. This ability to vary basic cycle parameters while running permits the retention of extremely high levels of thermal efficiency throughout its operating range. Table 2-2 is a list of the various engine components and operating parameters that can be altered. For extreme altitude operation, the most important variable parameters are the compression ratio and exhaust backpressure.

Table 2-2 AVCD engine variable cycle features.

⊕ AN ADVANCED VARIABLE CYCLE DIESEL ENGINE

- ⊕ VARIABLE BOOST
- ⊕ VARIABLE COMPRESSION RATIO
- ⊕ VARIABLE INTAKE/EXHAUST TIMING
- ⊕ VARIABLE INJECTIION TIMING
- ⊕ VARIABLE EXHAUST BACKPRESSURE
- ⊕ VARIABLE SCAVENGE FLOW
- ⊕ VARIABLE EXHAUST REHEAT

Table 2-3 is a description of the AVCD engine and pre-compression components used for the proposal and the initial study. The engine had 6 cylinders with 12 opposed pistons and was capable of achieving a maximum 80:1 compression ratio if required.

Table 2-3 Initial AVCD engine system characteristics.

AVCD 120000 FOOT CRUISE ENGINE DESCRIPTION

⊕ NR CYLINDERS/EFF. DISPL.	6/445
⊕ BORE/STROKE - INCHES	3.5/4.85
⊕ FLIGHT IDLE/MAX RPM	720/3600
⊕ MAX. COMP. RATIO	80:1
⊕ T/COMPRESSOR 100% P/P (2 SPOOL)	55:1
⊕ FLOW SIZE - LBS/SEC CORR.	270
⊕ TURBINE 100% P/P	81:1
⊕ FLOW SIZE - LBS/SEC CORR.	3.3
⊕ S/CHARGER 100% P/P	5:1
⊕ FLOW SIZE - LBS/SEC CORR.	6
⊕ T/COMPRESSOR STAGING	3 AX+6 AX/1CF
⊕ TURBINE STAGING	3 AX+1 AX
⊕ COOLER EFFECT. - %	85

3.0 Discussion

The technical plan for the study was broken into four tasks. Briefly, the tasks consisted of the following:

Task 1 – Definition of a suitable high altitude vehicle and AVCD engine propulsion system and a set of preliminary performance characteristics for the engine/heat rejection system.

Task 2 – Heat exchanger consultant analysis and configuration definition of a suitable system for rejecting the engine generated heat through a proscribed mission using the data developed in Task 1.

Task 3 – Incorporation of the consultant developed heat exchanger system performance maps back into the DAP code to permit more accurate AVCD engine performance estimates for the vehicle mission analysis.

Task 4 – Definition of an installation that incorporates the results of the heat exchanger work and AVCD engine system performance into the study vehicle. Out of this task came AVCD system weights, equipment arrangement and a cooling bay description. Final estimates of the vehicle TOGW and fuel loads were also made during this Task.

3.1 Task 1 – Definition of Vehicle and AVCD Engine Propulsion System

One of the first modifications of the proposal study vehicle was the increase in tankered fuel for the required 6000 n. mile range. For Task 1, it was estimated that 3000 lbs of fuel was needed (rather than the study proposal 1711 lbs) resulting in a new TOGW of 13,654 lbs. Other assumptions (see Table 3.1-1) were also made to permit the initial performance estimates of the AVCD propulsion system to be made.

Table 3.1-1 Study assumptions for initial, task 1 performance.

TASK 1 ASSUMPTIONS

10% TOTAL P_{RATIO} LOSS IN INTAKE SYSTEM

CONSTANT HX EFFECTIVENESS OF 85%

CONSTANT PROP EFFICIENCY OF 85%

VEHICLE TOGW OF 13654 POUNDS

6000 N. MILE MISSION FLOWN

STD +27 °F. @ SLTO, STD +18 °F. @ ALTITUDE

For all performance work, a Sea Level Standard + 27 Deg. F. Day was used for Sea Level Take Off (SLTO) conditions. A Standard + 18 Deg. F. Day temperature was assumed at all altitude conditions. This was done to insure a more conservative approach in sizing the engine and to provide more margin for typical operations.

The DAP charge air pressure loss assumption for the inlet manifold piping and heat exchangers was calculated as a function of intake corrected flow through the engine system. At the 100% flow point, the maximum pressure drop was 10% of the total pre-compression system pressure rise. This assumption was checked later in the study and found to be conservative.

Since boundary conditions for the heat exchanger consultant were partly a function of assumed heat exchanger system performance, a constant 85% exchanger effectiveness was assumed. Later results showed that this assumption was conservative for the heat exchanger structure eventually needed for all but the highest power levels above 85000 feet. The effectiveness assumption permitted the initial heat exchanger studies to be started but later iterations were required.

Based on other high altitude studies available in the literature (and to simplify the initial mission analysis), a constant 85% adiabatic efficiency prop was assumed. This assumption was altered later when a refined analysis of prop performance was made and required vehicle thrust levels were known more accurately.

These assumptions, along with the drag polars for the study vehicle (taken from reference 2), produced the Task 1 mission profile shown in Figure 3.1-1. Because of the increased TOGW, it was only possible to climb to 110,000 feet initially for the cruise out to the final sampling region. The vehicle time at and above 100,000 feet was more than 12.5 hours for the Task 1 mission. A 50 knot Indicated Air Speed (IAS) was maintained for climb and cruise until .7 Mn was achieved at 100,000 feet and above.

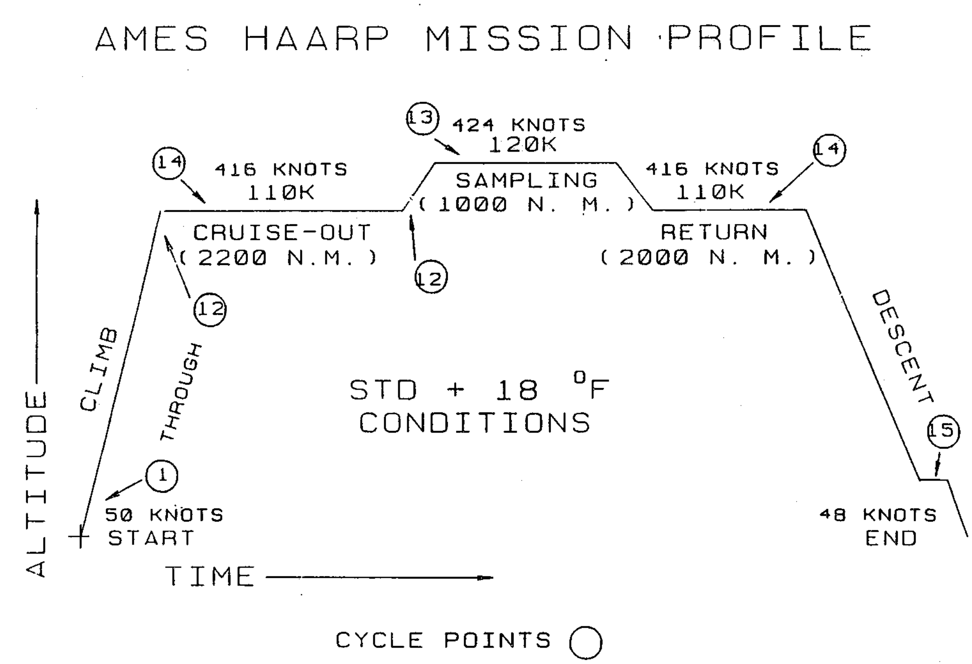

Figure 3.1-1 Initial task 1 HAARP flight profile.

A set of 15 propulsion system performance points was generated with the DAP code and supplied to the heat exchanger consultant. These boundary conditions defined all the physical conditions for the Task 2 heat exchanger analysis. The cycle data included altitude ambient conditions, vehicle Mach Number and knots, charge air pressures and temperatures and physical flow quantities throughout the system. An allowable external pressure drop of one half the dynamic ram pressure rise (q) through the convective exchangers was also established for each condition.

Figure 3.1-2 depicts some of the Task 1 DAP climb power performance projections for the AVCD engine. The mission climb profile Shaft Horsepower (SHP) was flat rated at 1200 up to 100,000 feet and then lapsed to 575 at the final cruise condition at 120,000 feet.

Net shaft thermal efficiency remained relatively constant around 36% for most of the climb with a maximum of nearly 40% being reached at 90,000 feet. (This result is due to the turbocompressor performance peaking at 90,000 feet) The Specific Fuel Consumption (SFC) also remained relatively constant at about .385 lbs fuel per horsepower hour with a minimum at 90,000 feet. (All figures include engine parasitics and a constant 10 Kilowatt (KW) vehicle power requirement.)

Due to extreme changes in ambient conditions from sea level to final cruise for the vehicle's AVCD propulsion system, multiple control methods were required for the turbocompressor. This is shown on Figure 3.1-2 with arrows indicating the combined wastegate/exhaustgate control from sea level to 70,000 feet, exhaustgate only above 70,000 feet with exhaust manifold re-heat required above 100,000 feet. (Exhaustgate control uses a variable exhaust backpressure method.)

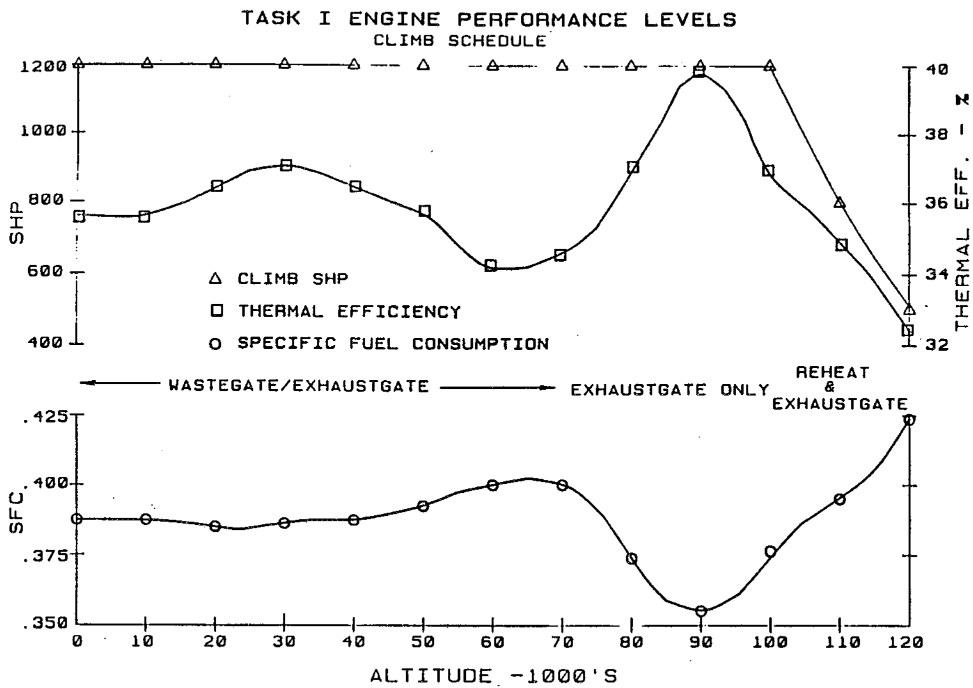

Figure 3.1-2 Initial task 1 engine mission climb performance.

Discharge temperatures from the turbocompressor and supercharger and aftercooler and intercooler charge air exit conditions are presented in Figure 3.1-3 for the Task 1 climb power profile. In Task 1, the maximum supercharger exit temperature reached just over 600 Deg. F. while the maximum turbocharger discharge temperature was 1350 Deg. F. The 85% heat exchanger effectiveness assumption resulted in relatively low intercooler/aftercooler exit temperatures. Later results from Task 2 revealed that intercooler/aftercooler temperatures would be higher at altitudes above 85,000 feet.

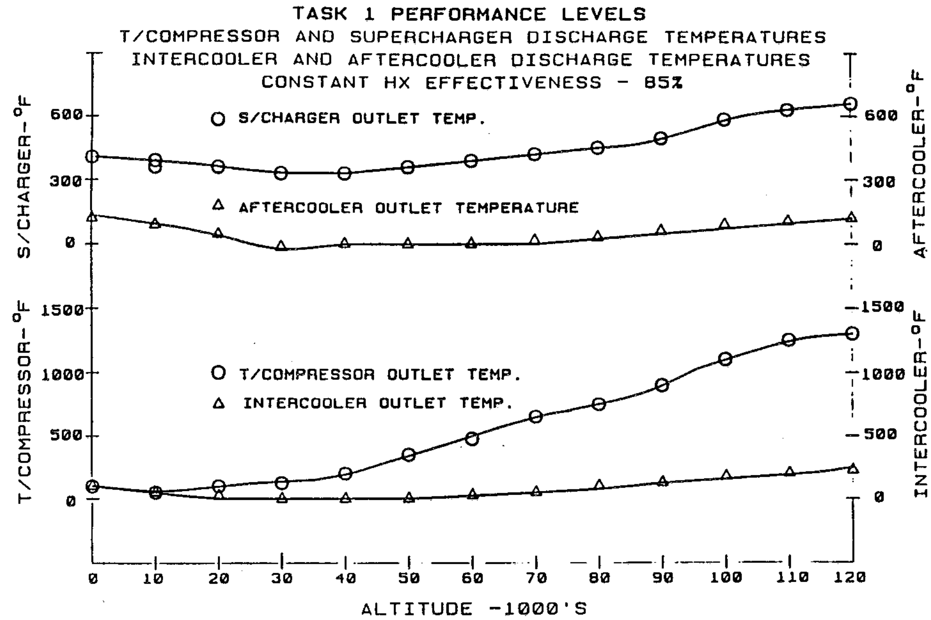

Figure 3.1-3 Initial task 1 heat exchanger results.

To help the heat exchanger consultant to trade heat exchanger pressure loss against heat exchanger effectiveness, several sensitivity studies were performed in Task 1. Figure 3.1-4 shows the engine fuel consumption sensitivity at 110,000 feet to changes in the piping/exchanger internal pressure losses. A 5% and 15% pressure drop (around the Task 1 assumed 10% drop) were evaluated to determine the marginal effect on fuel consumption at 460 and 560 SHP cruise power levels. The sensitivity study showed that tripling the internal pressure drop from 5% to 15% resulted in a .7 Gallon Per Hour (GPH) increase in fuel consumption at 460 SHP and a 1.2 GPH increase at 560 SHP.

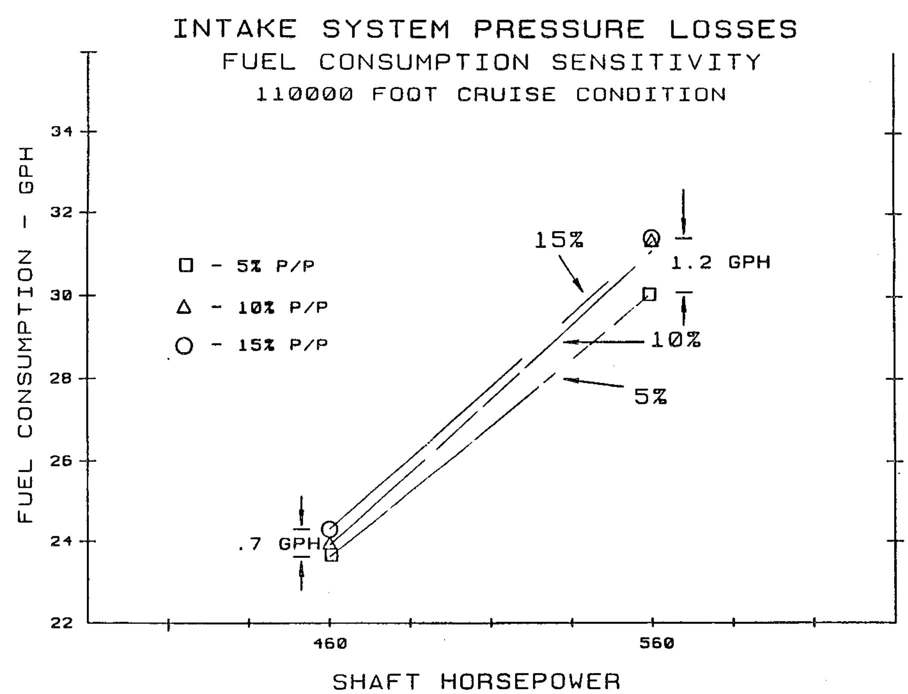

Figure 3.1-4 Fuel consumption sensitivity to exchanger pressure loss.

Fuel consumption sensitivity to changes in heat exchanger effectiveness changes is shown in Figure 3.1-5. Several combinations of aftercooler/intercooler effectiveness differences were calculated. These showed that a 5% change in intercooler/aftercooler effectiveness caused a 1.8 GPH change in fuel consumption at 460 SHP and a 2.3 GPH change at 560 SHP. Therefore, a 5% change in heat exchanger effectiveness has approximately twice the impact on fuel consumption as a 10% change in heat exchanger internal pressure drop.

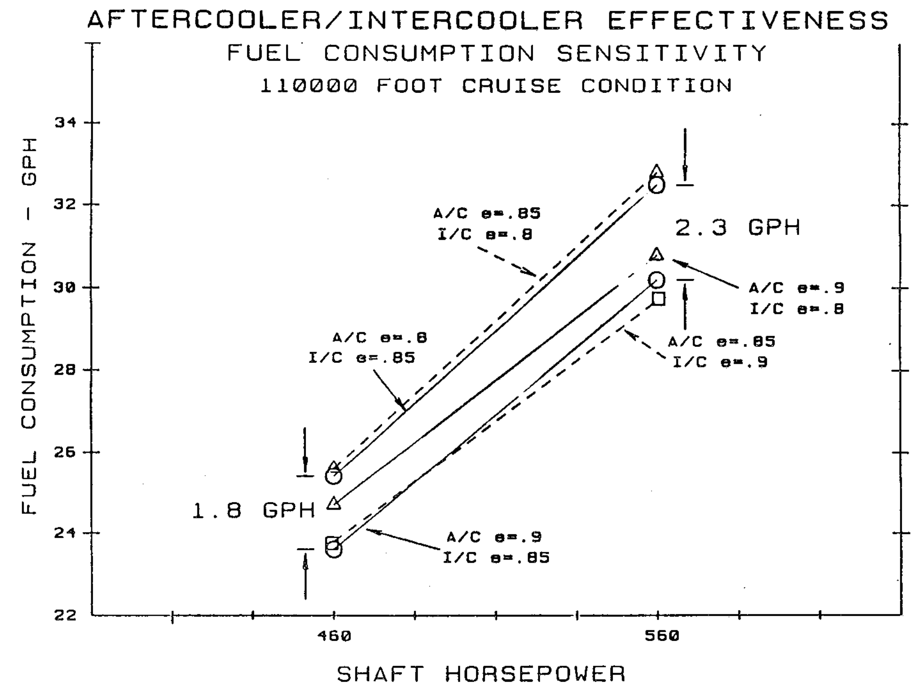

Figure 3.1-5 Fuel consumption sensitivity to exchanger effectiveness.

The Task 1 information was used by the heat exchanger consultant in Task 2 to guide heat rejection configuration studies. An initial cooling bay configuration (Figure 3.1-6) was also provided to begin installation work. This became the general layout that was altered and refined throughout Task 2, 3 and 4.

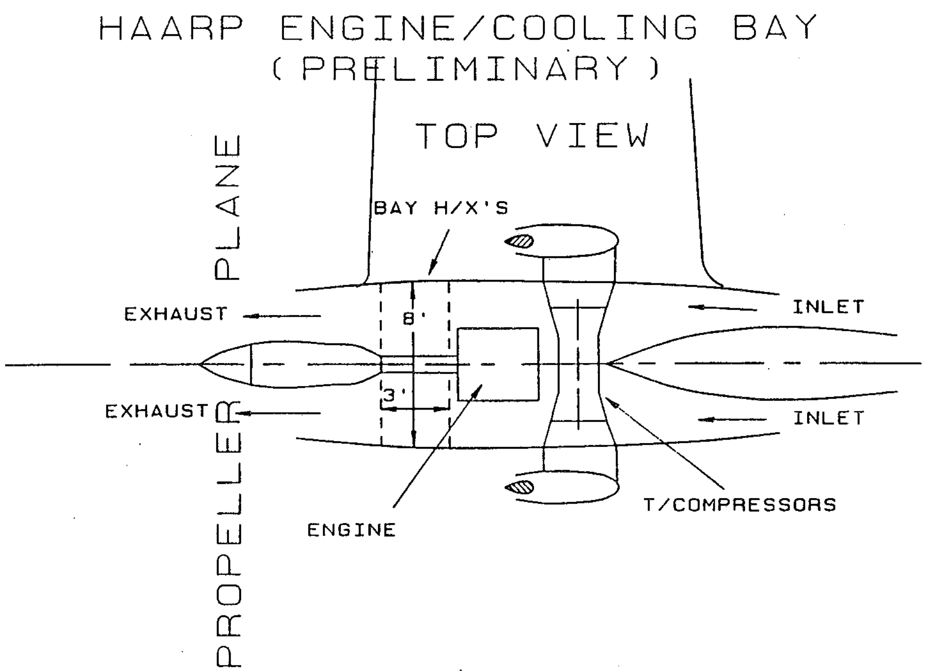

Figure 3.1-6 Initial HAARP cooling bay configuration.

3.2 Task 2 – Heat Exchanger Analysis and Evaluation

Several additional heat rejection conditions were made a part of the heat exchanger system analysis. Engine waste heat correlations were provided in British Thermal Units (BTU) for the rejection of heat from engine coolant and oil systems as a function of engine operating parameters. These correlations were as follows:

1. $Q_{coolant} = SHP * 1100 * SFC / .38$ (BTU / HR)
2. $Q_{oil} = SHP * 100$ (BTU / HR)

Gallon Per Minute (GPM) flow rate correlations were also provided as a function of engine Revolutions Per Minute (RPM) for both the coolant and oil systems. These helped set the internal heat exchanger boundary conditions. They also determined the temperature rise of the coolant and oil as it passed through the engine. These correlations were:

3. $Flow_{coolant} = 120 * RPM / 3400$ (GPM)
4. $Flow_{oil} = 20 * RPM / 3400$ (GPM)

From the consultant's prior heat exchanger experience, an upper air temperature limit of 600 Deg. F. was set for entry into the aluminum convectors. Although higher than usual practice, the low charge air absolute pressure levels at upper altitudes produced low convector stress levels. These reduced stress levels allowed a calculated 1000 Hr minimum creep life to be calculated for the aluminum convectors. This was felt to be adequate for the HAARP vehicle.

Early Task 2 work revealed that tube-fin convector designs would not provide adequate heat rejection with the allowable external pressure drop through the exchangers. However, plate-fin exchanger configurations did provide satisfactory heat rejection for the cooling bay portion of waste heat rejection. The system sizing heat rejection points were at the 110,000 and 120,000 foot climb conditions.

As a start, the cooling bay cross section was assumed to be an 8 foot by 8 foot area. This later had to be increased to a 10 by 10 foot section for adequate cooling air volume (and mass) flow for the final exchanger configuration. The limitation on total pressure drop through the convector cores (half the ram pressure rise) was found to be adequate for the final plate-fin cooling bay convector designs.

High temperature convectors made of nickel were designed and weighed to determine if they would be more weight effective than wing radiators for the turbocompressor. When adequate high temperature convectors were installed, their total weight was more than 400 lbs more than the final radiator/convector system. Bay size (and flow) would also have been larger than the final 10 X 10 area.

After the radiator/convector configuration was designed using charge air as the hot working fluid, an intermediate fluid system was evaluated to determine if it would have any advantage. By using an intermediate fluid/charge air heat exchanger, parasitic pressure work could have been reduced by pumping a fluid through the wing radiators. However, due to the lack of an intermediate fluid that could operate above 800 Deg. F., there was a large increase in radiator area (and weight) over the direct charge air radiator system.

Due to low oil system heat rejection levels compared to the primary engine coolant heat rejection (and similar operating temperatures), oil cooling was combined with the engine coolant system. This was done by passing cooled coolant through a small oil/coolant heat exchanger to remove the oil heat load.

A final cooling bay and wing radiator arrangement is shown in Figure 3.2-1. Although similar to the initial layout of Figure 3.1-6, several important changes were made. One of these was the arrangement of the convectors. It was necessary to use 320 square feet of face area to pass the necessary cooling air flow at the proper core velocity. To fit this into a 100 square foot cooling bay required that an accordion mount be used. Figure 3.2-2 is a side view and view looking aft of this arrangement. This figure also identifies the required convectors needed for the low temperature intercooler, aftercooler and engine coolant/oil functions.

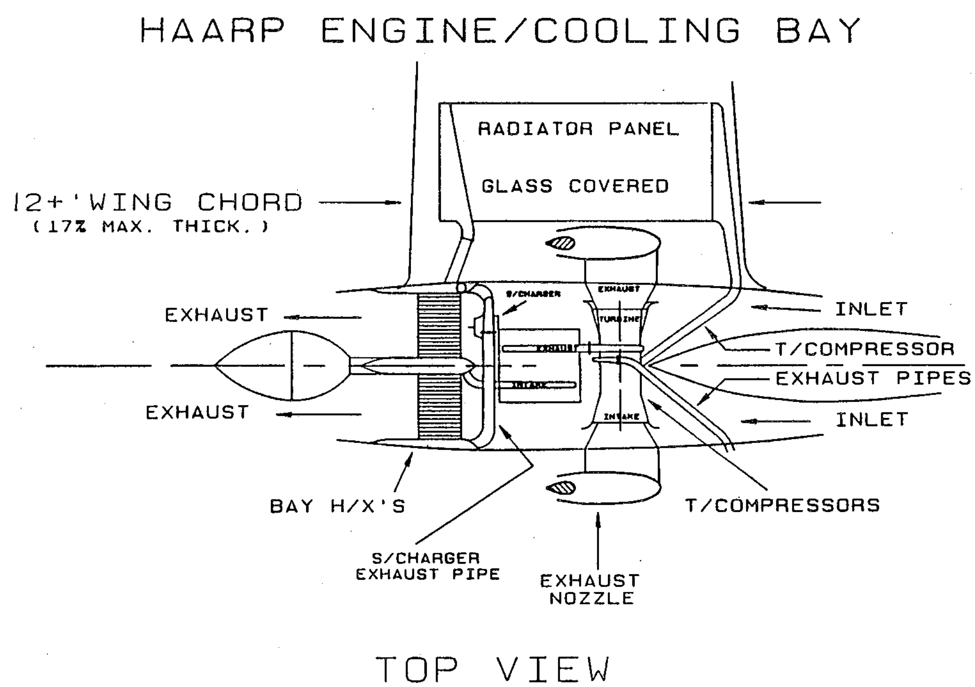

Figure 3.2-1 Final HAARP cooling bay/wing radiator arrangement.

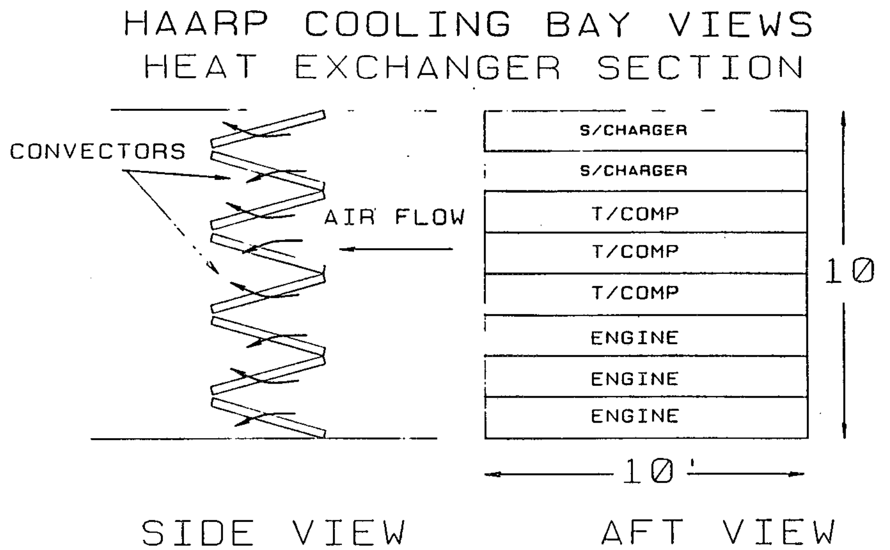

Figure 3.2-2 HAARP convector arrangement within cooling bay.

The aluminum plate-fin exchanger cores were 3 inches thick, 10 feet across and 4 feet wide. Figure A-4 shows the construction of the plate-fin core element as well as the manifolding arrangement for each core. Each of the convectors weighed 125 lbs while the entire installation (including manifolding) weighed 1060 lbs.

The wing radiator element was constructed of nickel alloy sheet and fin elements brazed together. Figure 3.2-3 illustrates a top view of the installed radiator in the wing with the direction of charge air flow. Internal fin sections were needed to convect the heat out of the hot charge air and then to conduct this heat to the surfaces of the radiator.

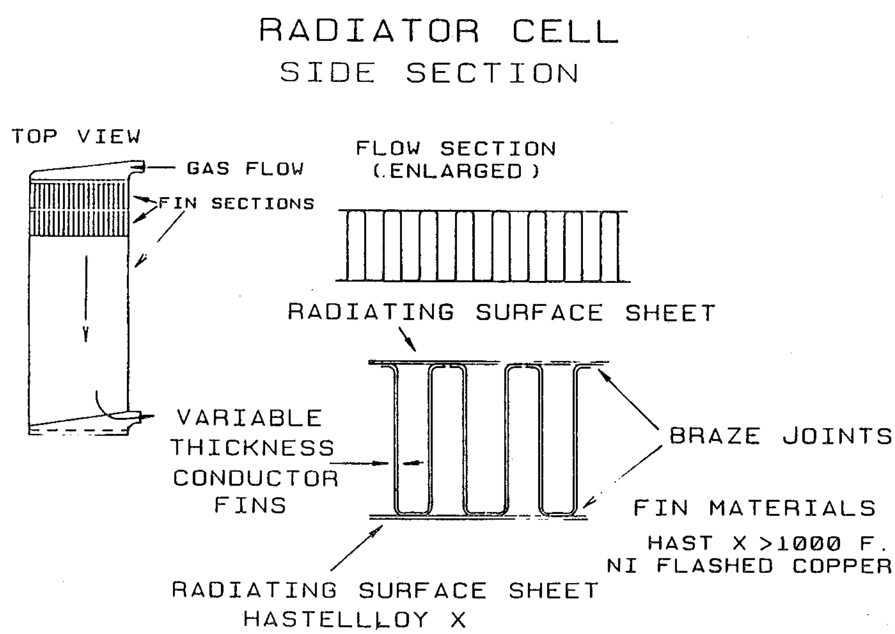

Figure 3.2-3 Wing radiator overall and detail design features.

Variable thickness fins were utilized to obtain a minimum weight radiator. At the entry to the radiator, high flux rates out of the charge air to the radiator surfaces required thicker fins. Toward the exit of the radiator, temperatures and heat flux rates are much lower and thinner fins were adequate.

Two fin materials were evaluated. One, a nearer term candidate, was a nickel alloy with a thermal conductivity (k) of approximately 40 BTU/(Foot-Deg. F.-Hr). Alloys are available that would provide this conductivity performance and good braze compatibility. A more advanced high temperature copper alloy could be flashed with nickel and would reduce the temperature gradient from the radiator interior to the radiator surface. One such alloy is now in development for use in cooled rocket nozzle applications but not yet available for general use. Application to the current radiator design would increase the surface temperature about 25 Deg. F. (and radiating efficiency) over the all nickel design.

The final radiator design consisted of two radiators, one in each wing, with a total of 200 square feet. The maximum charge air inlet temperature was 1362 Deg. F. at the 120,000 foot cruise condition. The radiating skin thickness was 10 mils thick while 4 mil thick nickel fins were spaced 7 to the inch. The total weight of the radiators was estimated to be 800 lbs.

To improve the efficiency of the radiators (and minimize weight), the radiators were installed so their bottom skins formed the wing bottom surface. The bottom surface view factor to the earth's surface was assumed to be unity.

To get the most heat rejection possible out of the least radiator, the upper surface of the wing was assumed to be constructed of a radiation transparent material. This arrangement would permit a significant amount of radiated heat to pass upward through the wing structure. Figure 3.2-4 is side view of the proposed installation showing radiation paths and shielding features to protect the internal structure of the wing. An upward view factor of .7 was assumed for this arrangement with all radiating surfaces assumed to be looking at a conservative 80 Deg. F. sink temperature.

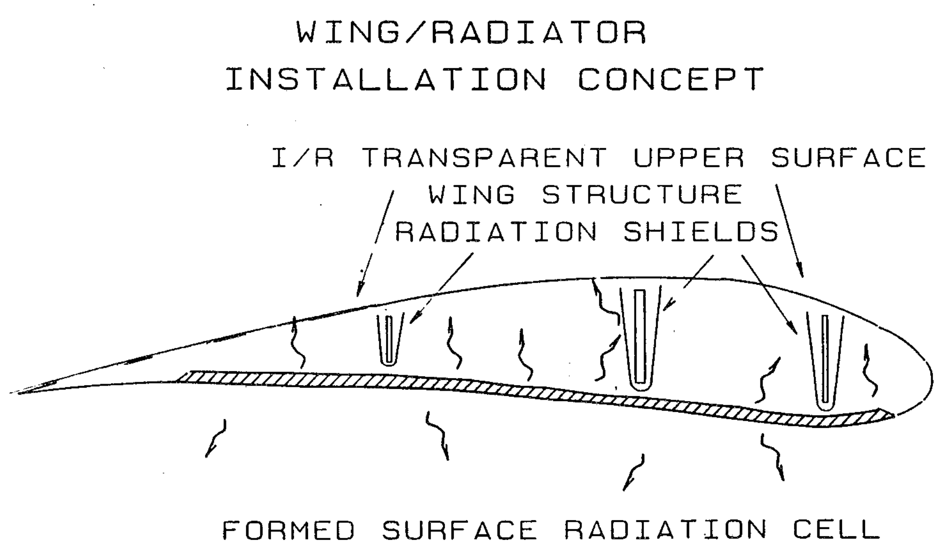

Figure 3.2-4 Wing radiator installation concept and features.

There are several glasses and materials available that are transparent in the wavelengths associated with the 1300 to 600 Deg. F. radiating surfaces associated with high altitude operation. Some potential materials would be zinc selenide, titanium dioxide or aluminum oxide. The biggest problem with available glasses is a current inability to fabricate the glasses into the large sheets needed for the upper airfoil surface.

The upper surface transparency is needed to prevent upper airfoil heating resulting in possible boundary layer separation and stall. Subsonic flight at the extreme altitudes of the HAARP mission produce high wing loadings with near stall buffet operation. Therefore, any phenomena that would weaken the boundary layer attachment must be avoided.

Figure 3.2-5 illustrates the final calculated heat exchanger system pressure drop compared to the DAP code assumption for operation throughout the mission. The lower line is the calculated pressure drop from the convecting and radiating elements for the charge air cooling. The upper line is the DAP loss assumption. The DAP code loss assumption was conservative at all altitudes compared to the final computed losses.

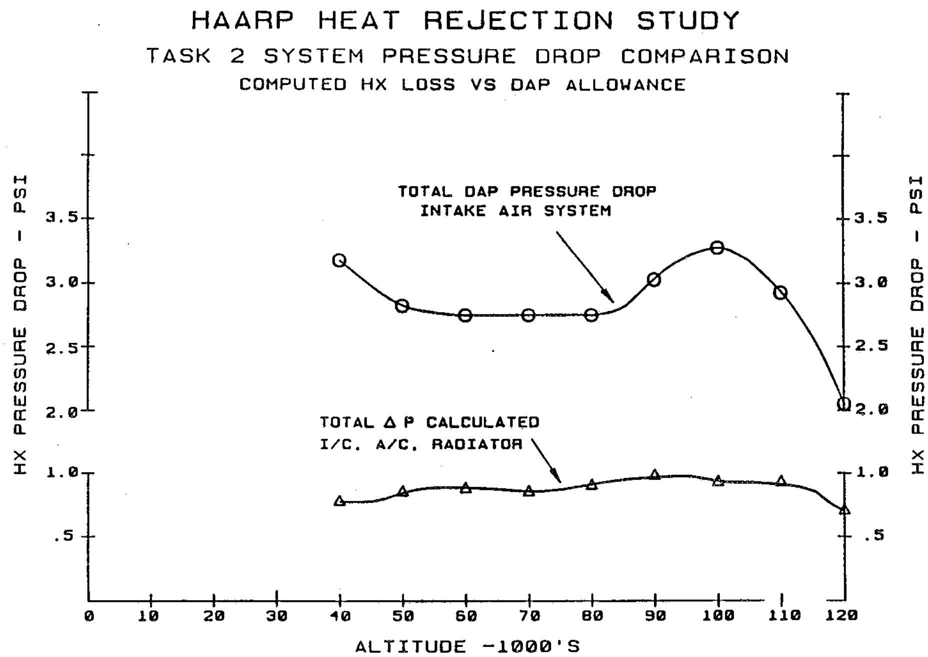

Figure 3.2-5 Actual exchanger pressure drop versus DAP assumption.

The Appendix contains a more complete description of the heat exchanger analysis portion of study. Cycle data and final exchanger performance data for all examined mission profile points are listed here along with additional descriptions of the various heat rejection elements.

3.3 Task 3 – Final AVCD Propulsion System Performance Evaluation

The main object in Task 3 was to incorporate the results of the heat exchanger assessment back into the DAP code analysis. This permitted a more accurate vehicle performance projection to be made. As part of the Task 2 work, correlations were developed for each of the main heat rejection systems as a function of vehicle operating conditions. These correlations were then incorporated into the DAP code.

Using these final exchanger assessments, the engine mission performance was reassessed. Table 3.3-1 illustrates the final changes that occurred during the study to the engine system itself. The changes that occurred are denoted by the values within the parentheses. Only minor changes resulted from the refined heat exchanger and vehicle evaluation.

Table 3.3-1 Final AVCD engine system characteristics.

AVCD 120000 FOOT CRUISE ENGINE DESCRIPTION

	PROPOSAL (FINAL)
⊕ NR CYLINDERS/EFF. DISPL.	6/445
⊕ BORE/STROKE - INCHES	3.5/4.85
⊕ FLIGHT IDLE/MAX RPM	720/3600
⊕ MAX. COMP. RATIO	80:1
⊕ T/COMPRESSOR 100% P/P (2 SPOOL)	55:1
⊕ FLOW SIZE - LBS/SEC CORR.	270
⊕ TURBINE 100% P/P	81:1
⊕ FLOW SIZE - LBS/SEC CORR.	3.3 (3)
⊕ S/CHARGER 100% P/P	5:1
⊕ FLOW SIZE - LBS/SEC CORR.	6 (5.5)
⊕ T/COMPRESSOR STAGING	3 AX+6(5)AX/1CF
⊕ TURBINE STAGING	3 AX+1(2)AX
⊕ COOLER EFFECT. - %	85 (VAR.)

Due to the steady increase in vehicle TOGW from fuel load and growth in heat exchanger system weight, two half flow size twin turbocompressors were used in place of a single large one. There were two reasons for this change. The first was a reduction of total turbocompressor machinery weight of approximately 300 lbs with the twin units over the weight of a single large unit. The second reason was the need to peak propulsion system efficiency at the 100,000 foot cruise condition.

Figure 3.3-1 is a side view of the arrangement of the twin units and AVCD engine. Each engine exhaust manifold led to its own turbocompressor but due to the cylinder porting arrangement, exhaust pressure levels were equalized. This exhaust crossflow feature of the ported AVCD engine was used to improve thermal efficiency at the 100,000 foot long duration cruise operating point. From Figure 1-2 of the final mission profile, 12.3 hrs were spent at the important 100,000 foot cruise covering nearly 4300 n. miles.

At the SHP and RPM for the most efficient operation at this critical point, a single large turbocompressor would have operated at a relatively poor efficiency. This is due to the low corrected flow levels (compared to design flow rates) through the compressor and turbine. By using a twin turbocompressor arrangement and shutting one off, the single operating turbocompressor was put on a near optimum operating point. The design features required to accomplish this are shown in Figure 3.3-1.

An exhaust diverter valve is closed to force all engine exhaust through just one unit. A check valve installed on the shut down unit prevents back flow through the idle compressor. The equalizing manifold permits charge air flow from the single turbocompressor to go out to both sides of the intercooler system. This produces a high level of cooling with the lowest possible pressure losses.

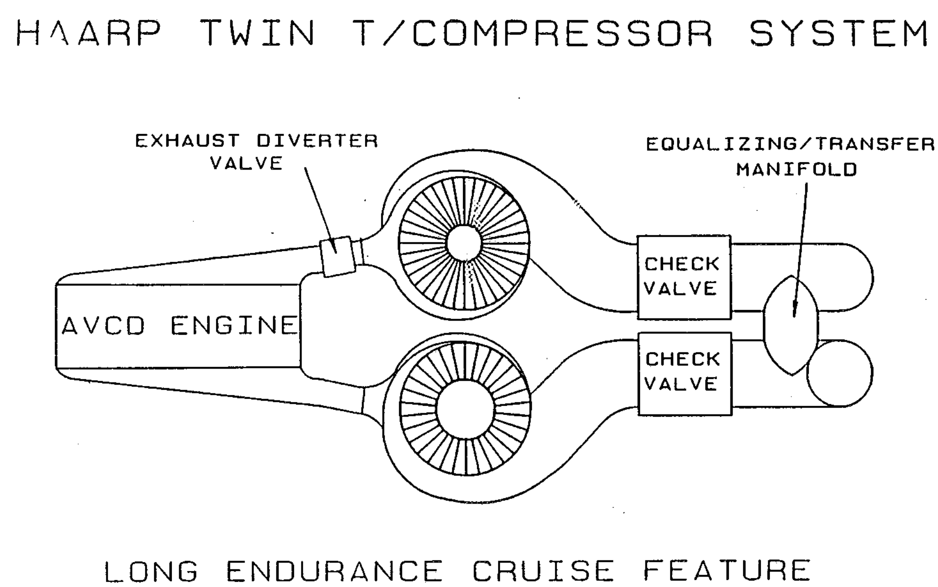

Figure 3.3-1 AVCD engine and twin turbocompressor arrangement.

A comparison of single versus dual turbocompressor operation is shown in Figure 3.3-2. Steady state cruise operation at 80,000, 90,000 and 100,000 feet was computed. The use of one unit is much more efficient than operating with both units at these endurance cruise conditions. Significant reductions in engine SFC and RPM levels were achieved with the single unit. This permitted net shaft thermal efficiency levels of more than 60% to be developed for both the 90000 and 100,000 foot cruise conditions.

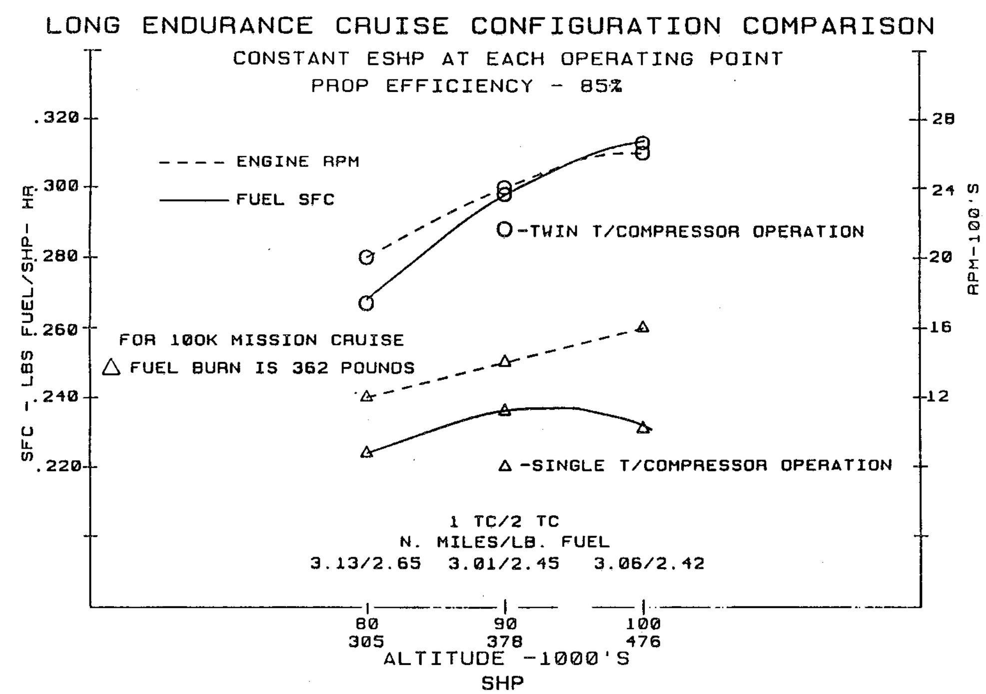

Figure 3.3-2 Long endurance cruise performance comparison.

Also shown at the bottom of Figure 3.3-2 is a comparison of n. mile per lb of fuel for single half size or full flow size unit operating modes. At the selected 100,000 foot cruise, the single half size turbocompressor mode resulted in a 26% improvement in fuel economy over a full flow size turbocompressor operation. This improvement converted into a 362 lb reduction in fuel consumption over the whole mission. Altogether, the twin turbocompressor concept was responsible for a nearly 700 lb reduction in TOGW over a more conventional arrangement and operation.

When the refined heat exchanger performance effects were fully incorporated, it was not possible to make the final mission midpoint climb above 115,000 feet. This was due to increased vehicle weight and hotter than desired charge air temperatures to the engine. This problem was eliminated by using a final fuel/air cooler downstream of both the intercooler and aftercooler convective units. At the start of final climb to maximum altitude, more than 1400 lbs of fuel remained in the vehicle's fuel tanks. This fuel was pumped through the fuel/air coolers above 115,000 feet to serve as a heat sink.

By recirculating the wing tank fuel, an additional 50 Deg. F. reduction in engine charge air temperature was achieved. This allowed the required 625 engine SHP to be developed to reach the final 120,000 foot cruise portion of the mission. The use of the fuel as a heat sink resulted in an eventual 160 Deg. F. rise in fuel temperatures assuming no wing to ambient heat transfer. A rise in fuel supply temperatures of this amount would not adversely affect the engine injectors or fuel quality. Figure 3.3-3 shows where the fuel air coolers (round circles with F/A within) were installed on the AVCD extreme altitude system. Use of the recirculating fuel/air coolers permitted an additional 100 SHP to be developed over that possible with the installed convective and radiative coolers.

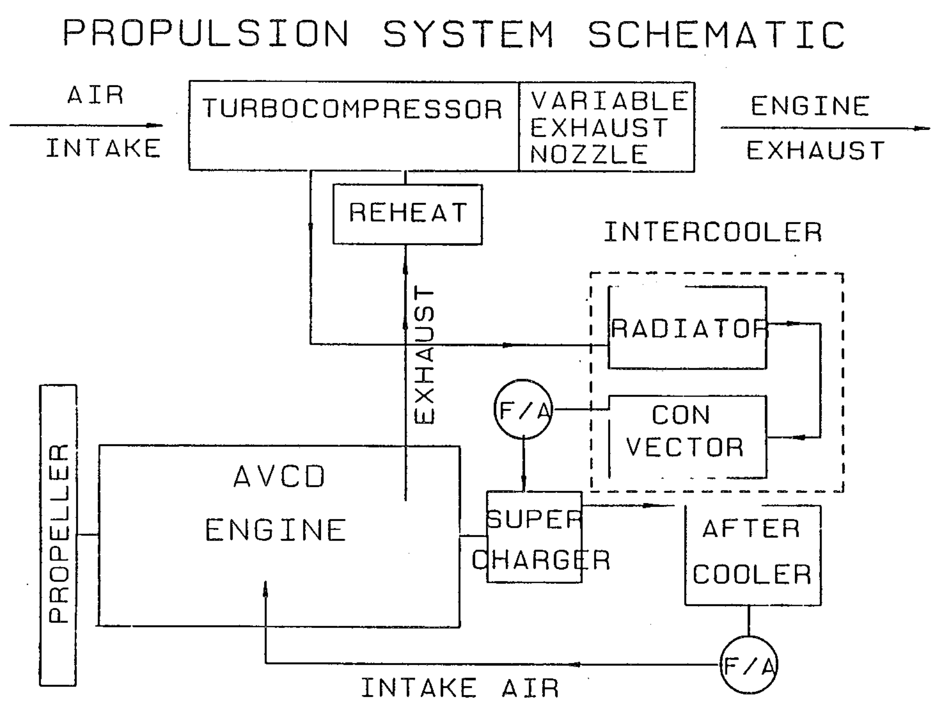

Figure 3.3-3 Final AVCD extreme altitude system schematic.

Figure 3.3-4 presents the final altitude dependent exchanger effectiveness levels achieved based on Task 2 results and use of the fuel air coolers at the final mission altitude. A pseudo effectiveness (based on ambient cooling air and charge air temperatures) is also shown for the radiating portion of the intercooler. As the cooling bay convector's effectiveness degrade with increasing altitude, the radiator's effectiveness improves since the charge air entry temperatures are increasing with altitude. This tendency of the radiator performance to improve as the convector performance degraded produced a relatively constant level of charge air cooling over the last 10,000 foot of climb.

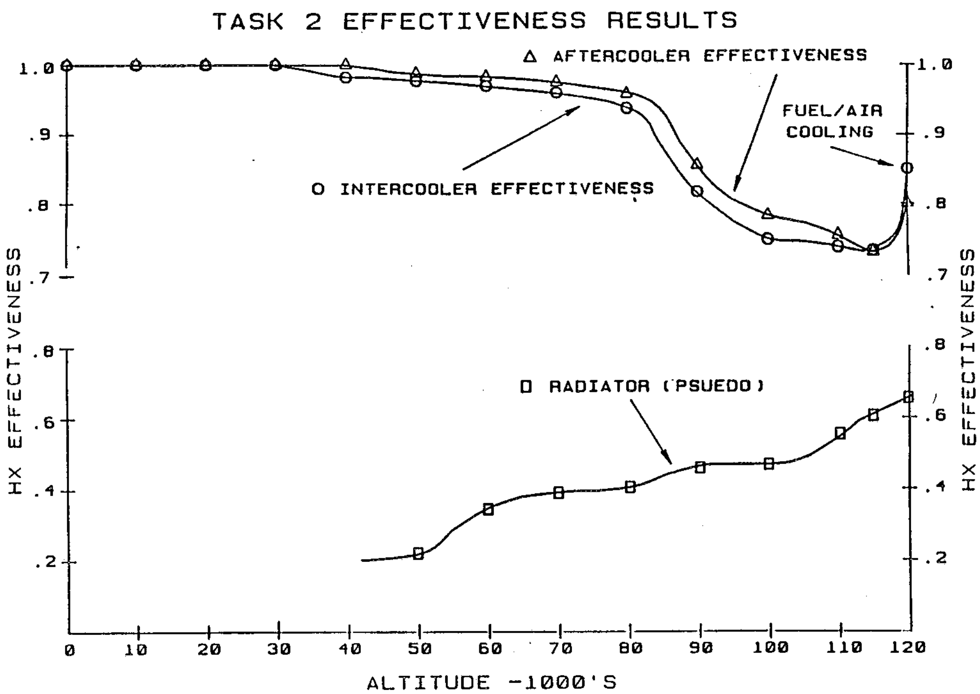

Figure 3.3-4 Final exchanger mission effectiveness.

Figure 3.3-5 presents relative heat rejection from each of the main charge air cooler elements with altitude. The increasing heat load handled by the radiator element with altitude is readily apparent. At the peak cruising altitude, the radiating heat exchanger is rejecting more charge air heat than all the other installed exchangers together.

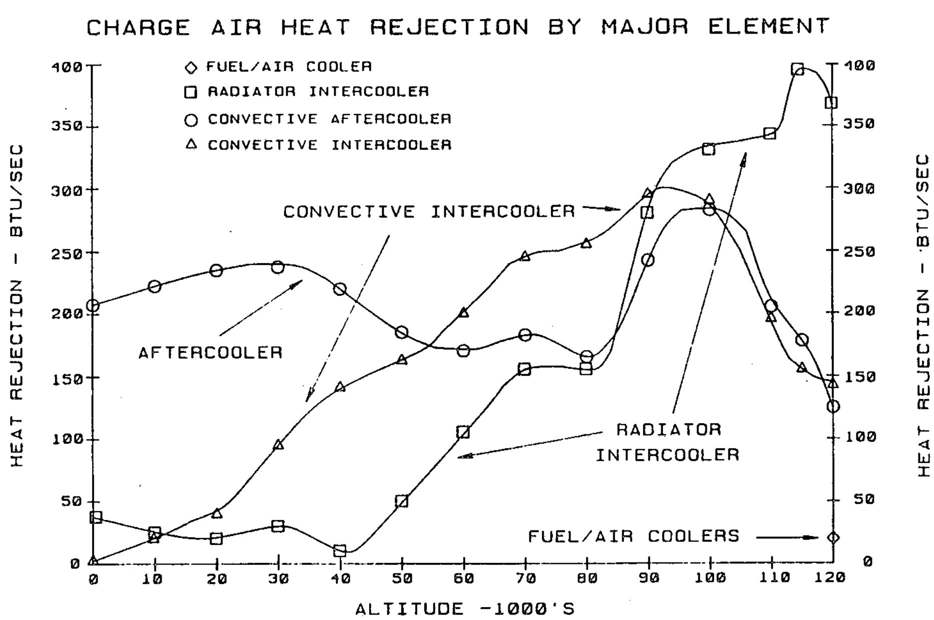

Figure 3.3-5 Charge air mission profile heat rejection by exchanger.

Figure 3.3-6 is the accompanying description of the charge air temperatures into and out of each of the charge air cooler systems. With the fuel/air cooling, the supercharger outlet temperatures were kept below 600 Deg. F.

even at the maximum sampling altitude. Radiator discharge temperatures never rose much above 500 Deg. F. so the 600 Deg. F. upper limit into the lightweight aluminum convectors was satisfied in the final heat exchanger evaluation.

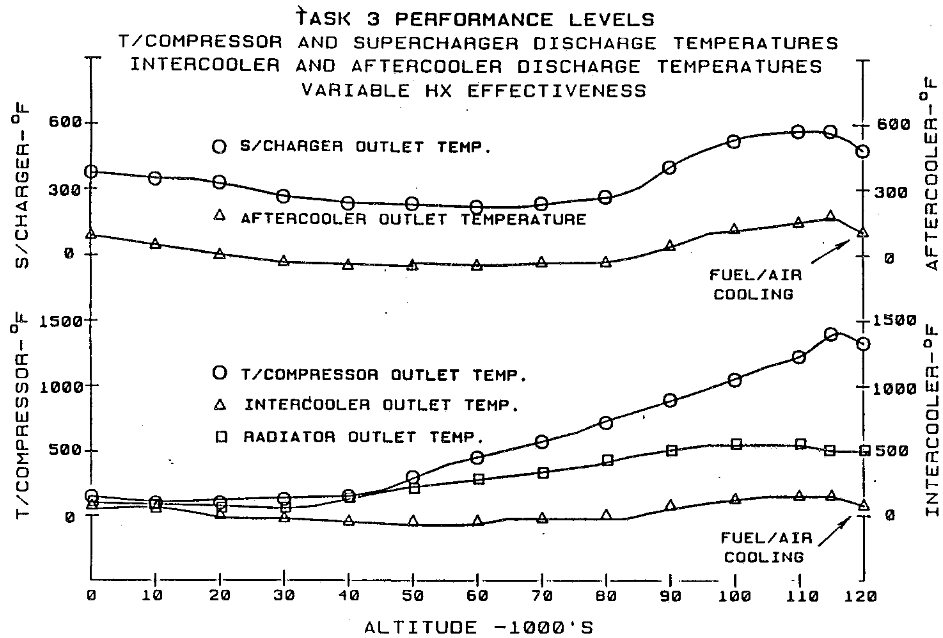

Figure 3.3-6 Charge air mission profile inlet/outlet temperatures.

3.4 Task 4 – AVCD Propulsion System Installation Characteristics

The AVCD propulsion system installation evaluation was performed in Task 4. Figure 1-1 showed the overall aircraft configuration and vehicle characteristics. Figure 3.2-1 illustrated a close-up top view of the final cooling bay arrangement and wing radiator locations. The twin turbocompressors were installed over/under each other and perpendicular to the cooling bay flow. This orientation was selected to forestall foreign object damage to the units or any significant impact ice build-up if the vehicle flew through freezing precipitation at lower altitudes.

The convectors were installed accordion style just aft of the AVCD engine with the pusher prop power shafting passing through the convector stack. Twin side inlets on the forward fuselage provided the needed cooling bay air flow while variable flaps aft of the convector stacks controlled the total bay air flow. Each turbocompressor exhausted into its own variable area exhaust jet nozzle as shown in Figure 3.2-1. Each turbocompressor's discharge air was routed to its own individual wing mounted radiator that radiated down and up through a radiation transparent glass upper airfoil surface.

Since cooling air volume flows and bay/convector dimensions were so large, the total cooling bay drag at extreme altitude was of concern. Therefore, a net cooling bay drag computation was performed for the final 120,000 foot cruise condition with results shown in Table 3.4-1. The vehicle and ambient conditions are shown at the top of the table. A 29 square foot inlet area was needed to provide the necessary capture area for the cooling bay volume flow rate of 21270 cubic feet per second.

Stagnation pressure recovery over the inlet area resulted in a total inlet ram drag of 109 lbs. A cooling bay pressure loss of 5% of the total inlet pressure was assessed for friction etc. A pressure drop through the convector stack of half the stagnation pressure rise (or .013 lbs per square inch for this case) was charged. In passing through the convector stack, there was an average cooling bay air temperature rise of 254 Deg. F. Subtracting the pressure losses resulted in a bay cooling air exhaust pressure of .0756 lbs/square inch. This provided an exhaust nozzle pressure ratio of 1.12 over the external atmospheric ambient conditions. Even assuming a very poor nozzle thrust coefficient of .9 resulted in a recovered cooling air thrust of 101 pounds. Thus, the net cooling bay drag was only 8 pounds for this worst case condition.

Table 3.4-1 Worst case cooling bay drag assessment.

COOLING BAY DRAG ASSESSMENT

120K CRUISE, .7 MN (433 KNOTS)

434^{0}R T_{AMB} . .067 PSIA P_{AMB}

INLET RECOVERY PRESS./TEMP.	.093 PSIA/497^{0} R
INLET AREA/FLOW	29 FT^{2} /21270 FT^{3} /SEC
INLET RAM DRAG	109 LBS
BAY FLOW AREA/ Δ P	100 FT^{2}/5% P_T
HX AIR ΔP/ TEMP. RISE	.013 PSI/+254^{0} F
NOZZLE INLET TOTAL PRESS./P_{RATIO}	.0756 PSIA/1.12
BAY NOZZLE C_T /THRUST	.9/101 LBS

NET COOLING BAY DRAG - 8 LBS

The prop and drive system produced some surprises not expected when the study began. First, the prop performance assessment pointed out the need for a two speed prop transmission. This was due to the need for nearly constant prop physical speeds at both the 100,000 foot cruise and the subsequent climb to 120,000 feet. Very low engine speeds were used at the 100,000 foot endurance cruise point but very high engine speeds were needed for the last 20,000 foot climb. Table 3.4-2 points out the major characteristics of the prop and gearbox arrangement for the vehicle. The required prop was also much larger than anticipated due to the steady growth in vehicle weight during the study over pre-study assumptions and the resulting growth in engine/prop SHP levels.

Table 3.4-2 Final prop and gearbox description.

FINAL PROP AND DRIVE CONFIGURATION

TWO SPEED OPERATION NEEDED FOR CLIMB AND LE CRUISE

OVERALL G/R OF 7.56:1 FOR CLIMB AND 120K CRUISE

OVERALL G/R OF 5.42:1 NEEDED FOR LE CRUISE AT 100K

FINAL 5-BRANCH PLANETARY G/BOX OF 4.75:1 G/R SELECTED

PRIMARY ENGINE REDUCTIONS OF 1.59 AND 1.14:1 REQUIRED

PROP ACTIVITY FACTOR OF 144 WITH MAX RPM OF 449

PROP DIAMETER OF 34 FEET WITH 12 BLADES

GRAPHITE COMPOSITE CONSTRUCTION ASSUMED

TIP SWEEP WILL BE REQUIRED FOR ASSUMED EFFICIENCY

Figure 3.4-1 is a schedule of prop efficiency, engine SHP, exhaust nozzle Equivalent Shaft Horsepower (ESHP) and vehicle drag ESHP during the climb from Sea Level to the final sampling altitude. The variable area jet nozzles contributed a significant propulsive effort during the upper altitude portion of the flight. The drag ESHP crosses the available propulsive ESHP at the final cruise altitude. For the selected prop, climb adiabatic efficiency peaked at 110,000 feet and fell off to a predicted 84% at the final cruise altitude. The prop characteristics were projected from data in Reference 2.

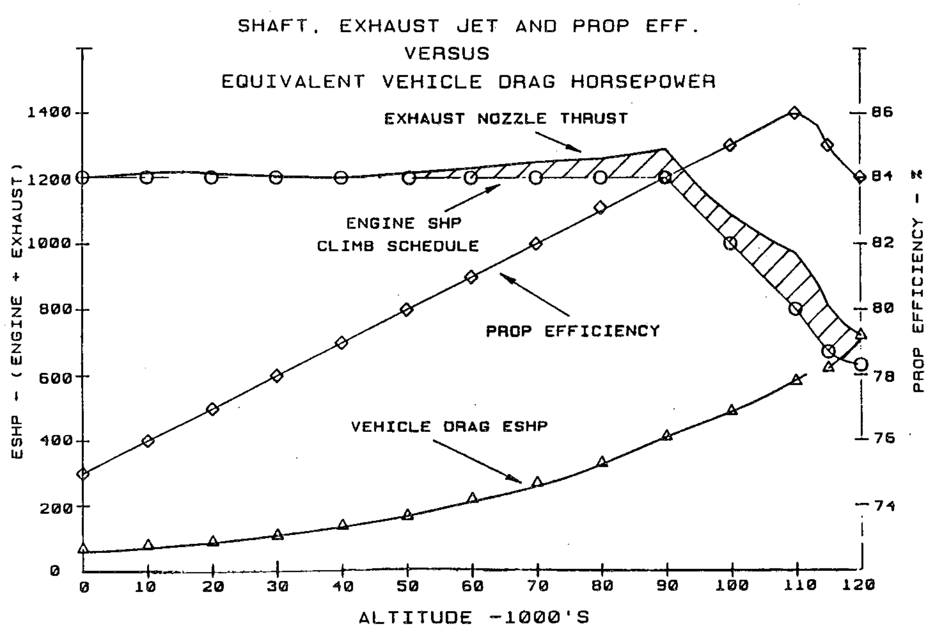

Figure 3.4-1 Mission drag/prop eff. versus shaft/exhaust jet ESHP.

A final roll up of the total propulsion system weight for the study vehicle is shown in Table 3.4-3. The four largest items were the engine, the twin turbocompressors, the heat exchangers and the prop. The final propulsion system weight approached 3 tons and was nearly twice as heavy as the initial study proposal weight projection. This system weight increase plus increase in range to 6000 n. miles required very careful mission minimum drag flight speed planning and multiple features to improve fuel consumption.

Table 3.4-3 Final propulsion system component weights.

FINAL PROPULSION SYSTEM WEIGHTS

COMPONENT	POUNDS
INST. ENGINE	1319
T/COMPRESSOR	1130
S/CHARGER & DRIVE	125
TORQUE SHAFT & GEAR	146
PIPING	160
RADIATOR	800
CONVECTORS (8)	1060
BAY AIR CONTROL	42
PROP	1075
TOTAL	5857

Figure 3.4-2 illustrates how the AVCD engine was flown and changed its operating parameters to achieve the exceptional thermal efficiencies projected for this mission. Compression ratio was varied from approximately 19:1 at Sea Level to over 40:1 at the top of climb at 120,000 feet. This permitted the engine to stay on its thermodynamic design point even as the turbocompressor/supercharger performance fell off at the final sampling altitudes. Due to this ability to vary its cycle, nearly uniform climb schedule thermal efficiencies were maintained by the AVCD engine all the way to 120,000 feet.

When the engine went off its climb schedule to the high efficiency 100,000 foot endurance cruise mode, the compression ratio was raised to 54:1. By doing this, near optimum combustion chamber conditions were maintained at the low RPM and pre-compression available under those conditions. Net shaft thermal efficiencies (all parasitics and 10KW instrument power included) of more than 60% were achieved for this cycle point. Only a variable cycle diesel engine with exhaustgate control of the turbocompressor can achieve net shaft efficiencies of this level under these extreme conditions.

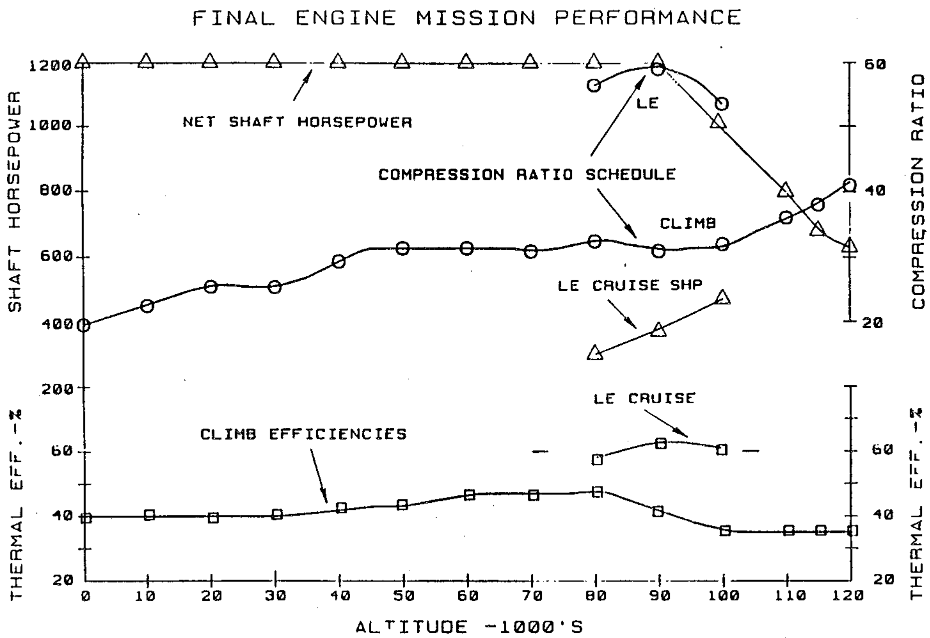

Figure 3.4-2 AVCD mission profile efficiency, SHP and C/R schedule.

4.0 Conclusions

NASA performance goals for a HAARP vehicle sampling at greater than 100,000 feet for at least 3 hrs. with a 6000 n. mile range carrying a 1000 pound instrument payload can be attained. Use of an advanced wing mounted high temperature radiator in series with a light alloy convector array was the most weight efficient method of rejecting engine waste heat. The AVCD propulsion system and an airframe based on Reference 2 results was able to carry out the extreme altitude/range mission while providing subsonic atmospheric sampling capability.

The required airframe and engine will need to be large compared to those of existing RPV vehicles. The study vehicle had a 273 foot wing span, a 1200 SHP rated engine and weighed more than 16,000 lbs at take-off. The extreme importance of weight in achieving ultra-high altitude operation was repeatedly demonstrated in assessing the effect of added equipment and fuel. Paraphrasing the old real estate saw, "The three most important design considerations for a HAARP are weight, weight and weight."

As far as study assumptions are concerned, there are several areas of concern. First, there are no manufacturing methods currently available to form available infra-red transparent glass into the large integral surfaces needed for the wing radiators. Second, performance projections for the cooling bay convectors are not backed up by test data since cooling bay ambient conditions are so far out of the experience base.

The prop performance levels used for the study are based on analytic projections rather than test results. At some operating points, it was necessary to extrapolate prop performance beyond available data. Therefore, achievement of the study results would be highly dependent on the actual prop performing at near study estimates.

Another technology shortfall is in the area of suitable large high pressure exhaust-driven turbocompressors. To reach the mission altitude goal will require efficient current technology compressor and turbine groups. The successful application of backpressure control rather than wastegate control at higher altitudes will also be needed. Although performance levels projected for the turbocompressor components have been de-rated at higher altitudes for Reynolds' Number effects, no test data exists to confirm the component data used above 80,000 feet altitude.

It is possible to develop the various critical technology pieces needed to achieve the mission goals over the next 3 to 5 years. It would require significant effort and investment in the engine, vehicle and turbocompressor technology. However, full achievement of the NASA HAARP goals would permit systematic high altitude sampling of the ozone layer. This sampling would provide credible information on the true cause of the variation of ozone in the upper atmosphere. This information could then be used for cost effective solutions if the data indicates there are social actions that can affect ozone depletion.

Dryden Flight Research Facility
National Aeronautics and Space Administration
Edwards, California, October 1991

APPENDIX

A DETAILED STUDY OF THE HEAT REJECTION SYSTEM FOR AN EXTREME ALTITUDE ATMOSPHERIC SAMPLING AIRCRAFT

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1.0 Purpose

1.1 High Altitude Flight

This report describes the design approach to a set of heat exchangers for application in a high altitude, long range aircraft using an advanced diesel engine. The heat exchange problem falls outside the realm of design experience since the gas turbine systems which presently operate at these altitudes do not require heat exchangers.

The design conditions specify a flight plan which has a climb from sea level to $\sim$100,000 feet, a long range cruise at this altitude, and a data sampling period at 100,000 ft to 120,000 ft. The elapsed time is approximately 14 hours, with a two hour sampling period. The design flight operating conditions are given in Table A-1.

Table A-1 DieselDyne cycle data reference sheet.

Point	SHP	RPM	Altitude ft	Mach No	SFC lb/hr-hp	Charge lb/sec
1	1200	2800	0	0.09	0.362	3.33
2	1200	3000	20000	0.12	0.357	3.17
3	1200	3000	40000	0.20	0.332	3.05
4	1200	2800	60000	0.30	0.309	2.78
5	1200	2600	80000	0.40	0.297	2.62
6	1200	3000	90000	0.44	0.336	2.82
7	1200	3400	100000	0.50	0.389	3.14
8	1200	3400	110000	0.60	0.397	2.17
9	1200	3400	115000	0.70	0.403	1.84
10	625	3400	120000	0.70	0.399	1.60
11	1200	1300	80000	0.46	0.238	0.46
12	800	1500	90000	0.50	0.226	0.63
13	500	1600	100000	0.60	0.226	0.71
14	460	3200	110000	0.70	0.388	1.36
15	120	3400	115000	0.70	0.417	1.40

1.1.2 Heat Loads

There are four sets of heat loads which must be removed:

1.1.2.1 Engine Cooler

The engine cooling fluid is a 50% mixture of ethylene glycol and water, which must be delivered to the engine at a temperature of 250 °F or below. Engine coolant heat loads are given on the left hand side of Table A-2. Heat loads and flows are determined from the following equations supplied by DieselDyne:

Flow: 120*RPM/3400 gpm
Heat Load: 1100*SHP*SFC/.38 Btu/hr

Table A-2 Engine coolant and lubrication systems.

	Engine Coolant System			Engine Lubric System		
Point	Heat Load Btu/hr	Flow gpm	TempRis F	Heat Loa Btu/hr	Flow gpm	TempRis F
1	1.26E+06	99	28.7	1.20E+05	16.5	35.1
2	1.24E+06	106	26.5	1.20E+05	17.6	32.7
3	1.15E+06	106	24.6	1.20E+05	17.6	32.7
4	1.07E+06	99	24.5	1.20E+05	16.5	35.1
5	1.03E+06	92	25.4	1.20E+05	15.3	37.8
6	1.17E+06	106	24.9	1.20E+05	17.6	32.7
7	1.35E+06	120	25.4	1.20E+05	20.0	28.9
8	1.38E+06	120	26.0	1.20E+05	20.0	28.9
9	1.40E+06	120	26.3	1.20E+05	20.0	28.9
10	7.22E+05	120	13.6	6.25E+04	20.0	15.0
11	8.27E+05	46	40.7	1.20E+05	7.6	75.5
12	5.23E+05	53	22.3	8.00E+04	8.8	43.6
13	3.27E+05	56	13.1	5.00E+04	9.4	25.6
14	5.17E+05	113	10.3	4.60E+04	18.8	11.8
15	1.45E+05	120	2.7	1.20E+04	20.0	2.9

1.1.2.2 Oil Cooler

The engine lubrication fluid analysis is based on the properties of SAE 30 oil. Again, the delivery temperature to the engine must be maintained below 250 °F. Engine coolant heat loads are given on the right hand side of Table A-2. Heat loads and flows are determined from the following equations supplied by DieselDyne:

Flow: 20*RPM/3400 gpm
Heat Load: 100*SHP

1.1.2.3 Intercooler

The intercooler receives the engine charge air from the turbocompressor at a low pressure and a high temperature and delivers cold air to the supercharger. Although these temperatures are low during climb, they peak at almost 1300 °F at maximum altitude. Thus the maximum heat load occurs at the most crucial altitude when heat sink capacity is least. Nominal inlet temperatures and pressures for the intercooler are given on Table A-3.

Table A-3 Intercooler and aftercooler nominal inlet temperatures and pressures.

	Heat Exchanger Operating Conditions				
	Intercooler		Aftercooler		
Point	P in psia	T in F	P in psia	T in F	Charge lb/sec
1	15.08	90	49.7	398	3.33
2	9.55	105	38.8	326	3.17
3	7.13	164	31.7	256	3.05
4	7.81	408	31.0	248	2.78
5	9.39	729	32.1	263	2.62
6	8.74	901	33.8	408	2.82
7	8.53	1137	37.7	536	3.14
8	6.45	1255	27.5	567	2.17
9	6.05	1335	23.8	563	1.84
10	4.67	1387	19.7	494	1.60
11	6.72	741	11.3	83	0.46
12	6.69	825	12.6	119	0.63
13	7.08	1022	13.9	152	0.71
14	4.49	1026	16.6	515	1.36
15	4.48	1139	17.7	556	1.40

1.1.2.4 Aftercooler

The aftercooler receives the engine charge air from the supercharger and delivers it to the engine. The pressure is moderately high and temperatures remain in the area between 300 and 600 °F over the entire flight plan. Nominal inlet temperatures and pressures are given on the right-hand side of Table A-3.

1.2 Design Approach

1.2.1 Overall

Figure A-1 shows a schematic of the engine airflow path through the rotating equipment and the heat exchangers. Very high temperature air from the turbocompressor enters the radiant section of the intercooler, wherein it is cooled to below 600 °F. Cooling is completed using the ram air heat sink in the convector portion of the intercooler. After the air leaves the supercharger, it passes through the ram air cooled aftercooler before entering the turbine. Fuel from the tanks can be used to trim the intercooler and aftercooler temperatures'for brief periods of time.

The engine cooling water rejects heat to another bank of ram air cooled heat exchangers. This fluid is used to cool the engine lubricant, which represents a relatively small heat load, in a liquid-to-liquid heat exchanger.

1.2.2 Radiant Intercooler

The radiant heat exchanger is placed in the wing to take maximum advantage of the line-of-sight to the surroundings. The exchanger consists of two ten foot wide, three inch high ducts which form the lower surface of the inboard section of each wing. The air from the turbocompressor is introduced through a manifold on the leading edge of this duct and flows axially through the ten foot long finned radiant section. It is then collected in an exhaust manifold on the back of the duct. Figure A-2 shows the plan view of one radiator.

The ducts are formed of stainless steel skins 10 mils thick to minimize weight penalty. The outside of these skins are painted black to minimize radiant transfer. The fins are formed of 4 mil nickel, seven fins per inch, offset each half inch to enhance heat transfer. Nickel was selected because it has the best thermal conductivity of any high temperature material.

The skin of the top surface of the wing forms a radiation window. The material selection is based on the ability to transmit radiation in the infrared regime (black body temperatures of 500 to 1300 °F) wherein the system will operate. The potential choices are zinc selenide, titanium dioxide, and aluminum oxide.

The structural components of the wing are shielded from the duct radiation. The windows are supported by an aluminum grid structure to minimized thickness. Figure A-3 shows the finned duct structure and window.

1.2.3 Ram Air Coolers

1.2.3.1 Plate Fin Heat Exchanger Frames

The ram air cooling system design is based on eight individual aluminum plate fin heat exchangers (frames). Each frame is ten feet high, four feet long, and three inches deep. The ten foot stack contains repeating layers of a plate, a half inch high corrugated sheet with the axis in the three inch direction, a second plate, and a lanced offset corrugated sheet with the axis in the four foot direction. The fins on the fluid side are tailored to match the properties of the individual fluid involved.

Ram cooling air enters the ten by four foot face. The fluid being cooled is distributed to the individual passages by a manifold along one ten foot edge. After flowing through the lanced offset fin, it is collected in a manifold on the other edge. Figure A-4 shows the construction.

1.2.3.2 Bay Cooling

All eight of the ram air cooling heat exchanger frames are arranged in one block in the center of the bay of the aircraft. The group of heat exchangers shared a face area of 100 square feet, through which the cooling air entered. The individual heat exchangers are placed on an angle so that they shared inlet and outlet manifolds for the ram air. The overall accordion configuration is shown in Figure A-5.

Three of the heat exchanger frames are required for the intercooler, located on the right-hand side of the bay facing aft. A single duct across the bottom of the bay at the rear connected the outlets of the two radiant wing intercoolers. The three inlet manifolds for the intercooler frames are connected to this duct. The outlet manifolds also fed into a single duct which led to the supercharger.

The two central heat exchanger frames are required for the aftercooler. The hot gas duct from the supercharger feeds the manifolds at the top rear bay. The duct to the engine is taken off the top of the front manifolds.

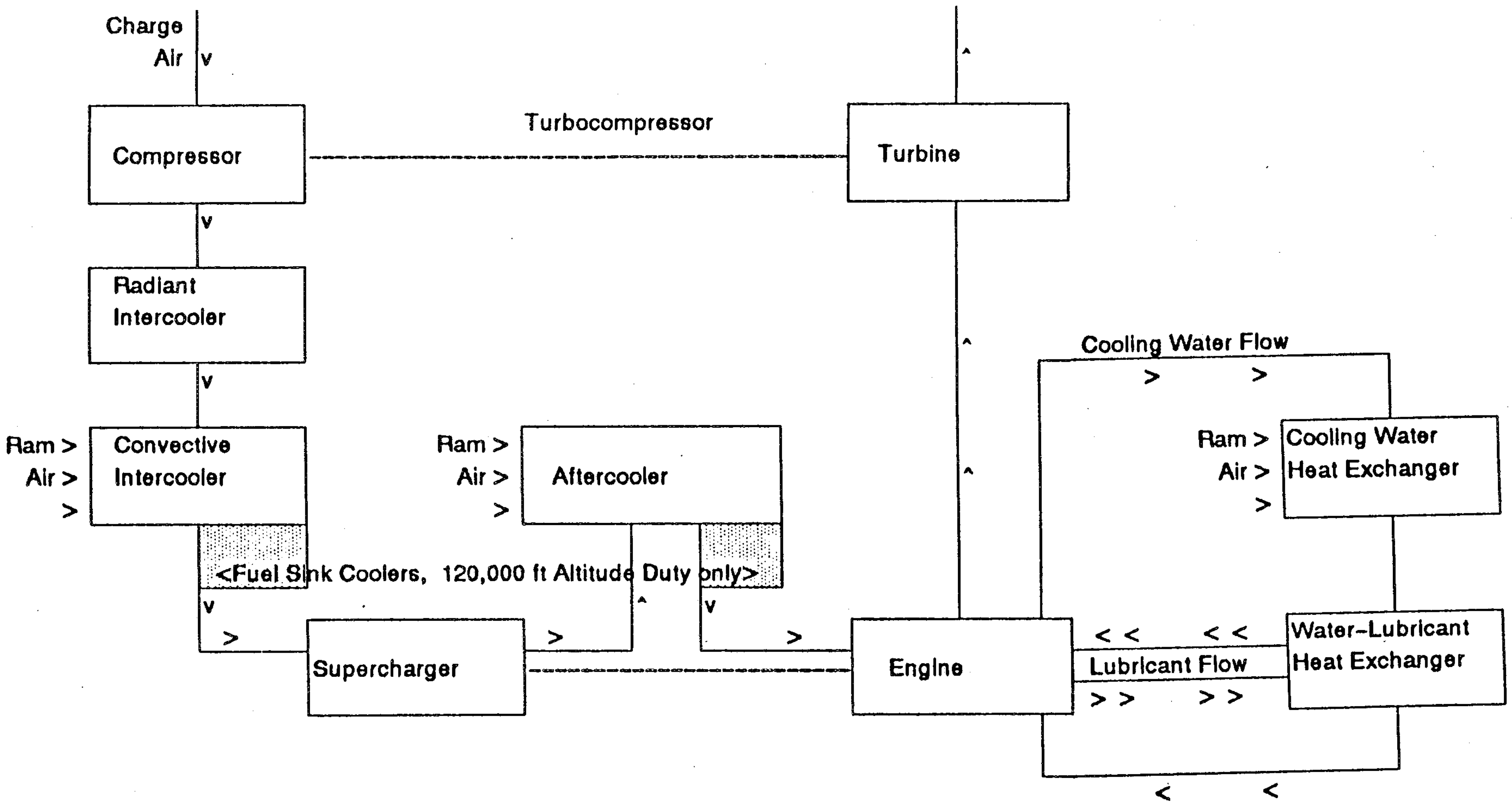

Figure A-1 Cooling system schematic.

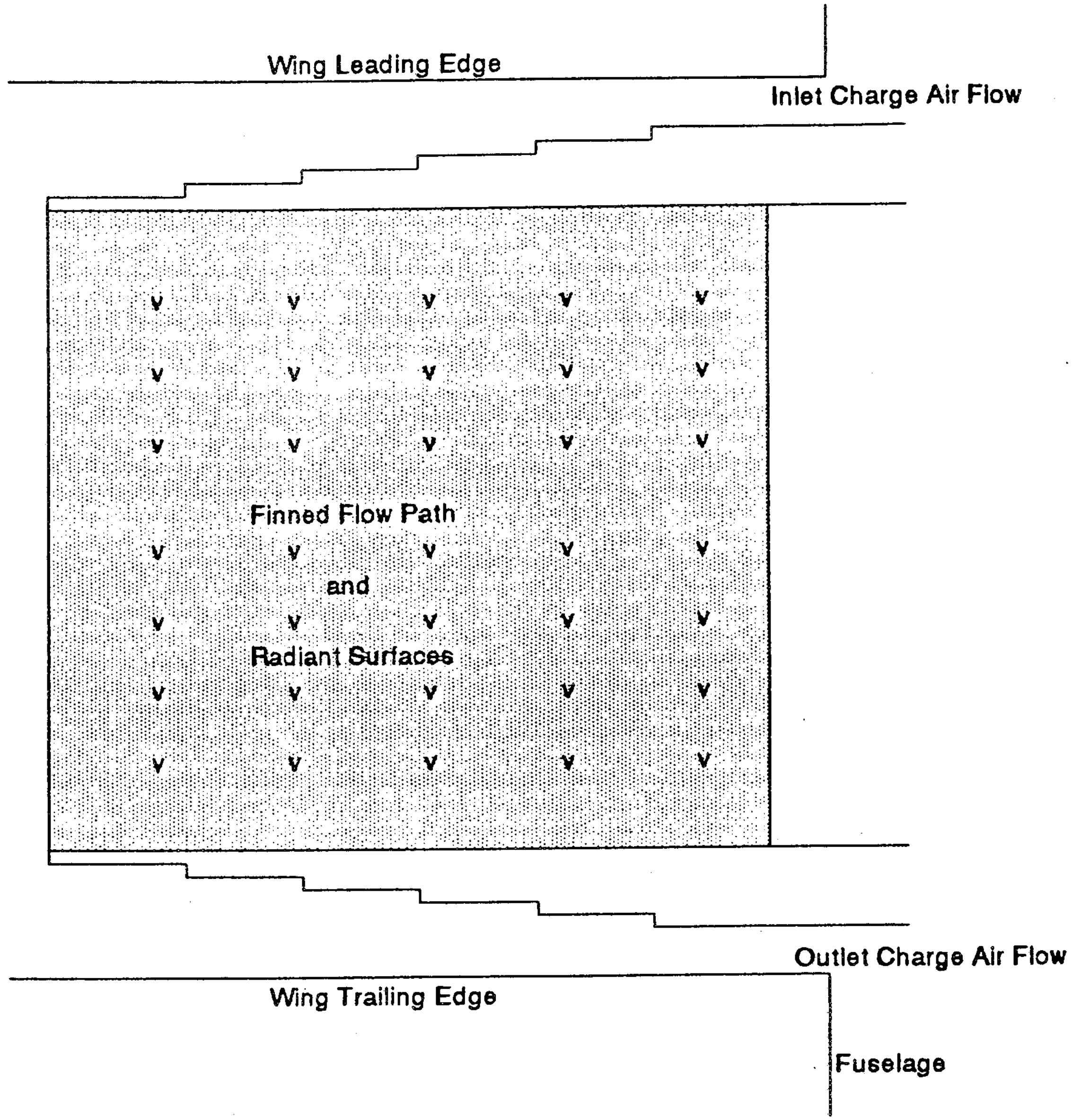

Figure A-2 Plan view of radiant intercooler (one of two).

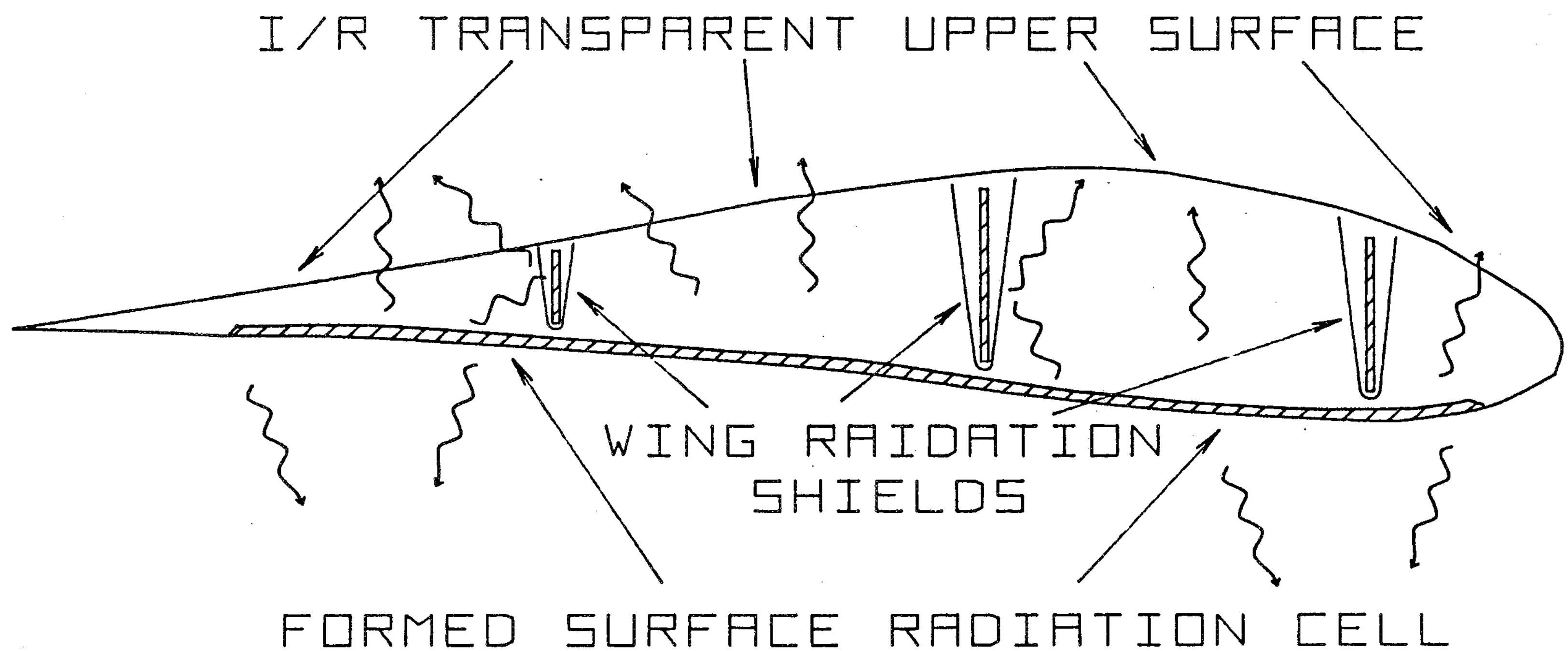

Figure A-3 Wing radiator installation concept.

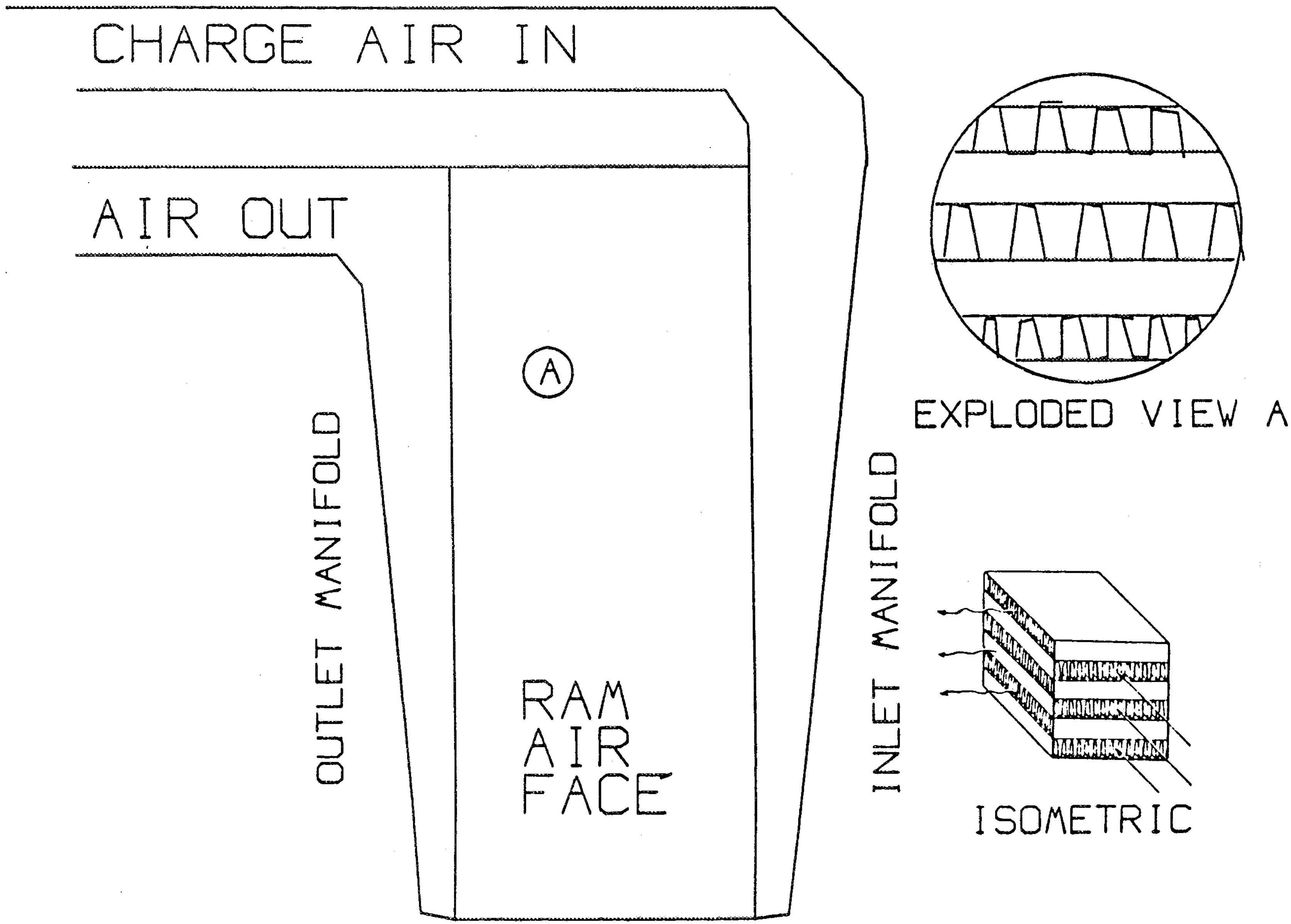

Figure A-4 Heat exchanger frame.

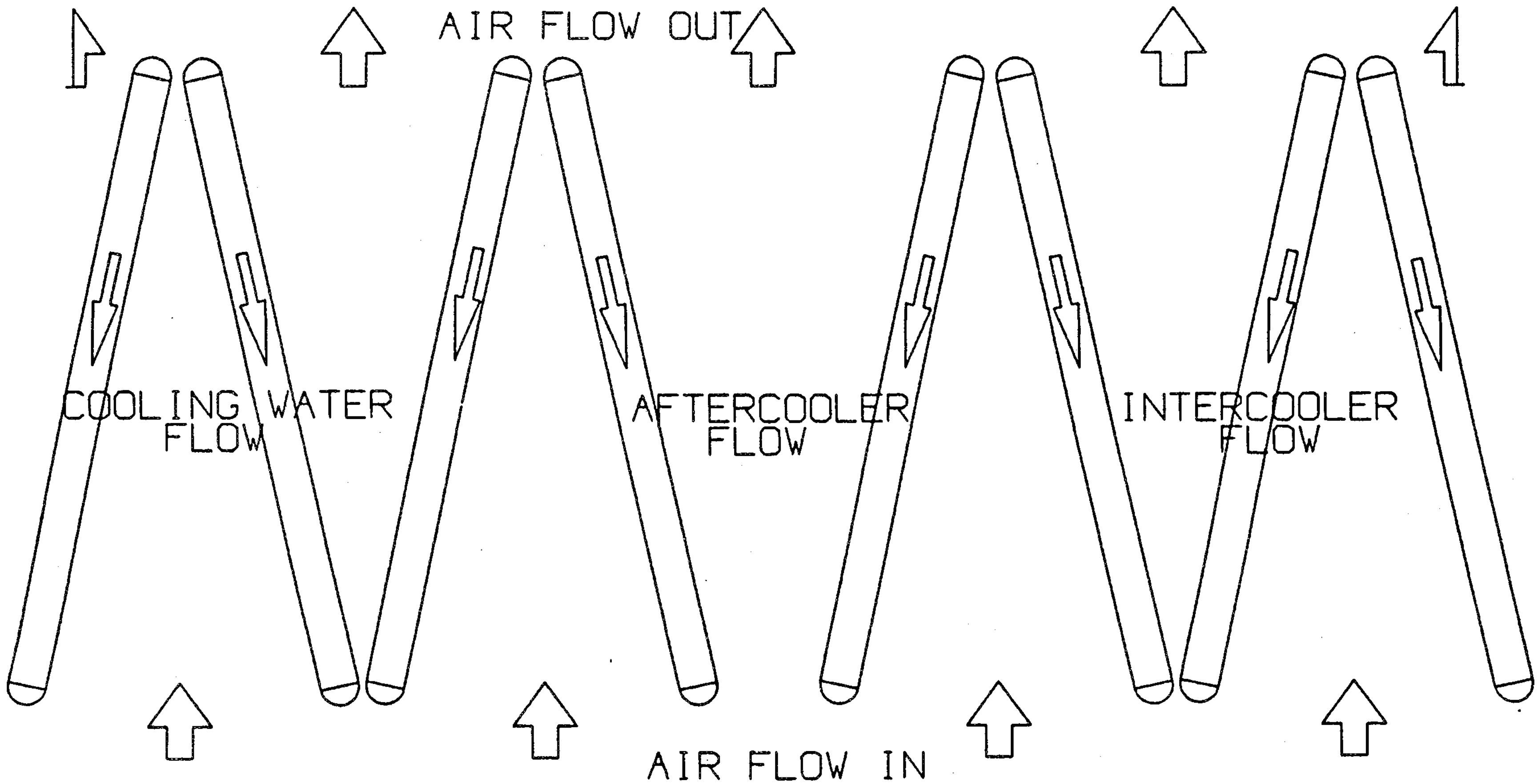

Figure A-5 Bay cooling layout.

The three engine cooling system frames are on the left of the bay. The piping from the engine cooling system feeds the manifolds at the rear of the bay at the top. The cold EGW is taken off at the front on the bottom and passes through the oil cooler. It is then pumped back through the engine.

1.2.3.3 Oil Cooler

The oil cooler is a plate fin, counterflow unit. The fin passages on the water side are higher than those on the oil side to accommodate the larger flow.

1.2.4 Weight Summary

The total weight of the heat exchanger system is 2000 pounds. The largest single component is the pair of radiant intercoolers at 800 pounds. The eight ram air cooling frames as a total are 1000 pounds, with 200 in the manifolding, ducting, and liquid loop components. Heat exchangers configurations are defined in Table A-4.

Table A-4 Heat exchanger component descriptions.

Ram Air Convectors:				
Material Of Construction – Aluminum Core Construction – Brazed Plate Fin Number of Convector Frames – Eight				
Frame Core Dimensions – All units No Flow Length – 10 feet Ram Air Flow Direction – 0.25 feet Cooled Fluid Flow Direction – 4 feet Manifolds – Tubular Sectors, 10 feet long Plate Thickness – 0.010 in				
Fin Dimensions				
Fluid	Height in	Fins 1/in	Offset in	Thickness in
Ram Air	0.5	10	3	0.004
Cooling Water	0.025	38	0.05	0.001
Aftercooler	0.178	12	0.178	0.004
Intercooler	0.5	10	3	0.004

Radiant Intercooler
Materials of Construction Outside Plates – Stainless Steel Fins – Nickel Windows – Zinc Selenide Window Framing – Aluminum Core Construction – Brazed Plate Fin Number of Radiant Cores – Two
Core Dimensions No Flow Length – 10 feet Charge Air Flow Direction – 10 feet Manifolds – Tapered Rectangular Ducts 10 feet long Plate Thickness – 0.010 in
Fin Dimensions Height – 3 inches Fins – 7 per inch Offset – 0.5 in Thickness – 0.004 in

Component Weights			
Item	Number	Unit Wgt. lbm	Total lbm
Radiant Intercooler	2	330	660
Ram Air Intercooler	3	129	387
Aftercooler	2	139	278
Engine Cooling HX	3	135	405
Lubricant Exchanger	1	25	25
Ducting, manifolds	–	–	200
Total			1955

1.3 Heat Exchanger Performance Analysis

1.3.1 Bay Cooling

In each case the ram airflow is calculated using one half the available stagnation pressure head. The frontal area was modified by increasing the no-flow length until the pressure drop criterion was satisfied. If the no-flow length was greater than ten feet, more than one leg of the accordion was used.

1.3.1.1 Engine Cooling

The engine cooling heat exchanger is designed for operation at 120,000 ft with the combined engine and lubricant cooling loads. The limiting condition is the ratio between the heat load and heat capacity of a given volume of air. The design is specified to hold the oil outlet temperature to 250 °F over all operating conditions.

The final configuration required three frames of the accordion. The design operating point is shown on Table A-5. With this configuration, the heat exchanger will satisfy thermal requirements at all other operating points. At intermediate altitudes, the engine cooling fluid will be overcooled, so that a thermostatically controlled bypass will be required to limit excessive cooling.

Table A-5 Engine cooling design operating point.

Design Point Operation Engine Cooling Water		
Hot Fluid	50% EGW	
Cold Fluid	Ram Air	
Altitude	120000 ft	
Mach Number	0.70	
Shaft Horsepower	625	
Fluid	Hot	Cold
Flow Rate, lbm/sec	16.77	4.29
Inlet Pres, psia	100.00	0.09
Inlet Temp, F	256	37
Outlet Temp, F	242	248
Pres Drop, psi	3.955	0.013
Effectiveness	0.067	

Table A-5 Continued.

Operating Performance Map		
Oper. Point	Ram Air Flow lbm/sec	Delivery Temp F
1	276.7	92
2	165.7	17
3	101.2	-33
4	53.6	-17
5	21.7	31
6	13.6	87
7	9.7	162
8	8.2	214
9	5.8	322
10	4.3	242
11	29.3	1
12	14.0	25
13	15.7	22
14	7.7	101
15	6.5	55

1.3.1.2 Intercooler

The low temperature stage of the intercooler is designed to receive air from the radiant high temperature section at temperatures below 600 °F. Pressure level is the factor which makes the design more difficult than that of the aftercooler. The overall pressure ratio is the criterion that determines the impact of the heat exchangers on system performance. Since the pressure level in the intercooler is lower than in the aftercooler, any given pressure drop will have a greater impact on the pressure ratio. In addition, since the fluid is less dense in the intercooler, there will be a greater volume flow for the equal mass flow, requiring more frontal area for a given pressure drop.

The charge air fin passages are taller and have fewer fins than the liquid coolers to more nearly satisfy the pressure drop requirements. The thermal design data are given in Table A-6.

Table A-6 Low-temperature intercooler design operating point.

Design Point Operation Low Temperature Intercooler		
Hot Fluid	Charge Air	
Cold Fluid	Ram Air	
Altitude	120000 ft	
Mach Number	0.70	
Shaft Horsepower	625	
Fluid	Hot	Cold
Flow Rate, lbm/sec	1.60	1.86
Inlet Pres, psia	4.67	0.09
Inlet Temp, F	498	37
Outlet Temp, F	143	344
Pres Drop, psi	0.098	0.013
Effectiveness	0.769	

Operating Performance Map			
Oper. Point	Ram Air Flow lbm/sec	Effectiveness	Charge Air DP psi
1	153.3	0.998	0.052
2	96.3	0.997	0.071
3	59.7	0.994	0.091
4	27.8	0.988	0.093
5	9.8	0.947	0.088
6	5.9	0.865	0.115
7	4.1	0.746	0.152
8	3.4	0.830	0.113
9	2.4	0.799	0.094
10	1.9	0.769	0.098
11	17.0	1.000	0.009
12	7.7	1.000	0.014
13	8.2	1.000	0.016
14	3.4	0.943	0.074
15	2.8	0.910	0.080

Variable Estimation Equations

Ram Air Flow, lbm/sec

Pt2*DPBAY > 0.01 psi^2

Airflow = 229*(Pt2*DPBAY)^0.546

Pt2*DPBAY < 0.01 psi^2

Airflow = 621*(Pt2*DPBAY)^.854/(Chargeflow)^0.14

Effectiveness

Eff = 1.134-0.35*(ChargeFlow/Airflow)
-.0372*ChargeFlow

Maximum Eff = 1.0

Charge Air Presure Drop, psi

PrDrop = .000319*(Charge Flow)^1.32*
(InletTemp/Pres)^.95

Inlet Temperature in Rankine

1.3.1.3 Aftercooler

The aftercooler design approach is the same as the intercooler, only with smaller charge air passages to take advantage of the higher pressure air. Design data for the aftercooler are given in Table A-7.

1.3.2 High Temperature Intercooler (Radiant System)

The high temperatures in the intercooler approached 1300 °F. These temperatures are high enough to provide efficient radiation from surfaces which can view a cold ambient. The radiation exchanger consists of a single plate fin exchanger on the bottom surface of the wing. Thus there is direct heat transfer from this surface to the earth surface ambient.

The top skin of the plate fin sees the outside ambient through transparent windows on the upper surface of the wing. These windows are cooled by the airflow over the wing to remove any transmissivity losses. An overall view factor of 1.7, based on the plan view area, is used to account for blockage of the line-of-sight on the top surface.

An effective sink temperature of 80 °F is used for the calculations. Because of the fourth power radiation parameter, this sink temperature has little effect on the net radiant performance.

Table A-8 gives the design point calculation for the 110,000 ft altitude, which gave the highest outlet temperature. The design outlet temperature is taken as 560 °F.

1.3.3 Oil Cooling

The oil cooler has approximately one tenth the heat load of the engine cooling water, leading to the selection of a liquid-to-liquid exchanger. The control on the cooling water exchanger prevents overcooling at intermediate altitudes. Design data are given in Table A-9.

Table A-7 Aftercooler design operating point.

Design Point Operation Aftercooler		
Hot Fluid	Charge Air	
Cold Fluid	Ram Air	
Altitude	120000 ft	
Mach Number	0.70	
Shaft Horsepower	625	
Fluid	Hot	Cold
Flow Rate, lbm/sec	1.60	1.81
Inlet Pres, psia	19.72	0.09
Inlet Temp, F	494	37
Outlet Temp, F	140	350
Pres Drop, psi	0.288	0.013
Effectiveness	0.773	

Operating Performance Map			
Oper. Point	Ram Air Flow lbm/sec	Effectiveness	Charge Air DP psi
1	127.1	1.000	0.319
2	81.7	0.999	0.328
3	53.9	0.998	0.329
4	28.8	0.995	0.289
5	11.5	0.968	0.264
6	6.4	0.895	0.337
7	4.2	0.822	0.385
8	3.3	0.841	0.338
9	2.3	0.801	0.311
10	1.8	0.773	0.288
11	18.2	1.000	0.050
12	8.5	1.000	0.073
13	9.0	1.000	0.083
14	3.1	0.943	0.276
15	2.5	0.904	0.282

Variable Estimation Equations
Ram Air Flow, lbm/sec Pt2*DPBAY > 0.01 psi^2 Airflow = 153*(Pt2*DPBAY)^0.548 Pt2*DPBAY < 0.01 psi^2 Airflow = 1557*(Pt2*DPBAY)^1.01/(Chargeflow)^0.144
Effectiveness Eff = 1.153-0.358*(ChargeFlow/Airflow)-.037*ChargeFlow Maximum Eff = 1.0
Charge Air Presure Drop, psi PrDrop = .00372*(Charge Flow)^1.39*(InletTemp/Pres)^.95 Inlet Temperature in Rankine

Table A-8 Radiant system high-temperature intercooler design calculation for an altitude of 110,000 ft.

Design Point Operation High Temperature Intercooler		
Hot Fluid	Charge Air	
Radiation Heat Sink, 80F	Line of Sight	
Altitude	110000 ft	
Mach Number	0.60	
Shaft Horsepower	1200	
Fluid	Air	Surface
Flow Rate, lbm/sec	2.17	
Inlet Pres, psia	6.45	
Inlet Temp, F	1255	1138
Outlet Temp, F	582	561
Pres Drop, psi	0.445	

Operating Performance Map		
Oper. Point	Outlet Temp F	Charge Air DP psi
1	88.8	0.103
2	101.8	0.156
3	151.3	0.224
4	325.9	0.267
5	485.5	0.287
6	562.6	0.394
7	595.5	0.444
8	582.3	0.445
9	540.4	0.385
10	497.8	0.414
11	176.2	0.033
12	237.5	0.055
13	260.9	0.069
14	426.5	0.281
15	444.1	0.313

Performance Estimate Equations

Outlet Temperature:

Temp = 337+2758*log(ChargeFlow)*
(0.402 *log(InletTemp) - 1)

Temperatures in Fahrenheit

Charge Air Pressure Drop, psi

PrDrop = .00372*(Charge Flow)^1.39*
(InletTemp/Pres)^.95

Inlet Temperature in Rankine

Table A-9 Lubricating oil cooler design operating point.

Design Point Operation Lubricating Oil Cooler		
Hot Fluid	Engine Oil	
Cold Fluid	50% EGW	
Altitude	120000 ft	
Mach Number	0.70	
Shaft Horsepower	625	
Fluid	Hot	Cold
Flow Rate, lbm/sec	2.48	16.77
Inlet Pres, psia	100.00	100.00
Inlet Temp, F	259	242
Outlet Temp, F	244	243
Pres Drop, psi	1.618	0.273
Effectiveness	0.857	

2.0 Analysis

The following sections give the system analysis approach which led to the design of the heat exchanger complex for the Ames HAARP aircraft.

2.1 Heat Sinks

2.1.1 Ambient Air

The traditional ultimate heat sink for any cooling system in flight is atmospheric air. The difficulty with this cooling medium at high altitudes is the extreme rarefication. Not only does a unit volume have a low heat capacity so that large volumes are required to remove a certain amount of heat, but also the low density leads to high pressure drops driving designs into the less efficient laminar flow regime. Fortunately, conditions do not lead to operation in the slip regime where air behaves like a group of individual molecules rather than a continuous medium.

The key to designing a heat exchanger in this region is to supply adequate airflow to remove the heat. This requires a large frontal area to permit the passage of air. If the cooling airflow is less than charge airflow, the maximum effectiveness of the heat exchanger is the ratio of the two flow heat capacities.

2.1.2 Radiation

Radiation becomes more effective at very high temperatures, since the heat transfer rate is proportional to the fourth power of the temperature. Figure A-6 gives the radiant heat flux from one square foot of a black body surface as a function of temperature. Thus high heat fluxes can be obtained from any surface that has line-of-sight to a reasonable heat sink. The temperature of this heat sink has very little effect on the radiant effectiveness of the system. Thus it is more effective to radiate from a 500 °F source (heat flux, 1500 Btu/hr) on the lower surface of a wing to an 80 °F heat sink (returning heat flux, 150 Btu/hr) than from the upper surface to outer space (potential solar heat flux, 250 Btu/hr).

Thus radiation is likely to be a very effective cooling system for part of the intercooler heat load, in the regime where temperatures are extremely high. Aftercooler temperatures do not get high enough to justify a radiator.

2.1.3 Convective Surfaces

The external convective tail surfaces of the aircraft were evaluated as potential heat sinks for engine liquids. It would be easy to pump the liquids back to the tail. Analysis showed that even at the 0.7 cruise Mach number, the heat transfer coefficients were about one Btu/hr-ft^2-F. This would require a thousand degree temperature difference to remove the engine cooling load from the 160 square feet available on the tail. The boundary layer thickness is large enough to effectively blanket any fins that could be put on the exposed surface.

2.1.4 Fuel

Fuel has been used as a heat sink in high altitude aircraft, particularly for electronic cooling. In this case the sink capacity is approximately 25,000 Btu/hr, which is one-fourth the minimum heat load (engine lubricating oil).

2.1.5 Latent Heat

In some cases, when there is a high transient heat load, a phase change heat sink is used to store the excess. The most effective of these for a single use is a water boil-off system. This could be used, for example, to make up the differential between the capacity of a system designed for 110,000 ft altitude and one designed for 120,000 ft.

The maximum heat capacity of such a system is 1000 Btu/lbm. Thus, each installed pound in a boil-off system could reduce the required heat load by 500 Btu/hr for the two hour mission. This weight penalty would act like a fuel penalty, in that it would only be charged against the first half of the mission.

2.1.6 Summary

Fuel, convective surfaces, and latent heat are not feasible heat sinks for the system heat loads. Radiation is only applicable to heat sources which are greater than 500 °F; at lower temperature levels the effective heat transfer rate is too low. Thus it is useful for the high temperature first stage of intercooling at altitude, and perhaps, for the high temperature trim of the aftercooler.

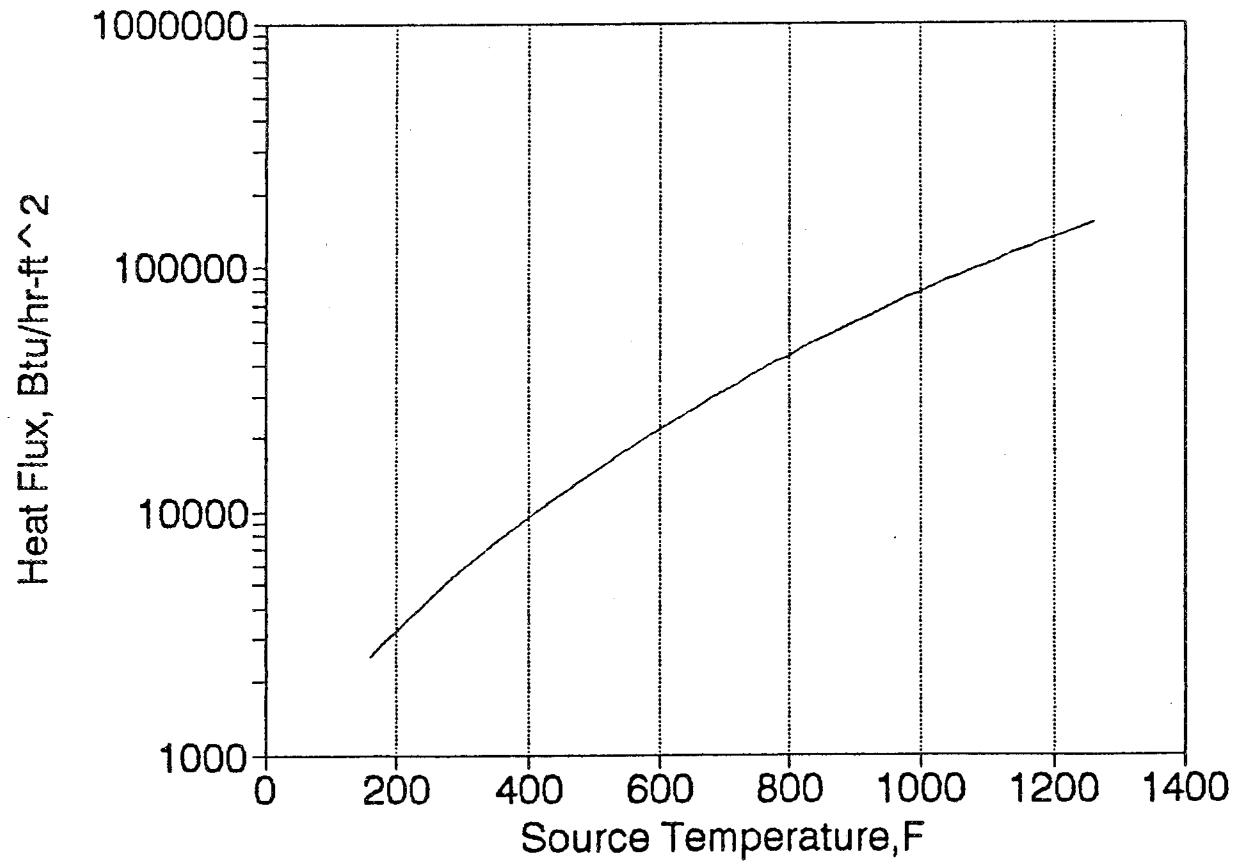

Figure A-6 Black body radiation.

2.2 Overall Approach

2.2.1 Ambient Air Cooling

The design of a heat exchanger is a trade-off between required performance and the penalties of surface area and pressure drop. In this high altitude operation, ram air is used for cooling and the pressure drop available is limited to approximately half the ram velocity head. Table A-10 gives design ram air conditions for a NACA standard atmosphere + 18 °F.

Table A-10 Design ram air conditions for a NACA standard atmosphere plus 18 °F.

Hot Day Altitude Operating Conditions						
Point	Altitude ft	Pres psi	Temp F	Mach No	PTi psia	TTi F
1	0	14.700	86.0	0.09	14.784	86.9
2	20000	6.753	5.7	0.12	6.821	7.0
3	40000	2.720	-51.7	0.20	2.797	-48.4
4	60000	1.040	-51.7	0.30	1.107	-44.4
5	80000	0.400	-43.9	0.40	0.447	-30.6
6	90000	0.251	-38.4	0.44	0.287	-22.0
7	100000	0.158	-32.8	0.50	0.188	-11.5
8	110000	0.103	-24.1	0.60	0.131	7.3
9	115000	0.083	-15.8	0.70	0.115	27.7
10	120000	0.067	-7.6	0.70	0.092	36.7
11	80000	0.400	-43.9	0.46	0.463	-26.3
12	90000	0.251	-38.4	0.50	0.298	-17.3
13	100000	0.158	-32.8	0.60	0.202	-2.1
14	110000	0.103	-24.1	0.70	0.142	18.7
15	115000	0.083	-15.8	0.70	0.115	27.7

As an initial cut at the design of the heat exchangers, a criterion is set that the mass flow of cooling air not be less than the mass flow of charge air at any operating condition. This set 120,000 ft as the design point.

2.2.1.1 Heat Exchanger Selection Process

The type of heat exchanger used in a specific application (shell-and-tube, finned tube, plate, regenerative, or plate fin) depends primarily on the fluid properties. Thus shell-and-tube and plate exchangers are aimed primarily at liquid to liquid applications, finned tube to liquid-to-gas, and regenerative to gas to gas cases where leakage between the two fluids is not critical. Plate fin covers the entire gamut of situation.

Both plate fin and finned tube exchangers were evaluated for the coolant and lubricant exchangers. The plate fin could pass more air for a given face area (due to the bluff body pressure drop across the tubes) and was selected. Only plate fin units were considered for the air-to-air exchangers.

2.2.1.2 Materials

The basic material of construction for flight heat exchangers is aluminum. It combines light weight, ease of fabrication, and high thermal conductivity to make an ideal material of construction. The primary drawback is the operating temperature limitation, which conventional rule of thumb sets at 500 °F. In this application we have used a slightly higher limit on inlet airflow (565 °F). In order to do so we have to keep stresses below the 5000 psi range and accept a lower total operating life (1000 hours) due to creep. The high temperature operating points are at low pressure, and thus low stress.

At elevated temperatures stainless steel is the material of choice. However, it has the combined drawbacks of low thermal conductivity and high density. In those parts where thermal conductivity is critical, nickel is substituted. It has favorable operating characteristics and a thermal conductivity 40% of that of aluminum.

2.2.2 Radiation Cooling

For radiation to be effective the exposed surface must be hot. In an aircraft hot surfaces with an adverse pressure gradient will lead to separation of the airflow, with increased drag and perhaps even stall. Thus only the lower surface of the wing, where there is a positive gradient, can be considered as a direct heat transfer surface. However, there are materials which will transmit radiation even into the low infrared region where these systems are operating. Such a material is zirconium oxide. This can potentially be used as a window on the upper wing surface to make maximum use of the available exposed surface.

2.2.3 Intermediate Loop

To take advantage of radiation, the hot fluid must be ducted to the wing where an area is available for heat transfer. This can be done either by ducting the low pressure air to the wings or by an intermediate liquid loop. The liquid loop has the advantage of reducing potential pressure losses causes by the flow of low pressure air through long ducts and a large heat exchanger. On the other hand, it imposes the complication of an extra fluid loop, with accompanying pump, controls and additional heat exchanger.

The fluid used is the most important single element in an intermediate loop. Preferred fluids from a heat transfer standpoint are liquid metals, such as NaK which was considered for the nuclear reactors. However, liquid metals are excessively corrosive and require special handling. Commercial heat transfer fluids are limited to approximately 800 °F.

2.2.4 Conclusion

Based on the evaluation of alternates considered above, the final system configuration is composed of:

Aluminum plate fin oil to EGW exchanger

Aluminum plate fin EGW to ram air exchanger

Aluminum plate fin charge air to ram air aftercooler

Aluminum plate fin charge air to ram air low temperature intercooler

Nickel and stainless steel finned duct charge to ambient radiant high temperature intercooler

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13. ABSTRACT *(Maximum 200 words)*

Satellite surveillance in such areas as the Antarctic indicates that from time to time concentration of ozone grows and shrinks. An effort to obtain useful atmospheric data for determining the causes of ozone depletion would require a flight vehicle capable of reaching altitudes of at least 100,000 ft and flying subsonically during the sampling portion of the mission. A study of a heat rejection system for an advanced variable cycle diesel (AVCD) engine was conducted. The engine was installed in an extreme altitude, high altitude advanced research platform. Results indicate that the waste heat from an AVCD engine propulsion system can be rejected at the maximum cruise altitude of 120,000 ft. Fifteen performance points, reflecting the behavior of the engine as the vehicle proceeded through the mission, were used to characterize the heat exchanger operation. That portion of the study is described in an appendix titled, "A Detailed Study of the Heat Rejection System for an Extreme Altitude Atmospheric Sampling Aircraft," by a consultant, Mr. James Bourne, Lytron, Incorporated.

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