Virginia Tech
&
Loughborough University

VIRGINIA POLYTECHNIC INSTITUTE AND STATE UNIVERSITY

VULTURE

1999 / 2000 AIAA Undergraduate Team Aircraft Design

Cruise Missile Carrier

VULTURE

1999/2000 AIAA Undergraduate Aircraft Design Competition

Virginia Polytechnic and State University and Loughborough University

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Executive Summary

An international team comprised of students from Virginia Tech and Loughborough University proudly presents the Vulture for the 1999/2000 AIAA Foundation Undergraduate Team Aircraft Design Competition. This proposal fulfills the request for a Cruise Missile Carrier (CMC) Aircraft with multirole capabilities. The Vulture was designed not only as a cost-effective solution that can deliver 10 AGM-86Cs Conventional Air-Launched Cruise Missiles (CALCM), but also to provide a flexible and adaptable transport aircraft to the military and other potential buyers.

In its design mission as a CMC aircraft, the Vulture is able to carry 10 CALCMs a range of 4000 NM, launch the missiles, and then return 1000 NM. As a multirole aircraft, the Vulture can fly up to 40000 lbs of a wide variety of payloads a total range of 5000 NM. These payloads include, but are not limited to, cargo in standard military and commercial palettes, HUMVEES, troops, and fuel. The Vulture is also capable of meeting tanker, electronic warfare (EW), airborne command center, and medical evacuation needs as required by the military. The Vulture owes its adaptability and versatility to its two-cell design: a lower unpressurized pannier or sponson carries CALCMs or it can be fitted to house other payloads such as fuel tanks for tanker missions. The larger, pressurized upper section houses the Vulture's large cargo bay. The CMC version of the Vulture will be made with provisions to perform the design mission, but since the upper cargo bay remains inviolate, the CMC version is even capable of carrying various cargo loads and performing multiple missions with situations dictate.

In designing the Vulture, the two major challenges that had to be met were cost effectiveness and multirole considerations. After examining how to make the Vulture a cost effective solution, it was determined that the multirole capability of the design was strongly linked to the cost effectiveness. After looking at current militaries, the team concluded that for a production run of 600 aircraft, as stated by the RFP, only about 20% would be purchased by the United States military to fulfill CMC mission needs. Therefore, the ability to sell the Vulture around its multi-mission adaptability and flexibility would be vital in giving it a competitive advantage. For this reason, the multirole capabilities of the Vulture helped drive the design to make it cost effective. Meeting this mutli-mission design driver proved to be challenging and required that many decisions be made.

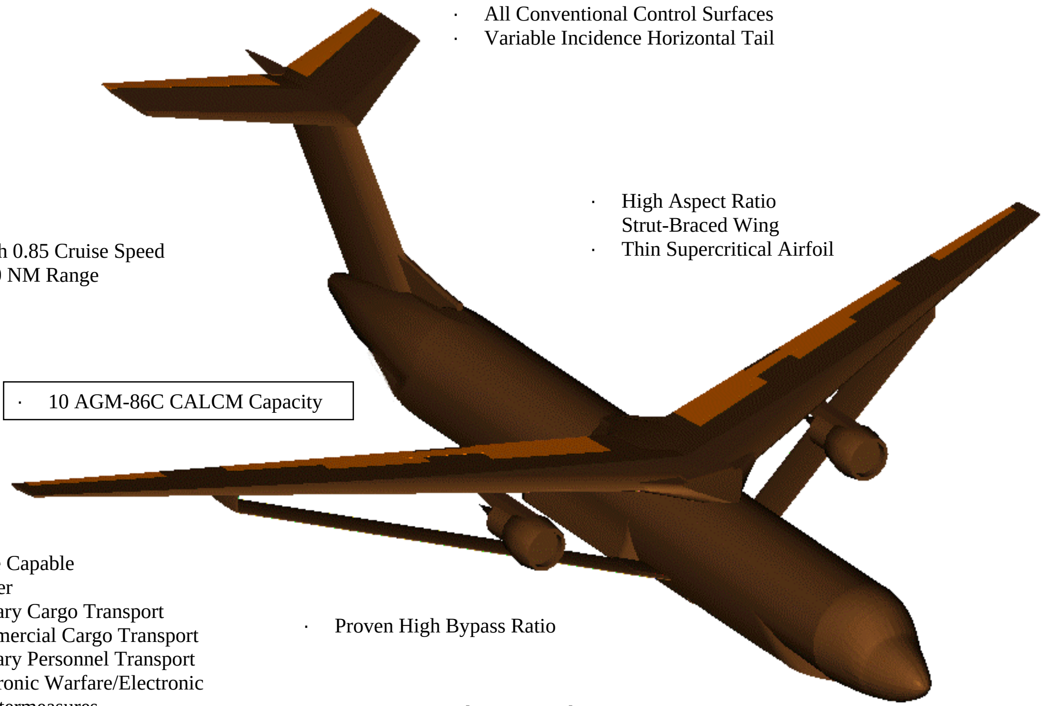
In addition to its multirole capabilities, the Vulture is set apart from other aircraft by its strut-braced wing (SBW) design. The aircraft employs a large strut spanning from the bottom of the fuselage to about 70 percent of the

half-span on the high-mounted wing. The basic concept behind the SBW design is that the strut reduces the bending moment in the wing during flight. When compared to a typical cantilever wing, the reduction in bending moment allows for a reduction in the wing thickness, which increases the aerodynamic efficiency of the wing. The SBW concept is a multi disciplinary design because aerodynamics and structures are so closely interrelated and must be optimized simultaneously. The promise of the SBW design and the adaptable use of the two-section cylindrical fuselage make the Vulture a cost-effective multirole aircraft. The Vulture is then the best solution for future military needs, providing a cost-effective multimission platform.

Acknowledgements

The members of team Vulture would like to thank our friends, families, and roommates for their emotional support and patience as we prepared this proposal. We also owe a debt of gratitude to Dr. William Mason and Mr. Nathan Kirschbaum of Virginia Tech and Dr. Gary Page, Dr. Lloyd Jenkinson, and Mr. Paul Simpkin of Loughborough University for their wealth of knowledge and expertise in the area of aircraft design. We would also like to extend our deepest gratitude to Andy Ko, who spent countless hours aiding us in the use and modification of the MDO code which helped make the Vulture, which started as a dream, a reality. Andy never once turned us away when we approached him or greeted us with a negative comment, and for that we owe him deeply.

The Vulture



Type: Long Range, Cruise Missile Carrier, Tanker, Medium Size Transport

Wings: High, Swept back, Strut-Braced, Supercritical airfoil

Fuselage: Conventional semi-monocoque construction of primarily aluminum

Tail Unit: T-tail, refueling boom installed on tanker variants

Landing Gear: Retractable tricycle type, Double-bogie main gear with eight 3.33ft dia wheels, Two steerable nose wheels of 3.33 ft dia

Power Plant: Two CFM56-5A3 high bypass turbofans producing 27,000 lbs of thrust each

Accommodations: Two crew member flight deck. Military – provisions for a weapons and electronic counter measures officer; loadmaster. Commercial – provisions for two loadmasters

Dimensions, External:

Wing Span:	149.8	ft
Length overall:	149.6	ft
Height overall:	36.21	ft
Wheelbase:	60.82	ft

Areas:

Wing:	1792.4	ft^2
Horizontal Tail:	425.0	ft^2
Vertical Tail:	323.9	ft^2
Spoilers:	109.5	ft^2
Elevators:	120.0	ft^2
Ailerons:	114.8	ft^2
Rudder:	92.6	ft^2

Weights:

Empty:	85,198	lb
TOGW:	181,200	lb

Flyaway Cost:

Tanker/Cargo:	$64.21 Million
Cruise Missile Carrier:	$67.50Million

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Nomenclature

Symbol	Description	Symbol	Description
CALCM	Conventional Air Launched Cruise Missile	X_i	X Location of Component
CMC	Cruise Missile Carrier	Y_i	Y Location of Component
EW	Electronic Warfare	Z_i	Z Location of Component
RFP	Request for Proposal	CG	Center of Gravity
SBW	Strut Braced Wing	I_{xx}	Second Moment of Inertia About the x-axis
ECM	Electronic Countermeasures	I_{yy}	Second Moment of Inertia About the y-axis
TOGW	Take-off Gross Weight	I_{zz}	Second Moment of Inertia About the x-axis
RDT&E	Research Development Testing and Evaluation	RWR	
BWB	Blended Wing Body	ELINT	
MEA	Modified Existing Aircraft	DOC	Direct Operating Cost
OEC	Overall Evaluation Criteria	IOC	Indirect Operating Cost
L/D	Lift to Drag ratio	GAO	General Accounting Office
P_{surv}	Probability of Surviving	C_{di}	Induced drag coefficient
P_D	Probability of Detection	C_{L1}	Lift coefficient at a given flight condition
$P_{H/D}$	Probability of Being Hit if Detected	C_{D1}	Drag coefficient at a given flight condition
$P_{K/H}$	Probability of Being Killed if Hit	C_{Tx1}	Thrust coefficient at a given flight condition
MCI	Mission Capability Index	$C_{L\alpha}$	Aircraft lift curve slope (1/rad)
$W_{payload}$	Payload Weight	C_{Lq}	Lift due to pitchrate deriv. (1/rad)
W_{fuel}	Fuel Weight	$C_{m\alpha}$	Pitching moment due to alpha deriv.(1/rad)
W_{empty}	Empty Weight	C_{mq}	Pitching moment due to pitch rate deriv. (1/rad)
EAI	Engine Caused Attrition Index	$C_{m\alpha dot}$	Pitching moment due to alpha rate deriv. (1/rad)
A_i	Inherent Availability	$C_{L\delta e}$	Lift due to elevator deflection deriv. (1/rad)
MTBF	Mean Time Between Failure	$C_{m\delta e}$	Pitching moment due to elevator deflection deriv. (1/rad)
MTTR	Mean Time To Repair	C_{lp}	Rolling moment due to roll rate deriv. (1/rad)
LCC	Life Cycle Cost	C_{np}	Yawing moment due to roll rate deriv (1/rad)
MDO	Multi-Disciplinary Optimization	C_{yr}	Side force due to yaw rate deriv. (1/rad)
t/c	Thickness to chord ratio	C_{lr}	Rolling moment due to yaw rate deriv. (1/rad)
AWACS	Airborne Warning and Control Systems	C_{nr}	Yawing moment due to yaw rate deriv. (1/rad)
JSTARS	Joint Surveillance Target Attack Radar System	$C_{y\beta}$	Side force due to sideslip deriv. (1/rad)
APU	Auxiliary Power Unit	$C_{l\beta}$	Rolling moment due to sideslip deriv. (1/rad)
C_{Lmax}	Maximum Lift Coefficient	$C_{n\beta}$	Yawing moment due to sideslip deriv. (1/rad)

Q	Dynamic Pressure	$C_{y\delta a}$	Side force due to aileron deflection deriv. (1/rad)
S	Surface Area of the Wing	$C_{l\delta a}$	Rolling moment due to aileron deflection deriv. (1/rad)
N	Load Factor	$C_{n\delta a}$	Yawing moment due to aileron deflection deriv. (1/rad)
K_g	Gust Alleviation Factor	$C_{y\delta r}$	Side force due to rudder deflection deriv. (1/rad)
U_{de}	Equivalent Gust Velocity	$C_{l\delta r}$	Rolling moment due to rudder deflection deriv. (1/rad)
V_e	Equivalent Velocity	$C_{n\delta r}$	Yawing moment due to rudder deflection deriv. (1/rad)
$C_{L\alpha}$	Lift Curve Slope of Entire Aircraft	N_α	Load factor derivative with respect to AOA (1/rad)
μ	Mass Factor	ξ_p	Phugoid damping ratio
C	Chord	ξ_{sp}	Short Period damping ratio
ρ	Density	ω_{sp}	Short Period frequency (rad/sec)
X_{CG}	X position of the CG	ξ_{dr}	Dutch roll damping ratio
Y_{CG}	Y position of the CG	ω_{dr}	Dutch Role frequency (rad/sec)
Z_{CG}	Z position of the CG	t_{30deg}	Time to roll to 30 deg (sec)
W_i	Weight of Component	τ_r	Roll mode time constant (1/sec)

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1. Overview

1.1. Introduction

Since World War I, air power has been perceived as paramount to protecting a nation and its interests. Over the past century, weapons have evolved from simple guns attached to planes and free-fall bombs, to sophisticated guided missiles systems and laser-guided “smart” bombs. In fact, with the existence of intercontinental ballistic weapons, some may wonder why the need exists for aircraft to deliver the weapon. As seen in recent events, such as NATO’s involvement in Kosovo, the military is fighting wars with cruise missiles, particularly air-launched cruise missiles.

Unlike conventional bombs and even “smart” bombs, a cruise missile can be launched hundreds, or even over a thousand miles from the target. Therefore, a mission’s success does not depend on whether the launching vehicle is directly over the target. Instead, the launching vehicle, whether an airplane or a naval vessel, can be kept safely outside of hostile enemy airspace or territories in most situations. So then why not use intercontinental ballistic missiles (ICBMs), armed with conventional warheads? In today’s dynamic political situations, agreements can be made late after a mission takes off that can cancel the missile launch. This is not possible with ICBMs: once they are launched, they cannot be recalled. An aircraft can be deployed to a target with a number of cruise missiles ready for launch. Yet, if deemed necessary, the missile launch can be canceled much later into the mission. The two primary methods of launching cruise missiles are from naval ships and from aircraft. A single naval ship can carry and launch many cruise missiles, where as an aircraft is limited in the number of missiles it can carry. Yet, naval ships suffer from slow deployment time: in the days or weeks it takes to deploy the vessel, the political situation at hand can change dramatically and by the time the ship is in range, it may be too late. Modern aircraft can be deployed around the world in a matter of hours. Furthermore, several aircraft can be flown into position to increase the firepower. Therefore, it is evident that air-launched cruise missile systems are a vital component of today’s military.

The United States’ military currently has three aircraft to fulfill the cruise missile carrier role: the B-52, the B-1, and the B-2. For a stand-off mission, none of these aircraft provides a cost effective solution. The B-52 has been in use since the 1960s. While it has proven venerable over the years, the B-52 is aging and becoming outdated. Boeing has delivered a proposal to the Air Force to re-engine the remaining fleet of B-52s (Reference 1.1). While the proposal will cost \$1.3 billion, it is estimated to save taxpayers \$6 billion over the remaining life of the B-52s. Yet, it will still be necessary to replace the B-52 over the next 10 to 20 years. The B-1, while designed to carry and launch nuclear-armed weapons, has been adapted over the years to also carry a variety of conventional weapons. However, the B-1 was designed for low-level penetration, and as a stand-off CMC aircraft. So, while it can fulfill the need for a CMC aircraft, it would do so inefficiently. Introduced in 1996, the B-2 is the latest in the USAF's arsenal of CMC-capable aircraft. It is the paradigm of stealth technology, and also designed as a penetrator aircraft. Yet, the costs associated with the B-2 do not justify its use for mission not requiring penetration: for every hour of flight time, the B-2 requires 119 man-hours of work to repair it (Reference 1.2). Therefore, to meet tighter future defense budgets, it is necessary to design an aircraft to meet stand-off CMC needs affordably.

This proposal will describe the Vulture, an aircraft that will affordably meet stand-off CMC mission needs. But the true advantage of the Vulture lies in its flexibility and adaptability as a multirole aircraft.

1.2. AIAA RFP

The AIAA Request for Proposal (RFP) describes the need for a new Cruise Missile Carrier aircraft capable of launching 10 Conventional Air-Launched Cruise Missiles (CALCMs). In addition to this design mission, the RFP cites that this aircraft design needs to consider multirole capabilities and must be cost-effective. The specific requirements of the RFP will be discussed in the following sections.

1.2.1. Overall Mission Requirements

The overall objective, as stated by the RFP, is "to design a new military aircraft that is capable of transporting and launching 10 AGM-86C missiles economically." The RFP places the constraint of affordability and cost-effectiveness on the design. This is in response to "increasingly tight Defense budgets" and to keep the designs feasible for real life. Furthermore, the a means of cost-effectiveness must be established. The RFP states that the design must be suitable for a 600-aircraft production run. Since it is unlikely that all 600 aircraft will be used for the design CMC mission, alternative missions and payloads must be explored. This multirole capability requirement includes considering roles such as cargo and passenger roles. The proposal should also describe what technologies would be incorporated into the design. Table 1-1 lists additional requirements of the aircraft.

Table 1-1 - General Aircraft Requirements

· Aircraft must meet and be compatible with current military and FAA airport operation requirements. · Military versions must be capable of in-flight refueling. · Aircraft should be designed for a structural load limit factor of 2.5 g's. · Maximum landing weight is equal to 60% of maximum take-off gross weight.

1.2.2. Design Mission Payload

In its design CMC mission, the proposed aircraft design must be capable of transporting and launching 10 AGM-86C cruise missiles. This missile is shown in Figure 1-1. The dimensions and performance of the AGM-86C are given in Table 1-2. Since the design mission is to carry 10 cruise missiles, the resulting payload weight is 32,500 lbs. Since the range of the AGM-86C is 1500 NM (Reference 1.3), the design aircraft can truly be stand-off, meaning that for most targets, it will not be required to fly deep into enemy airspace.

Table 1-2 - AGM-86C CALCM Characteristics

Length	20 ft 9 in
Diameter	24.5 in
Wingspan	12 ft
Weight	3,250 lbs
Range	1,500+ NM

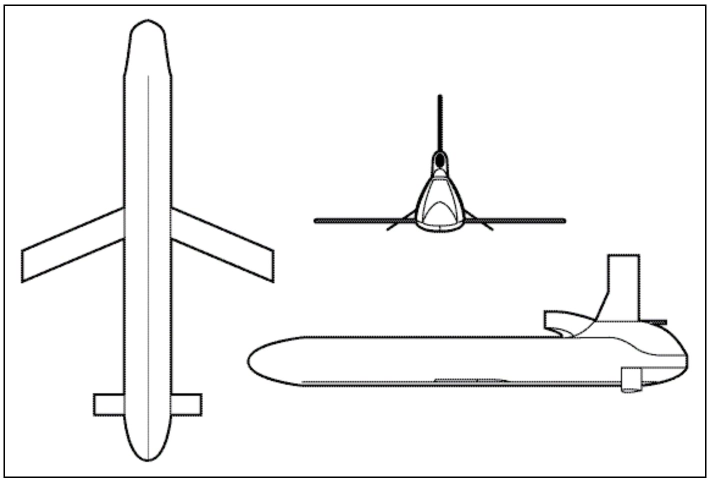
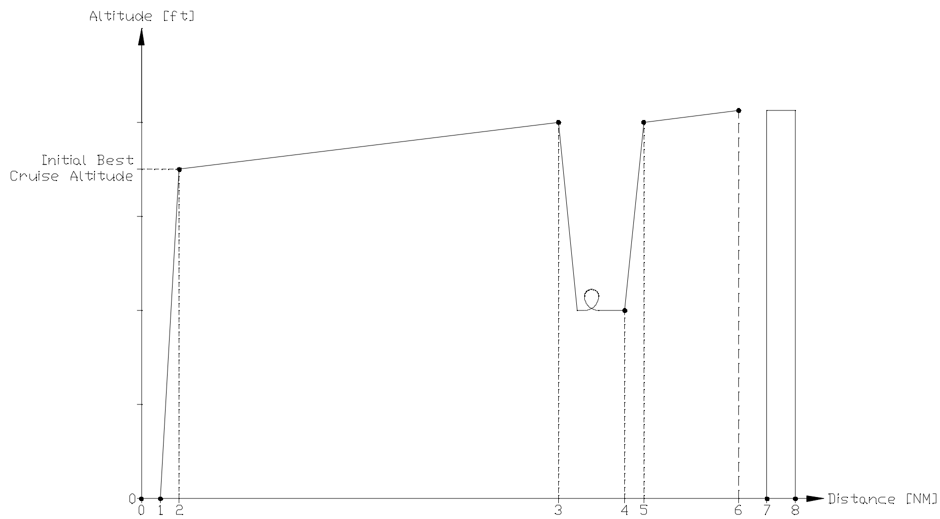


Figure 1-1 - AGM-86C CALCM 3-View

1.2.3. Design Mission Profile

The RFP requires that the design aircraft fly a total unrefueled range of 5000 NM. The aircraft must cruise 4000 NM to the cruise missile launch point, loiter for 30 minutes and launch the 10 cruise missiles, and return 1000

NM. The design aircraft must be able to cruise at no less than 0.85 Mach. The complete Mission Profile is illustrated in Figure 1-2.



0-1 Warm-up for 20 minutes and takeoff.
1-2 Climb at maximum power and best climb speed from sea level to best long-range cruise altitude.
2-3 Cruise at best long-range cruise Mach number to a launch point 4,000 NM from takeoff.
3-4 Descend to best loiter altitude and loiter for 30 minutes at best loiter Mach number.
4 Launch 10 AGM-86C cruise missiles.
4-5 Climb at maximum power and best climb speed to best cruise altitude.
5-6 Cruise at best cruise altitude and cruise Mach number to a field 1,000 NM from launch point.
6-7 Descend to landing field and land with no fuel, time, or distance credit.
7-8 Reserve fuel equal to 5% of total mission fuel.

Figure 1-2 - Design Mission Profile

1.2.4. Point Performance Requirements

The RFP also details several other performance requirements. These are listed in Table 1-3.

Table 1-3 - Point Performance Requirements

- Minimum cruise altitude is to be no less than 30,000 ft. Maximum cruise altitude is to be no greater than 45,000 ft.
- Takeoff distance at maximum TOGW less than 8,000 ft at sea level on an ICAO standard day.
- Landing distance over a 50-ft obstacle at maximum landing weight must be less than or equal to 6,000 ft on an ICAO standard day. The design must also be capable of landing within 6,800 ft at sea level on a USAF tropical day (89° F and 14.7 psi).
- Landing distance over a 50-ft obstacle at maximum landing weight with one engine inoperative must be less than or equal to 8,000 ft on an ICAO standard day.

These requirements will ultimately be used as constraints when sizing the aircraft.

2. Design Drivers

The major design drivers were cost effectiveness and multi-role capability. In the following sections, these design drivers will be discussed.

2.1. Multi-Role Capability

To design an aircraft around the long-range cruise missile carrier (CMC) mission, as outlined in the RFP, one could consider a wide variety of conceptual aircraft and missile carriage configurations -- conventional to exotic -- that could efficiently and effectively meet the design criteria. However, the RFP also requires an aircraft that is capable of performing multiple roles, increasing the complexity of the problem. Before a concept can even be conceived, it is necessary to determine what other roles the aircraft could perform. Is there a need for a new aircraft to fulfill this role? Or, what markets exist for a new type of aircraft? Once these questions have been considered, one must investigate whether or not the new aircraft can fulfill each role effectively and efficiently.

The RFP requires the investigation of aircraft variants to perform civil, cargo and passenger missions. But, the multi-role investigation cannot end there. The RFP bases the need for the primary CMC role on recent events. Yet, these recent events, specifically NATO involvement in Kosovo, point to the need for aircraft to also fulfill tanker, electronic warfare (EW), and electronic counter-measures (ECM) roles (Reference 2.7). The military cargo role should also be considered. One final aspect of multi-role capability is to investigate a variety of CMC ranges and payloads to understand how deviating from the design mission would affect costs. Generally, the range will decrease as the payload weight increases, and increase as the payload weight decreases.

One role that the design aircraft could perform is cargo transport with both commercial and military capacities. If the aircraft serves as a military cargo transport, it must be able to compete with the C-130J, the C-141, and the C-17. The payload capacities of these three aircraft are 45,000, 69,000, and 171,000 lbs, respectively (References 2.2, 2.3, 2.4). But, for military cargo transports, the payload weight class is only one consideration: much of the payload carried by transports is bulky, so payload size and not weight may be the limiting factor in how much the aircraft can carry. Typically, military transports must be able carry several small helicopters, military trucks, or military cargo containers. On the civil side, the design aircraft must be able to compete effectively against such planes as the Boeing 757 and 767-300ER, which carry payloads of about 44,000 lbs. Similar to military transports, civil cargo transports must also have a fuselage large enough to carry bulky cargo containers. Likewise, the fuselage cargo bay must be accessible: small doors limit the size of cargo that the plane can transport.

The RFP dictates that civilian passenger variants must also be investigated. A passenger aircraft offers many challenges: civilian lives are entrusted to the safety of the aircraft; more requirements and regulations must be met. Provisions must be made for the crew, cargo storage, entrance and exit ways, multi-class seating, food, etc. Furthermore, an aircraft must earn peoples' trust as they are usually not readily willing to accept a new and radical design. A new aircraft design may also suffer higher acquisition costs than existing aircraft, because it must be certified and demonstrate flight-worthiness. Therefore, it may be wise to design an aircraft that meets the demands of a new market. An aircraft converted from the basic military airframe could satisfy needs in long-range, point-to-point, thin passenger markets. To be accepted by airlines for this market, the new aircraft must offer better performance at a lower cost to be competitive in the large civil passenger market.

As shown in NATO involvement in Kosovo, there is a need for aircraft to fulfill the roles of tanker and EW/ECM support. Currently, the KC-135 and KC-10 fulfill the tanker role for the US military, carrying 200,000 and 300,000 lbs of fuel for refueling operations, respectively (References 2.5, 2.6). To be a competitive replacement for these aging aircraft, a new tanker must be capable of carrying a comparable amount of fuel. The fuel capacity can be extended beyond the design TOGW by making the tanker variant a 2-g aircraft at take-off. This is accomplished by filling the aircraft with an amount of fuel that limits its structural limit loading to only 2-g's at take-off. After take-off, fuel is burned-off, reducing the aircraft weight and eventually bringing the aircraft back up to the required 2.5-g limit load factor. (For reference, the RFP requires that an aircraft fulfilling the CMC role be a 2.5-g aircraft.) The tanker role requires the addition of refueling drogues (Navy, Marines), booms (Air Force), and camera equipment for the boom operator. A single boom can be placed on the fuselage tail cone and additional drogues can be attached to hard points under the wings. If the tanker has three booms/drogues, up to three aircraft can be refueled at a time, thus speeding up refueling operations. A reduction in mission time can then be translated into a cost reduction.

Currently, the EW/ECM role is being fulfilled by the Navy's EA-6B aircraft. As aircraft fly into a combat situation an EW/ECM flies along with them in order to help prevent these aircraft from being detected by an enemy's radar. EW aircraft can also be used to jam enemy communications in a combat area. Today, existing aircraft such as the B-52 and the B-1 are being outfitted with EW equipment (Reference 2.7). Yet, more aircraft are needed to support offensive missions. One idea is to add EW provisions to an aircraft flying a CMC role as suggested by Fulgram: this would provide protection to the aircraft, while reducing the number of required support aircraft. Consequently, the risk of loss and cost of the mission will decrease. This is a possibility since a typical EW system weighs about 5000 lbs (Reference 2.8). By adding EW to a CMC aircraft, a standoff aircraft would have increased protection if it must penetrate enemy airspace to launch its missile load.

The final variation that must be investigated is the effects of flying missions of different ranges and payloads. By varying the design mission, the operational costs would be affected. The more versatile the aircraft, the more competitive it is and the better advantage it has against other aircraft.

2.2. Cost Effectiveness

One of the most important design drivers for this aircraft is cost effectiveness. Two methods were considered to increase the cost effectiveness of this concept over existing aircraft. One of these methods was to incorporate multirole capability. The second method was to increase the performance of the aircraft while keeping the costs low by incorporating advanced technologies.

Currently 70 B-52Hs are operational in the United States which represents the primary platform for delivering the AGM-86C. To replace the current fleet with the aircraft specified in the RFP, a minimum of 140 aircraft would have to be produced to sustain the current cruise missile delivery capability. Therefore, 460 aircraft of the 600 aircraft production run would be available for alternative missions. Creating 600 CMC carriers would not be an effective solution because the current military does not require this many aircraft. Currently the US Air Force has an aging fleet of tankers, a shortage of cargo aircraft, and a deficiency of EW aircraft. For these reasons the most effective design for this RFP would be a multirole capable aircraft. If the military was to decide to create two separate aircraft: a CMC and tanker/cargo platform the cost would be far greater than the production of a single airframe that could perform both tasks. Figure 2-1. shows the result of separate production runs for 140 CMC, 460 tanker/cargo aircraft, and 600 multirole aircraft. This figure shows that a multirole airframe would have a lower flyaway cost than both the dedicated CMC and dedicated tanker cargo aircraft because of the increased production run.

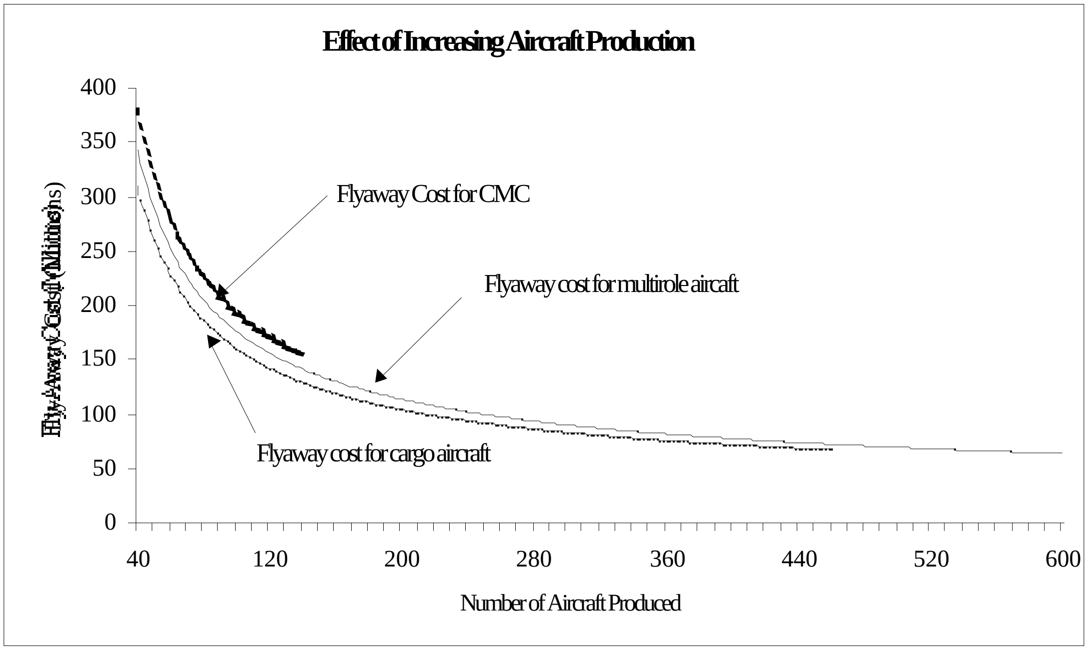


Figure 2-1 - Reduction of Flyaway Cost by Increasing Aircraft Production Runs

Table 2-1 shows the flyaway cost for the CMC, tanker/cargo, and multirole aircraft that were calculated in Figure 2-1.

Table 2-1 - Flyaway Cost for Single Role and Multirole Aircraft

Aircraft	CMC	Tanker/Cargo	Multirole
# of A/C produced	140	460	600
Flyaway Cost	154.12	67.21	63.57

* All cost in millions of US year 2000 dollars

Table 2-1 shows that if a multirole aircraft is used the flyaway cost can be reduced by nearly 90 million dollars for the CMC and by 4 million dollars for the tanker cargo version. Figure 2-2 shows a breakdown of the flyaway costs of 140 dedicated CMC and 460 dedicated tanker/cargo aircraft as well as 600 multirole aircraft capable of performing both roles. The CMC and Cargo group shows the combined cost of the CMC and cargo aircraft, which would have the same effectiveness as a single multirole aircraft. This figure shows that using a multirole airframe is the most cost effective means of meeting the current and future military and commercial customer needs.

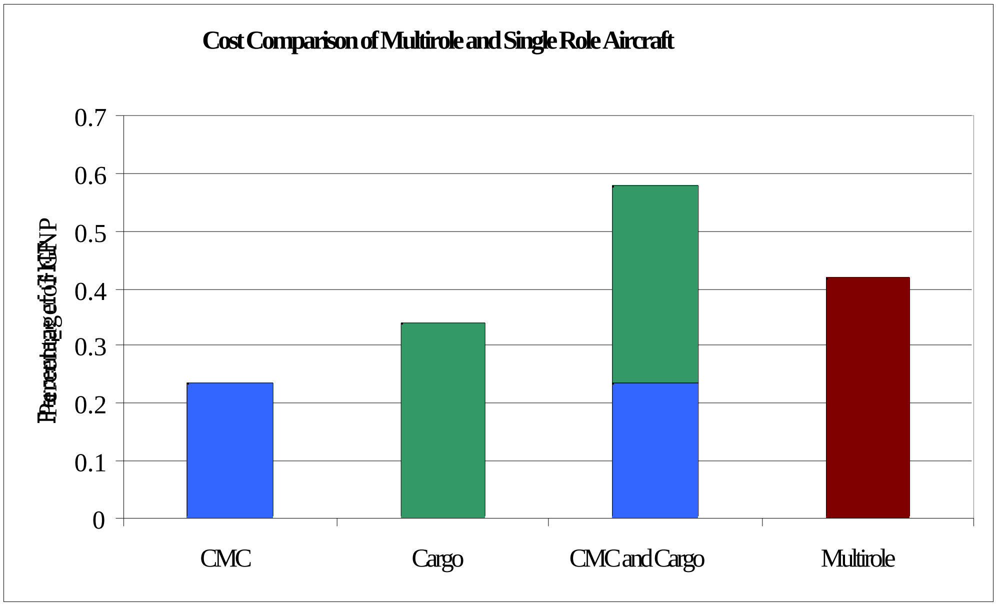


Figure 2-2 - Cost Reduction of Multirole Airframe Compared to Single-Use Airframes

Another method of increasing the cost effectiveness of an aircraft is to increase its performance over existing designs. Using new technologies are the most effective way to accomplish this goal. Using advanced aerodynamics could increase the fuel efficiency and reduce operating cost. The use of composite materials on the aircraft would reduce weight, decrease the maintenance times as well as decrease drag when used on surfaces. The main idea is to include as many advanced technologies to increase performance as much as possible while keeping a competitive flyaway cost as well as reduced operational cost compared to existing designs.

3. Concept Development and Selection

3.1. Initial Concepts

At the beginning of the design process, each team member investigated and developed an aircraft that would be considered for the preferred concept. After all of the concepts were submitted, it was found that there were five distinct ideas for an airplane that could successfully fulfill the role of a cruise missile carrier, and meet the RFP requirements. In the forthcoming sections these ideas will be presented and described in detail.

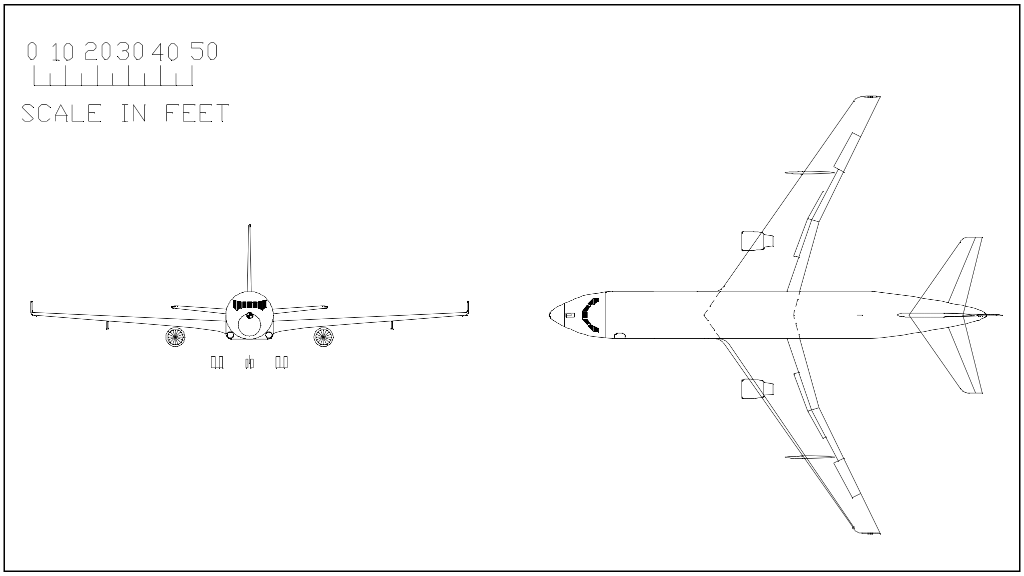


Figure 3-1 - Modified Existing Aircraft Concept

3.1.1. Modified Existing Aircraft Concept

The modified existing aircraft (MEA) concept, as illustrated in Figure 3-1, was a design driven by the desire for reliability and a cost-effective solution. The 767-300ER was chosen because of its design payload (45,238lbs) and range (5760 NM) both of which exceed RFP (Reference 3.1). In this design the 767-300ER was modified to accommodate ten cruise missiles. An unpressurized pannier as can be seen on the BAe Nimrod was attached to the bottom of the aircraft. The pannier was configured so that the required ten cruise missiles could be mounted inside. The advantage of this concept is that RDT&E costs will be greatly reduced, since the plane already exists. Another advantage of this design is that it lends itself well to the multi mission possibility. With its conventional design, there is plenty of cargo volume that could be used for commercial or military variants. The main setback to this idea is that the 767-300ER's cruise speed is Mach 0.82, which is below RFP requirements. To make this design meet the design cruise speed, it would be necessary to greatly modify the wing and/or re-engine the plane, or suffer the results of the increased specific fuel consumption. Another drawback to this concept is the 767-300ER's max takeoff weight, which is 400,000lbs. This capacity far exceeds the design requirement, and therefore lends itself to a much different market than is being sought.

3.1.2. New Conventional Design

The idea of modifying an existing aircraft, illustrated in Figure 3-2, requires that a suitable aircraft that meets the RFP requirements could be found. Rather than forcing an existing airplane to meet the RFP, a new conventional design was considered. This concept was considered to be a low risk design since the general arrangement is proven and has been used for four decades. By designing a new conventional aircraft, the plane can be specifically tailored to the RFP requirements. This concept uses a cylindrical fuselage with low-mounted wings and a T-tail. Twin turbofan engines would be mounted on the fuselage near the rear of the plane. The aircraft resembles a MD-90, but with a higher design TOGW. All ten missiles were mounted internally. The concept had many attributes that made it a cost-effective solution. Many existing aircraft employ a similar design to that which is proposed by this concept. Planes such as the BAC 111, DC-9, MD-90, and Boeing 727 offer proof that the design configuration is sound, minimizing the chance of unsuspected design problems. Another option was incorporate pre-existing parts such as wings, engines, tails, and landing gear, into the design to lower the acquisition of the cost of the airplane.

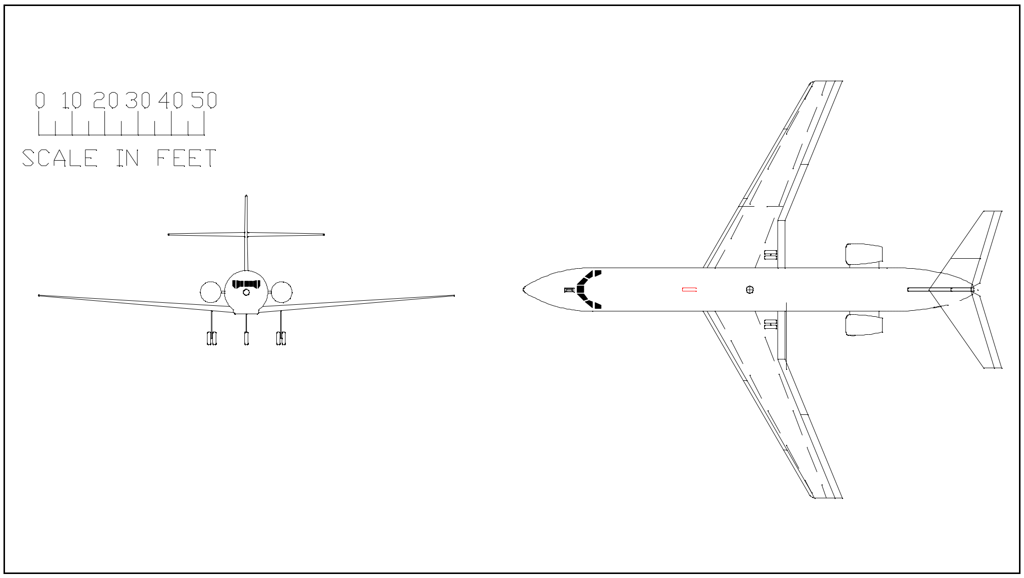


Figure 3-2 - New Convential Aircraft Concept

3.1.3. Strut-Braced Wing Concept

The preceding two concepts focused on adapting today's technology to the RFP. The next concept is a strut-braced wing (SBW), which uses new technology in an attempt to outperform today's transports. The strut-braced wing concept, pictured in Figure 3-3, was considered for this mission because research performed by NASA, Virginia Tech, and Lockheed Martin Aeronautical Systems (LMAS) (Reference 3.1) indicated that the strut-braced wing was an ideal solution for a transport with a high transonic cruise. The SBW concept featured a high wing mounted atop a cylindrical fuselage with a T-tail. A strut extended from the lower surface of each wing at 70% half span and was attached to a fuselage sponson. Twin turbofans were mounted on the wing at 32% half span. The SBW concept is a new design that resembles many conventional aircraft. The cylindrical fuselage in this design is comparable to the previous two configurations in terms of role versatility. This design incorporates large fuselage sponsons that house the landing gear, fuel, and missiles. The addition of the strut adds complexity to the design and would therefore raise RDT&E costs, but promises large life cycle cost savings. Research performed by Virginia Tech (Reference 3.1) showed that the best single strut-configuration had a 15% savings in TOGW, a 29% savings in fuel weight, 28% increase in L/D, and a 41% increase in seat-miles per gallon when compared to similar cantilever wing aircraft.

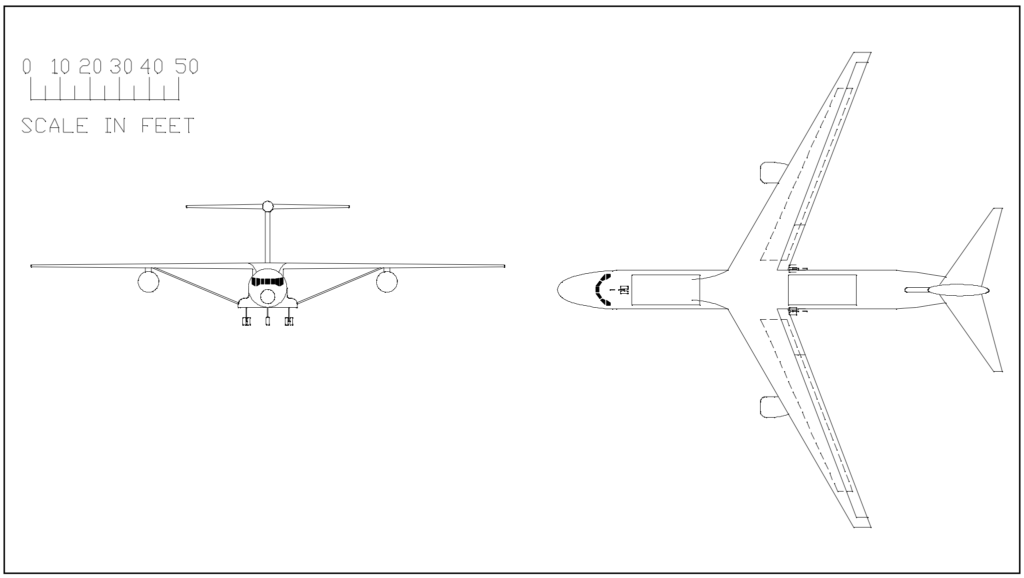


Figure 3-3 - Strut-Braced WIng Concet

3.1.4. Blended Wing Body Concept

Like the SBW concept, the blended wing body (BWB) concept, illustrated in Figure 3-4, is an unconventional design that capitalizes on new technology. The BWB concept was based research performed by NASA on the BWB aircraft (Reference 3.2). With the wing being blended into the fuselage, a large internal volume is created, which could be used for extra fuel or electronics. Two turbofans were mounted beneath the wing at 30% half span. The concept had two tandem weapons bays mounted in the fuselage that each held six cruise missiles. The BWB concept possessed limited role versatility due to the limited useable volume constraints imposed by its uniquely shaped fuselage. The concept could be used in tanker and electronic warfare roles. However the absence of cargo doors and a standard cargo bay eliminates the possibility of the cargo transport role especially for the vehicle size envisioned.

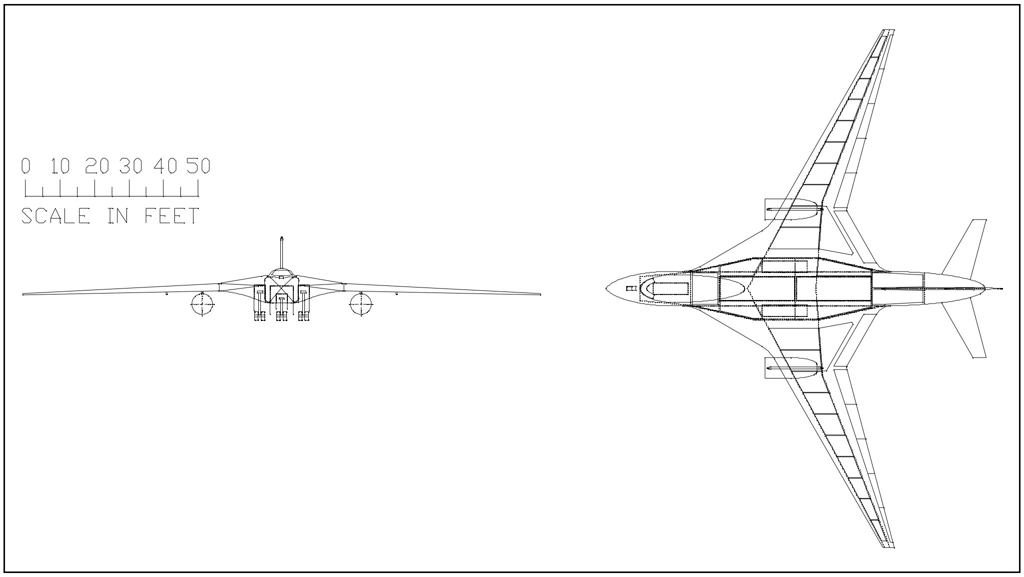


Figure 3-4 - Blended-Wing-Body Concept

3.1.5. Delta Wing Concept

The final concept considered was a delta wing aircraft, pictured in Figure 3-5. The aim of this design was a stealth aircraft that could penetrate into enemy territory undetected. Stealth characteristics incorporated into the design should increase its survivability in hostile environments. This concept employs a relatively short fuselage with a large area, low aspect ratio wing. Two tandem internal weapons bays house rotary launchers. To reduce radar cross-section, the leading edges of the aircraft were all straight, with a parallel-sided jagged trailing edge. Winglets were canted inward at a 20 degree angle to help lateral-directional stability and increase stealth. Snaking intake ducts direct air into four internally mounted low bypass turbofans. These ducts allow the fan blades of the engine to be hidden from enemy radar. The snaking ducts reduce the available internal volume, and therefore reduce the versatility of the aircraft. Other disadvantages to the delta wing concept are cost and the need for radar absorbing material and other requirements implied by stealth technology.

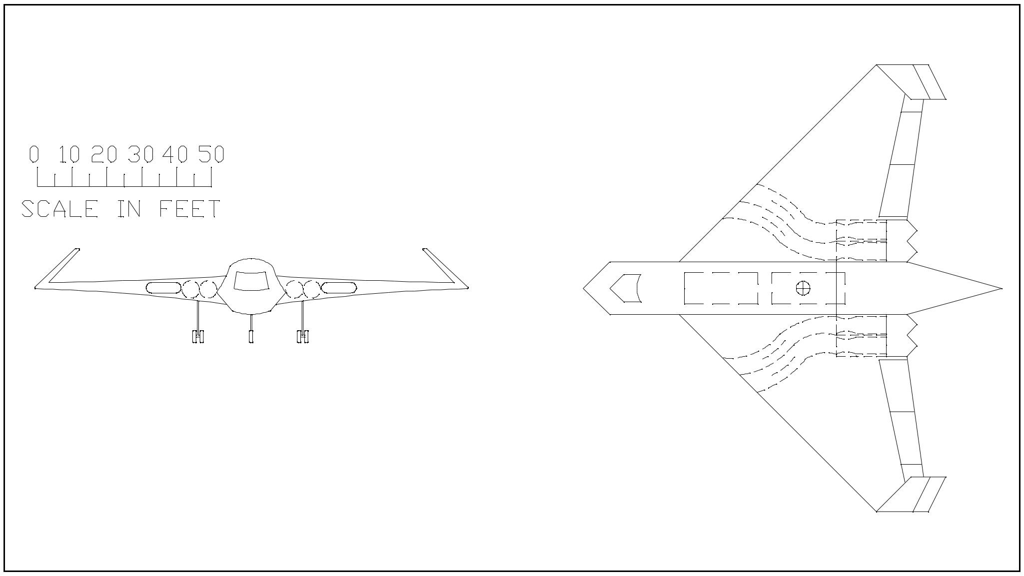


Figure 3-5 - Delta Wing Concept

3.2. Preliminary Concept Evaluation

After the initial concepts were defined a method had to be created to narrow the field of potential designs. Table 3-1 shows preliminary concept evaluations based on characteristics that are defined in the RFP. The major categories for the evaluation are: performance, multi-use, acquisition cost, operational cost, and stealth. The performance category rates the aircraft on its ability to perform as required in the RFP. The multi-use category rates the aircraft on its suitability for use in different types of aircraft. Acquisition cost and operational cost rank the aircraft based on the complexity of the system, advanced technologies used to reduce cost and the efficiency of the aircraft. From this table the MEA and SBW concept were selected for further study and evaluation because they offered the most promising solution to the RFP requirements.

3.3. Aircraft Effectiveness and Final Concept Selection

After the field of initial concepts was narrowed down to the MEA and SBW concepts a more specific evaluation was performed to select the most effective design of the two. Aircraft effectiveness is based on the performance of the design mission and additional requirements stated in the RFP. A method to evaluate the effectiveness was created to evaluate these concepts. The most effective concept was then chosen as the final concept.

Table 3-1 - Preliminary Concept Evaluation

	MEA	Blended Wing-Body	Strut-Braced Wing	Delta Wing	New Conventional
Take-off/Landing	**3**	3	4	2	3
Payload Weight	4	3	4	3	4
Range	4	4	4	2	4
Cruise Speed	3	3	3	5	3
Performance	**14**	**13**	**15**	**12**	**14**
Tanker	6	4	6	3	6
Cargo	4	2	4	1	4
Civil Passenger	6	1	5	1	6
Electronic Warfare	5	5	6	5	3
Multi-Use	**21**	**12**	**21**	**10**	**19**
Off Shelf Materials	5	2	4	1	4
Simple Design	10	3	6	3	8
Materials	2.5	1.5	2	1	2.5
Engine Number	2.5	2.5	2.5	1.25	1.25
Advanced Tech.	-1	-3	-2.5	-5	-1
Acquisition Cost	**19**	**6**	**12**	**1.25**	**14.75**
Fuel Efficiency	7	6	7	3	6
Maintenance	3	5	5	2	5
Reliability	5	4	5	3	5
Advanced Tech.	1	3	3	7	2
Crew Size	-1	-1	-1	-0.5	-1
Operational Cost	**15**	**17**	**19**	**14.5**	**17**
Stealth Tech.	0	0	0	10	0
Stealth	**0**	**0**	**0**	**10**	**0**
Total	**69**	**48**	**67**	**47.75**	**64.75**

To create a method of evaluating effectiveness, the design requirements were broken down into three major categories which were: performance, costs and mission capability. The representative characteristics that were used to analyze a concepts satisfaction of the requirements are shown in Table 3-2. The characteristics were chosen because they are the most critical parameters for the design requirements.

Table 3-2 - Effect iveness Drivers and Their Representative Characteristics

Performance	Costs	Mission Capability
Payload Weight Cruise Speed Range	Life Cycle Cost Acquisition Cost	Internal Volume Payload Weight Reliability

The performance characteristics were obtained from initial sizing of the concepts to ensure that minimum requirements could be met. The life cycle cost and the acquisition cost were obtained using a cost estimation algorithm based on historical data from Roskam (Reference 3.4). Mission capability characteristics were also obtained from initial sizing data as well as historical data. The mission capability category was used to estimate a concept's potential for multi-mission use. The ability of an aircraft to perform multiple missions increases its effectiveness because one aircraft is capable of replacing many other aircraft, which in turn, reduces total costs. The alternative missions considered were: military transport, commercial transport, passenger, tanker, electronic warfare (EW), and anti-submarine warfare (ASW).

Evaluation of the concepts was based on judging their relative effectiveness in satisfying the design requirements. Preliminary concept evaluation narrowed down the final concepts to the strut-braced wing (SBW) and modifying existing aircraft (MEA) concepts. These concepts were evaluated using an algorithm which quantifies the effectiveness of an aircraft design using an overall evaluation criteria (OEC). The OEC is defined using five categories, which include: affordability, survivability, readiness, capability, and safety (Reference 3.5). These categories were used because they were easily calculated and they are critical in meeting the design requirements. The OEC assumes that the minimum performance requirements such as cruise speed, payload weight, and range have been met. This assumption is valid because the concepts were not evaluated unless these requirements were met. The capability category was used to compare a concept's performance and it's ability to perform multiple missions which are based on the size of the aircraft.

The affordability of an aircraft was evaluated using the acquisition cost and the life cycle cost of the aircraft estimated from the cost algorithm developed by Roskam. This algorithm breaks the life cycle costs into four categories: RDT&E, acquisition, operational, and disposal. Table 3-3 shows the estimated cost associated with the SBW and the MEA concepts. Table 3-3 also shows the estimated cost breakdown for the two concepts assuming a production run of 600 aircraft and an average of 1200 flight hours per year.

Table 3-3 - Life Cycle Cost Breakdown

Costs	SBW concept	MEA concept
RDT&E	8174	5651
Acquisition (total)	44005	43604
Acquisition (per A/C)	83.6	82.1
Operational (total)	117296	123067
Operational (per flgt hr)	9557	9570
Disposal	1691	1740
Life Cycle Cost	169168	174063

*All costs indicated in millions of year 2000 dollars

The survivability of the aircraft is judged on the susceptibility and vulnerability of the aircraft to enemy defenses. The susceptibility and vulnerability are functions of the maneuverability, speed, size, decoys, and ECM gear. The survivability of the aircraft is an important quality because it again affects a wide range of things from life cycle cost (replacement cost for an aircraft and personnel with low survivability) to completion of the mission. A quantitative procedure for defining the survivability is shown below:

$$P_{surv} = [1-(P_D \bullet P_{H/D} \bullet P_{K/H})] \tag{3.1}$$

Where Psurv is the probability of surviving, P_D is the probability of detection, $P_{H/D}$ is the probability of being hit if detected, and $P_{K/H}$ is the probability of being killed if hit. The probability of survival was calculated using data on the effectiveness of the ECM gear as well as historical data for aircraft lost after being hit by enemy fire.

The capability of an aircraft is the measure of the aircraft's ability to complete its given mission as well as the potential for multipurpose capability. The mission capability index (MCI) is used to evaluate the concept's capability. The MCI quantifies how much payload can be delivered to a location and how this capability effects the size and weight. The equation for MCI is shown below:

$$MCI = (W_{payload} \bullet Range)/(W_{fuel} + W_{empty}) \tag{3.2}$$

The MCI increases as the payload weight and range increase, and decreases with an increase in fuel weight and empty weight. This index judges efficiency and size, which are critical in the performance of both the primary and secondary missions.

The final two characteristics, readiness and safety, are the most difficult to quantify because they rely solely on historical data. Since one of the final concepts has never flown before, the values must be extrapolated. The first of these to be examined is operational safety, which is a measure of aircraft losses. Aircraft accident data is usually shown in loss rates (accidents/100,000 flight hrs) and these loss rates tend to fluctuate based on the maturity of the aircraft, technology level, mission type, etc (Reference 3.6). The reason for aircraft losses are usually unknown in wartime, so only peacetime loss rates were used (Reference 3.5). Engine failure dominates the attrition profiles, so it was used as a measure of operational safety. This measure is known as the engine-caused attrition index (EAI), which is given below:

$$EAI = (\#\ of\ A/C\ procured - \#\ of\ Engine\text{-}Caused\ Losses) / \#\ of\ A/C\ procured \quad (3.3)$$

The final metric is operational readiness, which is a measurement of the amount of time a weapon system is ready and capable to perform its design mission. To quantify this area the inherent availability of the aircraft is used which is defined as "a measure of the degree to which an item is in the operable and committal state at the start of the mission when the mission is called for, at an unknown time" (Reference 3.4). The measure of inherent availability is a function of the system reliability and maintainability. The equation for the inherent availability is given below:

Where MTBF is the mean time between failure and MTTR is the mean time to repair. "For new concepts such data can be carefully inferred from data of modern aircraft of similar types" (Reference 3.5).

$$A_i = MTBF_i / (MTBF_i + MTTR_o) \quad (3.4)$$

Now that each part of the evaluation criteria has been defined and calculated they must be combined to calculate the OEC. The OEC is composed of weighted values of the affordability, survivability, readiness, capability, and safety. To achieve an accurate evaluation the results of the five categories are normalized with baseline values. The equation for the OEC is given below:

$$OEC = \alpha\left(\frac{LCC_{BL}}{LCC}\right) + \beta\left(\frac{MCI}{MCI_{BL}}\right) + \chi\left(\frac{EAI}{EAI_{BL}}\right) + \delta\left(\frac{P_{SURV}}{P_{SURV_{BL}}}\right) + \varepsilon\left(\frac{A_i}{A_{i_{BL}}}\right) \quad (3.5)$$

The coefficients $\alpha, \beta, \chi, \delta$, and ε are weighting factors and determine the percentage that each quantity contributes to the OEC. These weighting factors are determined based on the relative importance of the given metric based on the requirements. In the evaluation for the cruise missile carrier, the baseline aircraft was selected to be the B-52H because it currently performs a primary mission similar to that of the concept aircraft.

To perform the most effective evaluation possible, a range of values were used as the weighting factors. The values of the weight factors were determined using the design requirements. The weighting factors representing life-cycle cost, capability, and survivability were varied over a wider range than safety and readiness. This was done so that the most important factors were varied to show how they effect the OEC. The result of varying the weighting factors in the OEC is an evaluation matrix. Table 3-4 shows the evaluation matrix for the SBW.

Table 3-4 - SBW Concept Evaluation Matrix

			α_1			α_2			α_3		
			β_1	β_2	β_3	β_1	β_2	β_3	β_1	β_2	β_3
ε_1	δ_1	χ_1	-	-	-	-	-	-	-	-	0.9057
		χ_2	-	-	-	-	-	0.9107	-	0.9692	-
		χ_3	-	-	0.9156	-	0.9726	-	1.0296	-	-
	δ_2	χ_1	-	-	-	-	-	0.8922	-	0.9523	-
		χ_2	-	-	0.8972	-	0.9557	-	1.0142	-	-
		χ_3	-	0.9591	-	1.0161	-	-	-	-	-

Weighting Factor	**Represents**	**Values**
$\alpha_1, \alpha_2, \alpha_3$	Life Cycle Cost	0.2, 0.25, 0.3
$\beta_1, \beta_2, \beta_3$	Capability	0.15, 0.2, 0.25
χ_1, χ_2, χ_3	Survivability	0.15, 0.2, 0.25
δ_1, δ_2	Operational Safety	0.15, 0.2
ε_1	Operational Readiness	0.15

The boxes with a slash indicate that the weighting factors did not sum to one hundred percent, and therefore were not considered in the evaluation. The boxes with numbers indicate the OEC score for the SBW concept with the B-52H used as the baseline aircraft. Each box in the matrix is associated with a weighting factor that was used in the OEC which was then used to calculate the score. The associated weighting factors can be found in the box below the matrix. The physical characteristics of the SBW concept were obtained from an optimization code created by Joel Grasmeyer (Reference 3.3). The scores obtained in this evaluation were used as raw scores to compare final two concepts. A similar concept evaluation matrix was created for the MEA concept, which was based on the 767-300 ER. The differences between these evaluations were calculated and are shown in Table 3-5

The survivability of the two concepts would be about the same because the aircraft are of similar type and would carry similar defense systems. The readiness and the safety of the SBW concept were affected because of the immaturity of the aircraft system. The MTTF and MTTR values were larger for the SBW concept compared to the MEA concept because of the lack of experience with this type of design. The attrition rate for the SBW concept was higher because it is an unproven design. The effect of multi-purpose capability in this comparison was small because both aircraft were designed or selected to be able to perform all of the desired secondary missions. This table shows that even with increased repair time and attrition rates the SBW concept was superior over a greater range of weighting factors. The MEA concept was superior to the SBW concept only when the cost was ranked as the highest weighting factor. One major problem with the MEA concept was ensuring that it would be able to meet the cruise speed requirement. Very few existing aircraft in this size range could meet this requirement which made this option less attractive for the final concept. The 767-300 has a cruise speed of mach 0.82, which does not meet the RFP requirements (Reference 3.6).

Table 3-5 - First Concept Comparison Matrix

			α_1			α_2			α_3		
			β_1	β_2	β_3	β_1	β_2	β_3	β_1	β_2	β_3
ε_1	δ_1	χ_1	-	-	-	-	-	-	-	-	0.0012 MEA
		χ_2	-	-	-	-	-	0.0087 SBW	-	0.0002 SBW	-
		χ_3	-	-	0.0185 SBW	-	0.0087 SBW	-	0.0012 MEA	-	-
	δ_2	χ_1	-	-	-	-	-	0.0011 SBW	-	0.0059 MEA	-
		χ_2	-	-	0.0110 SBW	-	0.0025 SBW	-	0.0059 MEA	-	-
		χ_3	-	0.0110 SBW	-	0.0011 SBW	-	-	-	-	-

SBW - indicates strut braced wing won by indicated margin
MEA – indicates modifying 767-300 ER won by the indicated margin

Weighting Factor	**Represents**	**Values**
α_1, α_2, α_3	Life Cycle Cost	0.2, 0.25, 0.3
β_1, β_2, β_3	Capability	0.15, 0.2, 0.25
χ_1, χ_2, χ_3	Survivability	0.15, 0.2, 0.25
δ_1, δ_2	Operational Safety	0.15, 0.2
ε_1	Operational Readiness	0.15

The first concept matrix did not take into account the fact that the repair cost, and attrition rate would decrease as the system matured. Because the first matrix was conservative, a second concept matrix was created to make a more realistic account of the decrease in cost. Table 3-6 shows the results of the second evaluation.

Table 3-6 - Second Concept Comparison Matrix

			α_1			α_2			α_3		
			β_1	β_2	β_3	β_1	β_2	β_3	β_1	β_2	β_3
ε_1	δ_1	χ_1	-	-	-	-	-	-	-	-	0.0285 SBW
		χ_2	-	-	-	-	-	0.0368 SBW	-	0.0322 SBW	-
		χ_3	-	-	0.0452 SBW	-	0.0391 SBW	-	0.0331 SBW	-	-
	δ_2	χ_1	-	-	-	-	-	0.0270 SBW	-	0.0237 SBW	-
		χ_2	-	-	0.0353 SBW	-	0.0307 SBW	-	0.0260 SBW	-	-
		χ_3	-	0.0376 SBW	-	0.0316 SBW	-	-	-	-	-

SBW - indicates strut braced wing won by indicated margin
MEA – INDICATES MODIFYING 767-300 ER WON BY THE INDICATED MARGIN

Weighting Factor	Represents	Values
α_1, α_2, α_3	Life Cycle Cost	0.2, 0.25, 0.3
β_1, β_2, β_3	Capability	0.15, 0.2, 0.25
χ_1, χ_2, χ_3	Survivability	0.15, 0.2, 0.25
δ_1, δ_2	Operational Safety	0.15, 0.2
ε_1	Operational Readiness	0.15

This table shows when the decrease in repair cost and attrition rates over time are taken into account, the SBW concept is the superior concept over all weighting factors. Therefore, the SBW concept is the most effective and was selected as the final concept.

A concept decision tree, presented in Figure 3-6, shows how the initial concepts evolved into the final Vulture concept.

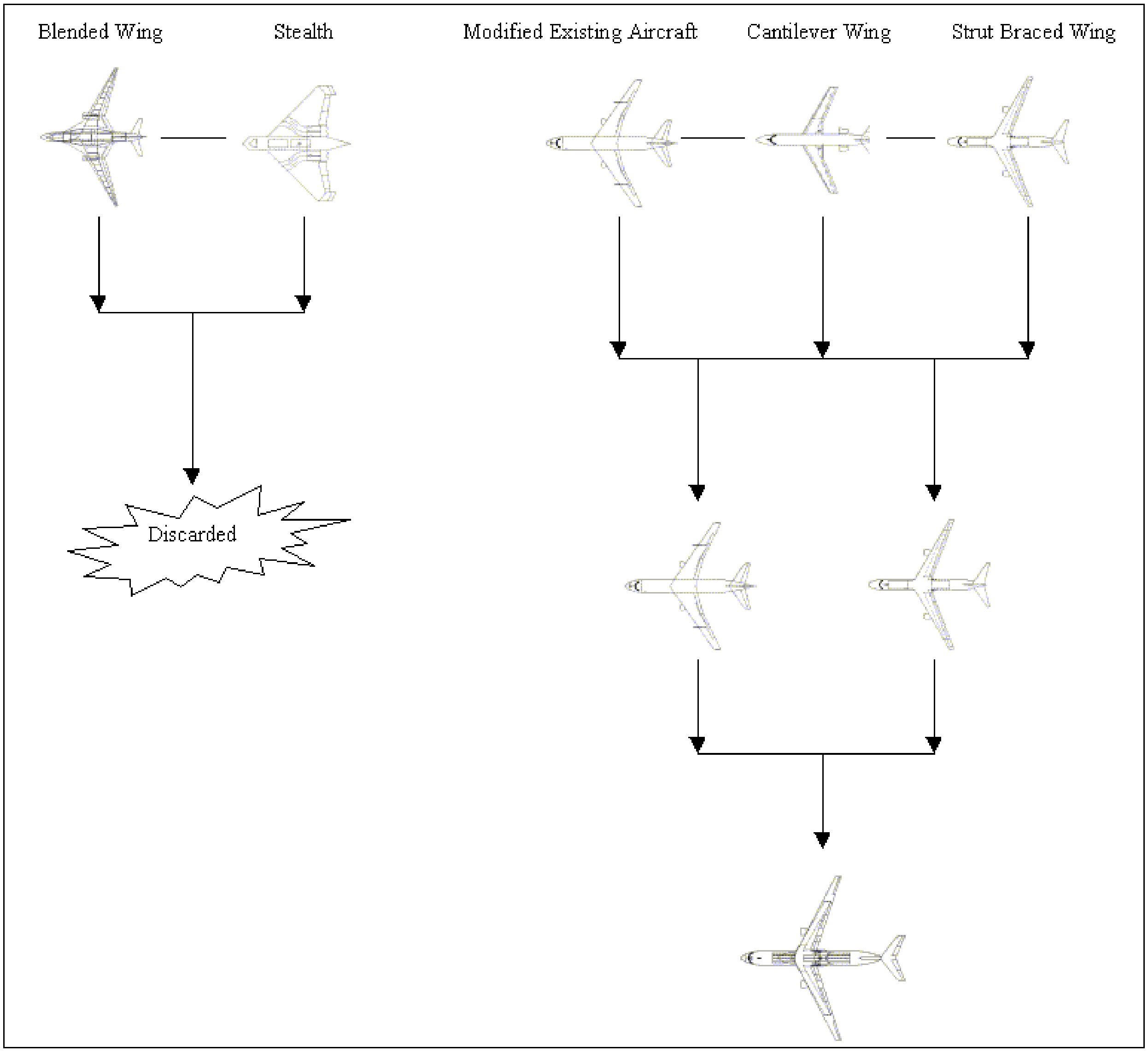


Figure 3-6 - Concept Decision Tree

4. Multi Disciplinary Optimization

4.1. Explanation Of The Code

Multidisciplinary optimization (MDO) was used to create an optimal design for the Vulture by evaluating aerodynamics, structures, stability and controls, performance, and propulsion simultaneously. Using MDO, the result is a minimum weight airplane that correctly integrates all the different disciplines. An MDO computer code, which has been developed by Virginia Tech in conjunction with NASA and Lockheed Martin, was used to size the final design. This computer code has the capability of accurately calculating the optimal design point of both SBW and standard cantilever wing aircraft. A flow chart showing the basic structure of the MDO code can be seen below, in Figure 4-1.

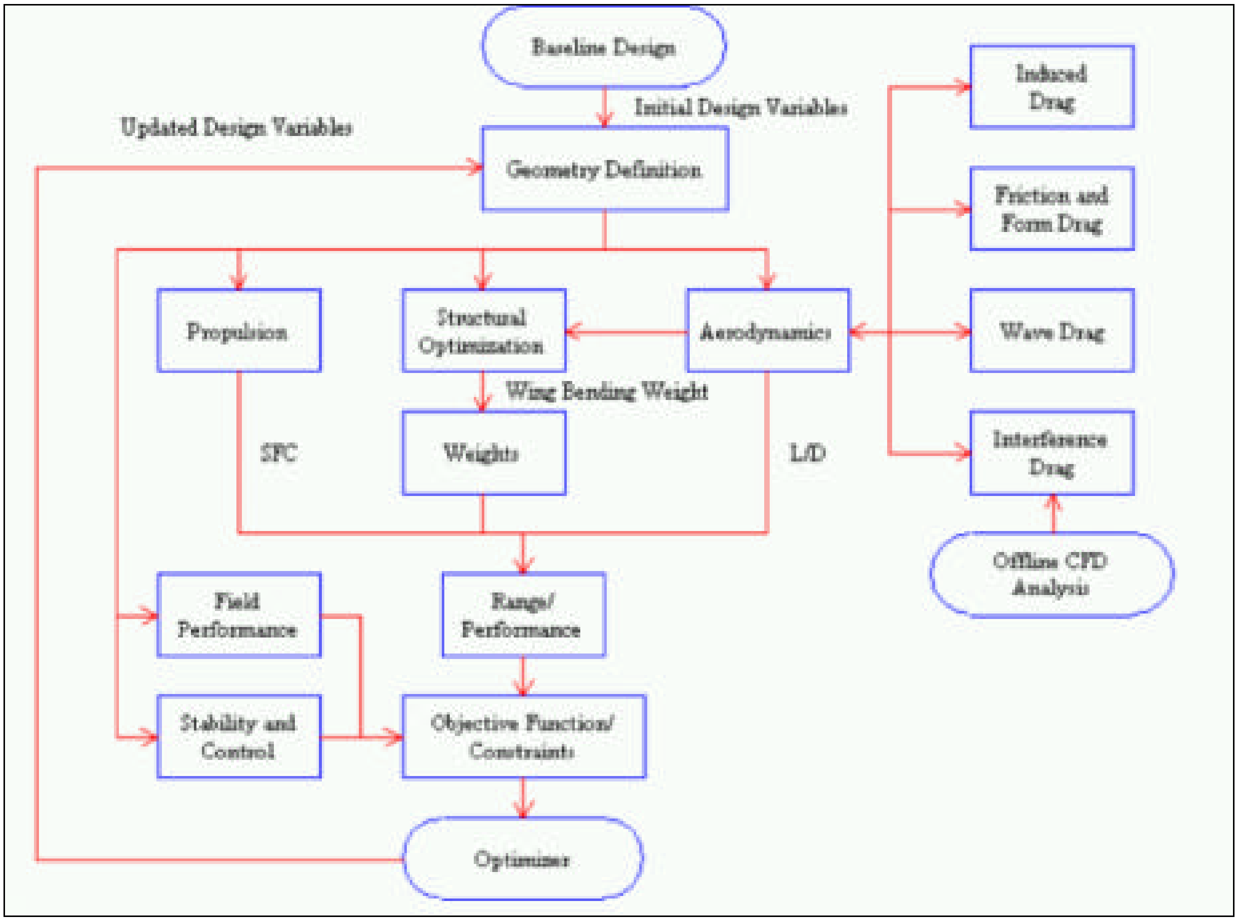


Figure 4-1 - Flow Chart Showing Structure of the MDO Code

To obtain the optimal design, the MDO code uses 20 design variables which are shown in Table 4-1. This allows the code to optimize the wing planform as well as the engine thrust.

Table 4-1 - Design Variables

· Semi-Span Of Wing – Strut Intersection	· Wing Thickness At Strut Intersection
· Wing Span	· Wing Tip Thickness
· Wing Quarter Chord Sweep	· Strut Thickness
· Strut Quarter Chord Sweep	· Skin Thickness
· Strut Chordwise Offset	· Strut Force
· Strut Vertical Offset	· Fuel Weight
· Wing Root Chord	· Zero Fuel Weight
· Wing Tip Chord	· Required Thrust
· Strut Chord	· Engine Location
· Wing Root Thickness	· Cruise Altitude

Twelve constrains are implemented in the MDO code in order to ensure a feasible design. These constraints are listed in Table 4-2.

Table 4-2 - Design Constraints

· Zero Fuel Weight Convergence
· Calculated Range > Design Range
· Rate Of Climb At Initial Cruise > 500 ft/min
· Cruise Section $C_L < .8$
· Fuel Weight < Fuel Capacity
· C_n Available > C_n Required
· Wing Tip Deflection < Max Wing Tip Deflection At Taxi Bump
· Second Segment Climb Gradient > 2.4%
· Balanced Field Length < 8,000 ft
· Approach Velocity < 140 knots
· Missed Approach Climb Gradient > 2.1%
· Landing Distance < 6,000 ft

If any of the above constraints are violated during iteration, the program updates the design variables, and returns to find an optimal design that is deemed to be feasible by meeting each one of the above constraints.

4.2. Payload Weight Determination

To begin the optimization process it was necessary to decide upon a design payload weight. The RFP requires that the payload weight must be at least 32,500 lbs, to carry 10 CALCMs. The team had to decide whether it would design solely around this minimum payload of 32,500 lbs, or whether a greater payload weight would be used. On the one hand, a payload of 32,500 lbs meant that the design aircraft could be optimized for the CMC mission. This would result in a smaller aircraft, costing less to manufacture and costing less to operate (less fuel required). However, as described earlier, other aircraft that this concept would be competing against in other roles (the KC-135, KC-10, C-141, Boeing 767 Freighter) have payload weights greater than the 32,500 lbs minimum payload weight. If this aircraft design is accepted, it must be able to compete and perform competitively with existing aircraft. If the design cannot offer increased performance at decreased weight compared to existing aircraft, it will not sell. Also, if the aircraft could perform two or more roles simultaneously, such as fly a CMC mission and serve as a EW/ECM aircraft too, it would require a greater payload weight than the minimum 32,500 lbs. After much study, it was decided that optimizing the plane around the design CMC/EW/ECM mission would take precedence. A final payload weight of 40,000 lbs was therefore decided upon. While optimizing the design at close to the minimum CALCM payload weight, this payload weight would also allow for the integration of a 5,000-lb

EW/ECM system to perform both roles simultaneously. Also, at reduced ranges, the payload weight would increase and the aircraft would be competitive with the C-130J which can transport 45,000 lbs only 2800nm.

4.3. Design Optimization

One of the first decisions the team encountered in the design process was where to place the engines. The options considered were: fuselage mounted engines, inboard wing mounted engines, pylon mounted engines, and tip mounted engines. Each of these locations presents their own advantages and disadvantages, and therefore required further investigation. Fuselage mounted engines allow for clean flow over the strut, but may realize disturbed flow into the intake coming from the wake of the wing. Engines mounted on the inboard section of the wing allow for clean engine intake flow and provide inertial relief, but they introduce disturbed air over the strut. Strut-pylon mounted engines increase the inertial relief generated, as well as allowing clean airflow over the strut, but under engine out conditions there would be a large yawing moment. Tip mounted engines provide the same positive aspects of the pylon mounted engines, but create an even larger yawing moment under engine out conditions. Tip mounted engines could reduce the vortex coming off the wing tip. The MDO code was run allowing it to optimize a design for each engine location and the results were recorded below in Table 4-3.

Table 4-3 - Weights for Different Engine Locations

	Fuselage Mounted	Inboard Wing Mounted	Tip Mounted
TOGW, lbs	185,371	181,742	205,921
Zero Fuel Weight, lbs	128,746	125,599	118,339
Fuel Weight, lbs	56,625	56,143	87,582
Wing Weight, lbs	21,067	20,445	14,081

When the engines were placed on the tip of the wing, the optimizer had a difficult time finding a feasible solution. This arose from the engine-out constraint. Balanced field length, engine out, and second segment climb gradient became severe problems in the design optimization leading to a drastic increase in aircraft weight. Either of these engine placements would require the use of advanced technologies such as circulation control on the vertical tail, translating into a higher cost. Based on this study, it was decided that the best choice would be to mount the engines under the inboard section of the wing. The inboard wing mounted engines reduced TOGW and fuel weight. This relates to a decrease in both acquisition and life cycle costs. The position of the engines determined by the MDO code was approximately 32% of the half span. This is approximately the same engine location as would be used on standard cantilever wing aircraft.

With the engine placement determined, the code was utilized to decide upon the optimal design point. It became apparent that the size of the wing was being driven by the amount of fuel having to be stored in the wing. With wing size being driven by the fuel weight, the wing sizing did not realize the theoretical advantages of the SBW, i.e. low thickness to chord ratios and less wing sweep. So another decision was to be made as to the wing size to employ, one where all of the fuel was stored in the wings, or where some of the fuel would be housed in the fuselage to allow optimal size and geometry.

With the results of these two optimizations, presented in Table 4-4, it was decided to house some of the fuel in the fuselage. While this presents a problem of allotting room for the relocated fuel, it will allow for a thinner wing, as well as a reduction in fuel weight due to increased performance. This makes a concept aircraft more appealing, since it demonstrates configuration advantages such as the lower wing thickness and the lower sweep back angle not available on current competitor aircraft. The reduction in fuel weight results in cost savings further proving the SBW concept is superior when compared to existing aircraft.

Table 4-4 - Weight Differences Based on Fuel Placement

	All Fuel In Wings	**Fuel In Fuselage And Wings**
Wing Quarter Chord Sweep	28^{o}	25^{o}
Wing Root t/c	12.7	10.1
Wing Break t/c	7.0	6.6
Wing Tip t/c	9.0	9.0
Wing Weight, lbs	20,726	20,445
Fuel Weight, lbs	56,524	56,143
Zero Fuel Weight, lbs	125,981	125,599
TOGW, lbs	182,505	181,742

With the major decisions made, the optimizer was run several more times to ensure a true optimal point had been reached. Upon completing these runs a carpet plot was constructed to show the optimal design point as well as show the sensitivities of thrust to weight ratio and wing loading at TOGW. The final design point had a thrust to weight ratio of .279 and a wing loading of 101.4 pounds per square foot. As seen in Figure 4-2, the minimal TOGW is 182,000 lbs occuring at the above wing loading and thrust to weight ratio. As seen by the added constraint lines, the balanced field length and engine out conditions increase the design thrust to weight ratio.

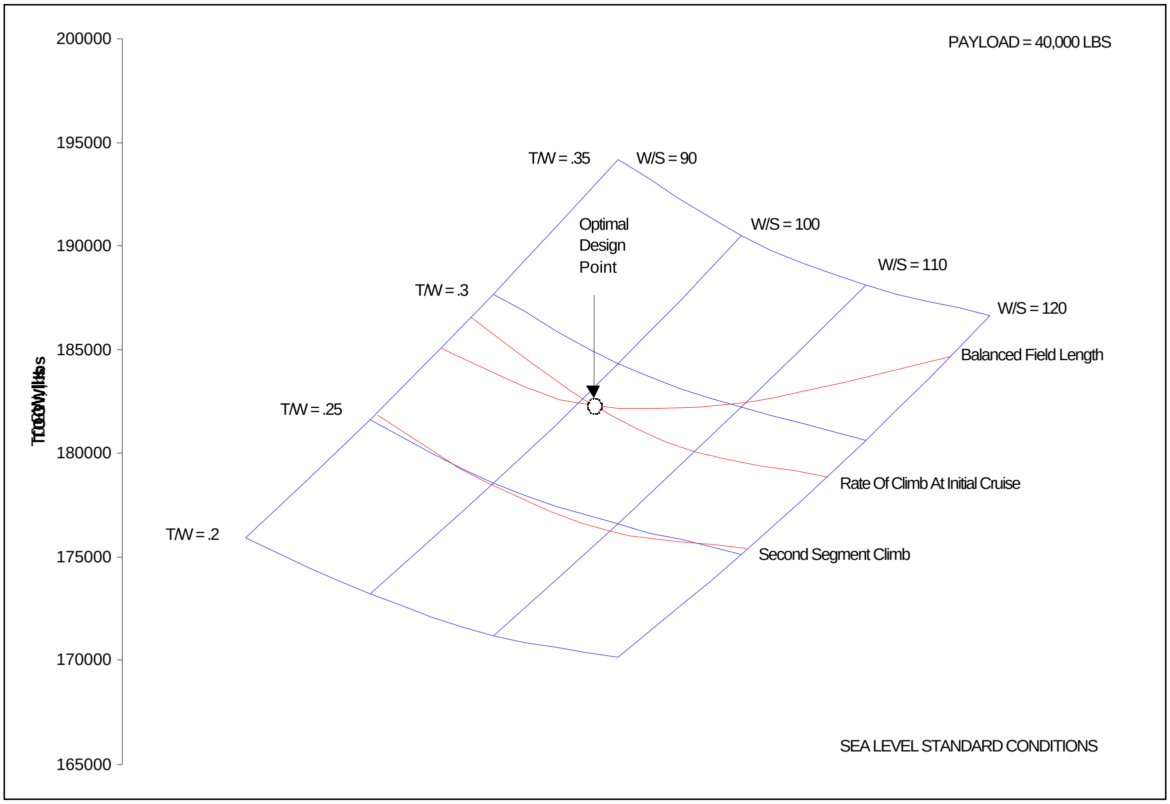


Figure 4-2 - Carpet Plot Showing Optimal Design Point for the SBW Design

In order to show that a competitive advantage exists with respect to the standard cantilever wing, it was necessary to prove the ability of the SBW to decrease TOGW from the strut-induced load reduction on the wing. A study was therefore performed, using the MDO code, to compare the SBW concept and the standard cantilever wing. The results are shown in Table 4-5.

Table 4-5 - Weight Differences Between SBW and Comparative Cantilever Wing

	SBW	**Cantilever**
TOGW, lbs	181,742	196,070
Zero Fuel Weight, lbs	125,599	132,534
Fuel Weight, lbs	56,143	62,536
Wing Weight, lbs	20,445	24,290

The SBW demonstrates a weight saving of 14,328 lbs in TOGW, as well as a fuel saving of 6,393 lbs, translating into a weight savings of 7.3% over the advanced technology cantilever wing. These numbers show the advantage of the SBW concept.

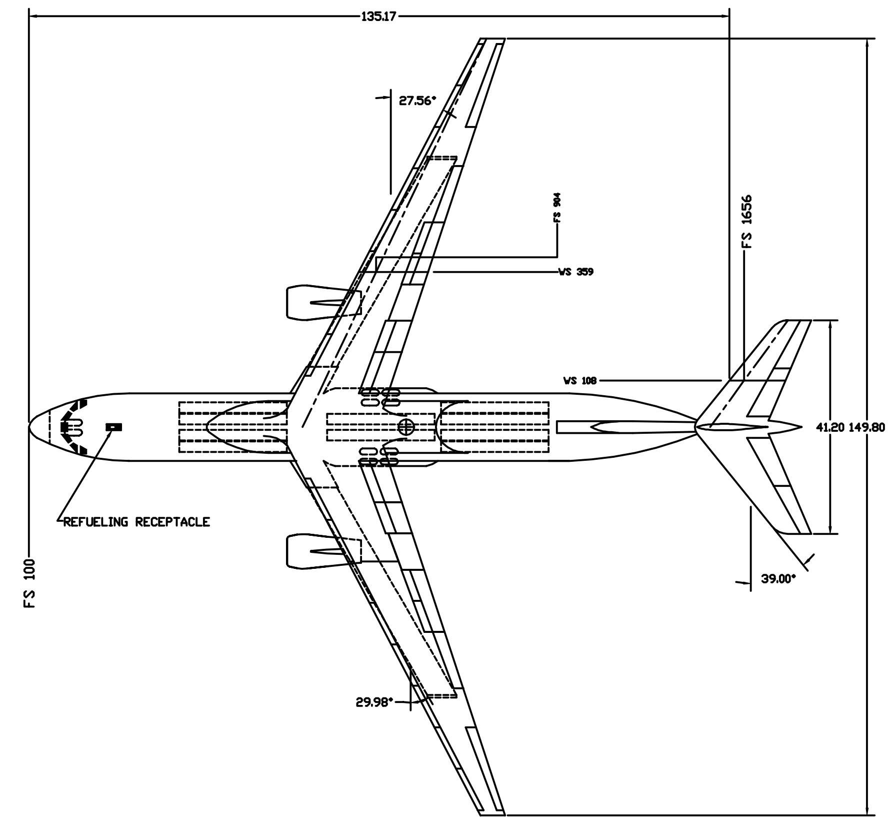
The cantilever wing shows the need for an increase in thrust to weight to ratio at a lower wing loading, as well as a significantly higher TOGW when compared to the SBW. The optimal thrust to weight ratio was found to be .286 with a wing loading of 98.4 pounds per square foot.

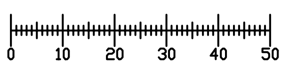
In conclusion, the MDO code proved to be a solid tool in the optimization of the SBW concept. It showed the possibility of a tremendous weight reduction over current aircraft. With the assumption that weight is proportional to cost, this translates into a large savings in purchase and life cycle cost.

5. Configuration

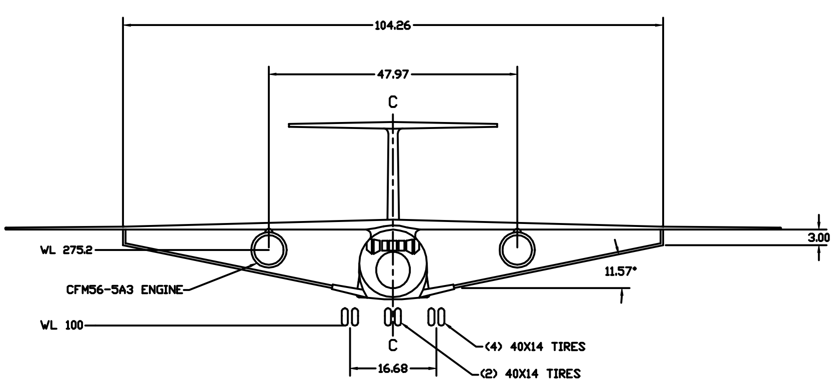
Figure 5-1 is a 3-view drawing of the general arrangement of the Vulture.

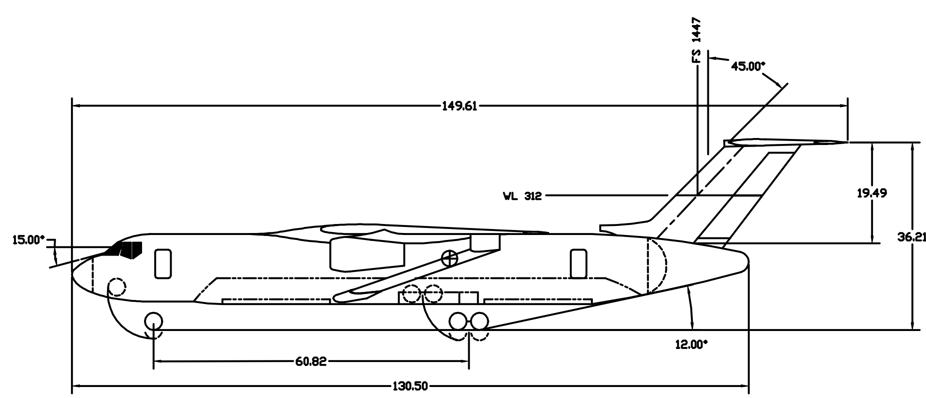
		WING	HORIZ	VERT
AREA	FT^2	1792.36	424.98	323.91
ASPECT RATIO		12.52	4.0	1.17
TAPER RATIO		.25	.45	.75
SPAN	FT	149.8	41.2	19.49
ROOT CHORD	FT	19.18	14.23	19.02
MAC	FT	13.42	10.8	16.47
TIP CHORD	FT	4.75	6.4	14.23
THICKNESS ROOT/TIP	%	.101/.090	.09	.12
SWEEPBACK AT .25C	DEG	25.35	36.11	43.18





SCALE IN FEET





TEAM VULTURE	GENERAL ARRANGEMENT CRUISE MISSILE CARRIER	FIG 5.1

6. Payload Configuration

The multi-role nature of the aircraft concept presented numerous challenges in configuring the payload carriage, in addition to deciding the basic payload weight. The carriage and launch system of the CALCMs had to be decided upon. The multirole effectiveness of the carriage and launch system also had to be evaluated. If not, would extensive modifications or multiple airframes need to be constructed. What other roles would this design perform. Upon answering these questions, an optimal payload configuration could be constructed.

6.1. Weapons Bay Configuration

The configuration of the weapons bay was the next challenge. How would these CALCMs be stored and then launched? The first concept investigated was to use a conventional rotary launcher. The rotary launcher is currently used to launch CALCMs from B-52s, B-1s, and B-2s. The rotary launcher system allows all the missiles to be connected, ready to launch. It is also able to re-index the missiles if a missile cannot launch. However, several drawbacks exist: the only rotary missile launcher available carries and launches only eight missiles, meaning that two missiles would need to be stored elsewhere. The rotary launcher system is also large, requiring long landing gear to allow the launcher to be lifted up into the fuselage from below, and requiring a large, centrally-mounted weapons bay that would not be useful for other roles. The system is also weighty, requiring bulkheads to be located on either side of the launcher.

Another launching system concept was to store the missiles in a stacked system two missiles wide and five missiles high. This would reduce weight, but would still require a large weapons bay, again centrally mounted, that did lend itself well to cargo or transport roles without extensive fuselage modifications. Furthermore, no feasible solution could be found to launch the missiles: if the missiles were dropped from their weapon racks, turbulence inside the weapons bay could cause the missile to tumble within the bay and damage the missile or aircraft.

A third idea was to use one airframe with a tube launched weapons system, which used a charge to “kick” the cruise missile, which was housed in a sabot, out of a tube at the rear of the aircraft. This concept utilized a complex system of rollers, which would allow the cruise missiles to be moved into launching positions. Computerized machines would move the missiles from their stored positions to their launch positions. The tubes would not have to be removed for the secondary missions because they would only protrude three feet into usable cargo area. A pro-con chart of the tube launch system is shown in Table 6-1.

Table 6-1 - Pro-Con Chart of Tube Launch Concept

Pro	Con
· The weapons area required few changes to accommodate the secondary missions. · Would not require any external additions to the fuselage to carry out the primary mission · Less maintenance required · No requirement for large bomb bay doors	· Roller system to maneuver the cruise missiles would be very complex, heavy, costly · Unproven launch dynamic on AGM-86C cruise missile · The concept used sabots which encased the missile while it was being launched from the tube hence an increase weight and cost · Missile jams within tubes could slow or prevent missile launch · Alternative to umbilical connection between missiles and aircraft required · Increased structural weight where the tubes penetrated the pressure bulkhead and the fuselage skin

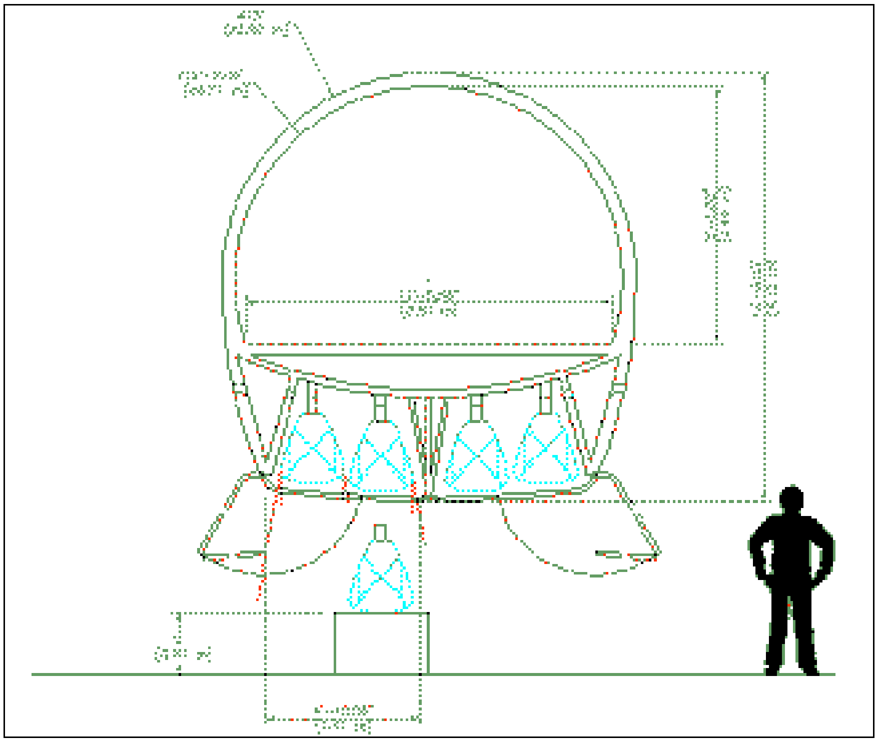


Figure 6-1 - Cross Sectional View of the Final Weapons Bay

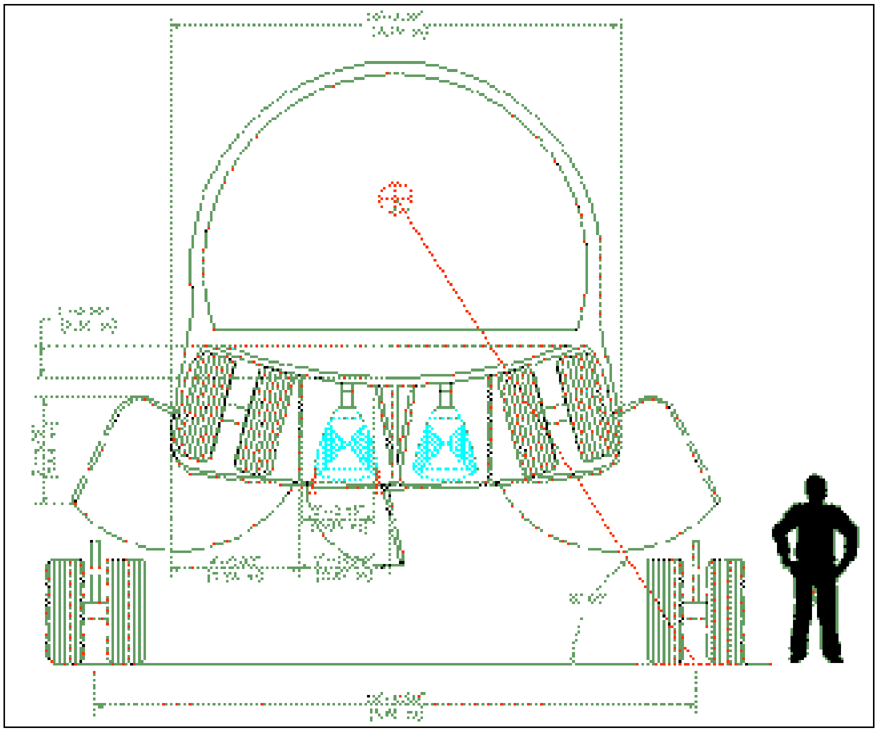


Figure 6-2 - Cross Sectional View of the Aft Landing Gear Bay

The final weapons bay concept was to create an inviolate, pressurized cargo bay area, under which the cruise missiles would be stored in their own close-fitting unpressurized missile bays. A pannier or sponson no larger in cross section than the main landing gear “bulge” would be extended along the length of the fuselage that would fit up to four missiles across. A cross sectional view the fuselage for this concept is Figure 6-1 and Figure 6-2.

The shallow close-fitting missile bays would virtually eliminate any chance for the ejected missile to get caught in turbulence when launched. If a missile became jammed, only one missile would be unexpended. This system is similar to that used on the British Nimrod. When not performing CMC missions, the missile bays could be outfitted with refueling tanks or “speedpak” containers. The only drawback to this configuration is that the aircraft design would only be able to carry and launch 10 CALCMs internally, with the possibility of two overload CALCMs mounted outboard on the wings. A pro-con chart for the weapons bay concept is shown in Table 6-2.

Table 6-2 - Pro-Con Chart for the Final Weapons Bay Concept

Pro	Con
· Allows primary and secondary missions to be performed without the removal of the weapons system · One or two missiles housed in a "separate" bay which ensures the primary mission will not be compromised by one missile jam · Standard umbilical connection to missile · Proven loading and ejection system	· The weapons bay are and the sponson are not utilized completely during the secondary missions · Increased structural weight from the weapons bays and the sponson · Increased drag from the addition of the sponson

After thoroughly investigating each of the above weapons bay configurations, it became apparent that the separate missile bay concept would be the best choice. It utilizes already existing and proven loading and ejection systems, appeared to incur the lowest weight penalty from the launching mechanism, provided the greatest reliability, and maintained an inviolate pressurized cargo bay, requiring minimum modifications for alternate missions. This configuration would allow manufacturing to be simplified, as only one airframe would be needed. In the final configuration, four CALCMs are placed forward of bulkheads supporting the wing, two missiles are placed directly behind the strut support bulkheads, between the main landing gear stowage, and the final four missiles are placed aft of the main landing gear. As shown on Figure 6-1, a centerline keel would support the floor of the cargo bay and provide a land for the weapons bay doors. Each set of missiles would have its own bomb bay doors.

6.2. Cargo Configuration

With the CALCMs being carried in their own missile bays beneath the pressurized fuselage, the cargo cabin floor and is left inviolate. The 75-foot long, 8.25-foot high, and 12-foot wide cargo bay, comparable to the size of an Airbus A320, can be utilized for cargo, passengers or systems operators without the need of removing equipment needed for the CMC role. The cargo bay is free to carry military cargo palettes, HUMVEEs, and other bulky items. As a cargo bay it would be accessible through a loading door in the side of the fuselage. This would allow currently available forklifts to move the payload into the cargo bay. By keeping the cargo bay free, the aircraft has the capability to be a true multi-role aircraft in which little needs to be done in order to transition it between roles.

6.3. Other Roles

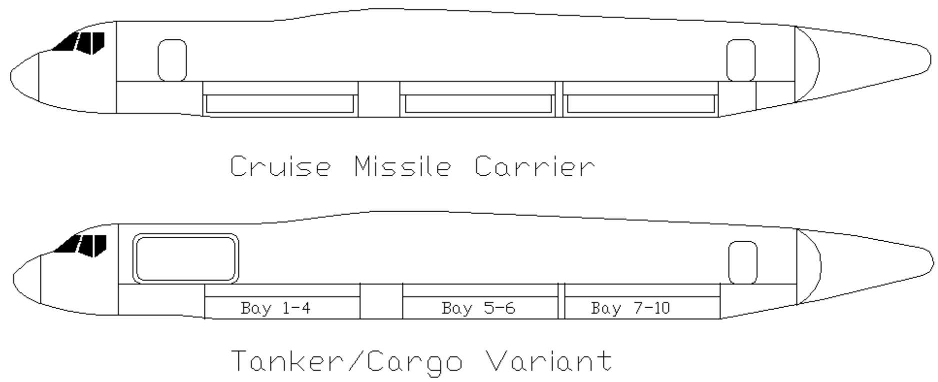
In addition to performing the CMC mission, the design aircraft is suitable for fulfilling numerous other roles. These roles are listed below in Table 6-3.

Table 6-3 - Multirole Capabilities

· Military Tanker
· Military cargo transport
· Commercial cargo transport
· Military personnel transport
· EW/ECM
· Medical evacuation transport
· Airborne command center
· Anti-submarine warfare

Of the roles listed in Table 6-3, most will simply use the large cargo bay to store bulky items. Military variants of the aircraft can be equipped with EW/ECM equipment (designed for the CMC) so that the aircraft will be able to fulfill the EW/ECM role regardless of it's normal role. The military tanker role would require the addition of a tail mounted refueling boom (for USAF), or three refueling drogues (for USN, NATO) attached at the tail, and on the strut pylons on the wings. Fuel tanks could be placed in the cargo bay or replace the missile bay set-up (for a dedicated tanker). The widely used KC-135 tanker aircraft are four decades old and will need replacing in the time frame of the CMC. The military personnel transport would require the addition of standard linear-sited troop benches or palletted seat trays, inserted via the cargo door, in the cargo bay. The Vulture could very well replace and supplement AWACS and JSTARS aircraft, which approach their third decade of operation. While performing the airborne command center role, the pressurized cargo bay could house computers and other electronics and communications equipment necessary for a command center. Additional power sources such as APUs or batteries can be located in the now unused pannier weapons bay. The aircraft's multi-role design allows for flexibility and adaptability, which is ever-important today's dynamic market.

Figure 6-3 shows an inboard profile of the CMC and cargo versions of the Vulture. The tanker/cargo variant would require a strengthen floor, a side loading cargo door, and a cargo restraint bulkhead.



- Strengthened floor
- Side loading cargo door
- Cargo restraint bulkhead

Figure 6-3 - Inboard Profile of the Vulture

7. Propulsion and Propulsion System Integration

7.1. Introduction

This report aims to fulfill two main requirements; to provide the performance specialist with the engine thrust and fuel flows throughout the flight envelope at the appropriate ratings, and to assess the installation effects and recommend the configuration that would reduce the installation penalties to a minimum.

As discussed in an above section, it was decided with the aid of the Virginia Tech MDO code that inboard wing mounted engines offer the best engine location for our aircraft.

7.2. Engine Selection

Based on the MDO code, it was determined that the two engines are required to produce a combined maximum take-off sea-level static thrust of 50,426 lbs. to power our aircraft. This implies that each engine must provide 25,213 lbs. of thrust.

There are two different approaches to engine design and selection. A specific engine for our aircraft needs could be designed, tested and built based on generic engine test data curves, this is called a ‘rubberized engine’; or a powerplant that is currently available to the aircraft manufacturer could be chosen. The second option results in a cheaper engine purchase price, and is, therefore, the procedure adopted for our design, as one of the main RFP requirements is cost effectiveness.

To enable our aircraft to be fitted with an off the shelf engine, there is a need to consider the choice of powerplants available on the market.

Table 7-1 below shows a list of jet engines producing a take-off thrust in the range of 25,000 – 30,000 lbs.

Table 7-1 - Engine Characteristics

Manufacturer	Engine	T-O Thrust (lbs)	Bypass Ratio
CFM International	CFM56-5A3	26500	6
CFM International	CFM56-7B	26400	5.1
CFM International	CFM56-5A1	25000	6
General Electric	F110-100	28000	0.87
General Electric	F110-400	26080	0.87
International Aero Engines	V2527-A5	26500	4.75
International Aero Engines	V2528-D5	28000	4.8
International Aero Engines	V2500-A1	25000	5.4
International Aero Engines	V2525-D5	25000	4.8
KKBM (Russian Federation)	NK-86	28660	1.6
Pratt and Whitney	F100-220P	27000	0.36
Pratt and Whitney	F100-229	29100	0.36
Saturn (Russian Federation)	AL-31FM	27560	0.57

It is clear that not all the engines listed above would be suitable for installing on our aircraft. The CFM56-A1, V2500-A1 and V2525-D5 are unsuitable for our application as they produce less thrust than required to power our aircraft. The two Pratt and Whitney engines, the AL-31FM, the NK-86 and the two General Electric engines listed have very low bypass ratios which implies they are military engines and therefore unsuitable for our aircraft. It should also be noted that the CFM56-7B, V2527-A5,V2528-D5 and V2525-D5 have relatively low bypass ratios , this is because they are essentially older engines based on large cores. This analysis reduces the above table to the CFM56-5A3. This engine produces more thrust than is required; in order to produce exactly 25213 lbs of sea level static thrust it could be scaled down to the appropriate value. However, it is more cost effective to choose an engine that is currently available to the aircraft manufacturer, as discussed above.

It was decided that the CFM56-5A3 was the best off the shelf engine for our design. The main characteristics of the chosen powerplant are show below in Table 7-2.

Table 7-2 - CFM56-5A3 Characteristics

CFM56-5A3			
Performance		**Dimensions**	
Take-Off Rating (lbs)	26500	Length (in)	95.4
SFC (lb/hr/lb)	0.33	Fan diameter (in)	68.3
Bypass Ratio	6	Basic dry weight (lbs)	4995
Airflow (lbs/sec)	876		
35000ft, 0.8 M ISA		**Arrangement**	1F+3A, 9A
Max Climb Thrust (lbs)	5260		
OPR at Max Climb	31.3		
Max Cruise Thrust	5000		

The characteristics of the CFM56-5A3 detailed in the table above in Table 7-2 are in accordance with what was expected for our given thrust obtained from trend plots (Reference 7.1).

Table 7-3 - Engine Performannce Assumptions

- Data is uninstalled with zero bleed and zero power extraction considered. Installation losses will be accounted for in a later section.
- The thrusts and fuel flows were given as reference values and so they were converted to net values.

At Take-Off Conditions

- The engine was operated at maximum take-off power setting.
- The engine was run at different ambient conditions to replicate ISA altitude ranges from 0 to 5000ft.
- The thrust was scaled up from a value of 25000 lbs sea-level static to 26500 lbs.
- The fuel flow was scaled to meet the SFC value of the CFM-56 5A3 at maximum power sea-level static, 0.33

At Cruise Conditions

- The engine was operated at maximum cruise power setting. This test data was not directly available and was taken to be 77% of maximum power setting.
- The engine was run at different ambient conditions to replicate ISA altitude ranges from 0 to 45000 ft.
- The thrust was scaled to meet the value of the CFM56-5A3, 5000 lbs (35000 ft, 0.8 M ISA). Thus the maximum cruise power setting was found to be 89% of maximum power setting.
- The fuel flow was scaled to meet the SFC value of the CFM56-5A3 at maximum cruise setting, 0.59
- Engine data was not directly available for the Mach 0.85 condition. The net thrust values were taken to be the same as for the Mach 0.8 condition, which may introduce a small error. Values for fuel flow were extrapolated from the engine performance curves.

At Climb Conditions

- The engine was operated at maximum climb power setting. This test data was not directly available and was taken to be 83% of maximum power setting.
- The engine was run at different ambient conditions to replicate ISA altitude ranges from 0 to 45000 ft.
- The thrust was scaled to meet the value of the CFM56-5A3, 5260 lbs (35000 ft, 0.8 M ISA). Thus the maximum climb power setting was found to be 93% of maximum power setting.
- The same process as for the maximum cruise power setting was applied for the Mach 0.85 condition.

At Flight Idle Conditions

- The engine was operated at flight idle power setting.
- The engine was run at different ambient conditions to replicate ISA altitude ranges from 0 to 40000 ft.

The CFM56-A3 is a cost –effective, advanced derivative engine of the CFM56-2 and CFM56-3 families. It is characterized by enhanced efficiency of all components and an improved thermodynamic cycle. The specific fuel consumption is 10-11% lower compared to its predecessors. The CFM56-5A is equipped with a three-dimensional aerodynamic fan design and a Full Authority Digital Electronic Control (FADEC) that efficiently unifies the engine and aircraft systems.

The CFM56-5A and –5B are the preferred engines for the Airbus Industry A320 family, they power 60% of all A319/A320/A321 aircraft ordered.

7.3. Engine Performance

Engines perform differently at different altitudes, Mach numbers, throttle settings, and operating conditions. The performance of the CFM-56 5A3 was thoroughly investigated to ascertain that it would be able to meet the aircraft propulsion requirements at every stage in the flight envelope. However, not enough data for the thorough performance analysis needed was available. To be able to have access to the amount of engine test data needed a compromise was reached. The AIAA generic engine data gives details of thrusts, airflows, fuel flows, turbine entry temperatures and dimensions for various flight conditions. This engine produces 25,000 lbs of maximum take-off sea-level static thrust. It was decided to scale the AIAA engine to meet the CFM-56 characteristics, taking account of the better specific fuel consumption of the CFM engine. The assumptions made in this process are stated in Table 7-3.

The maximum continuous rating data was not directly available and was suggested to be the same as the maximum climb rating. In theory the maximum continuous rating could be used continuously, in practice because of the requirement for long engine lives, it is only used in emergency situations. The maximum climb rating is set by the engine manufacturer in order to attain the engine life required for economic operation.

The loiter condition is covered by 0.4 maximum cruise thrust.

The derived performance data described above is shown in Figure 7-1 and Figure 7-2.

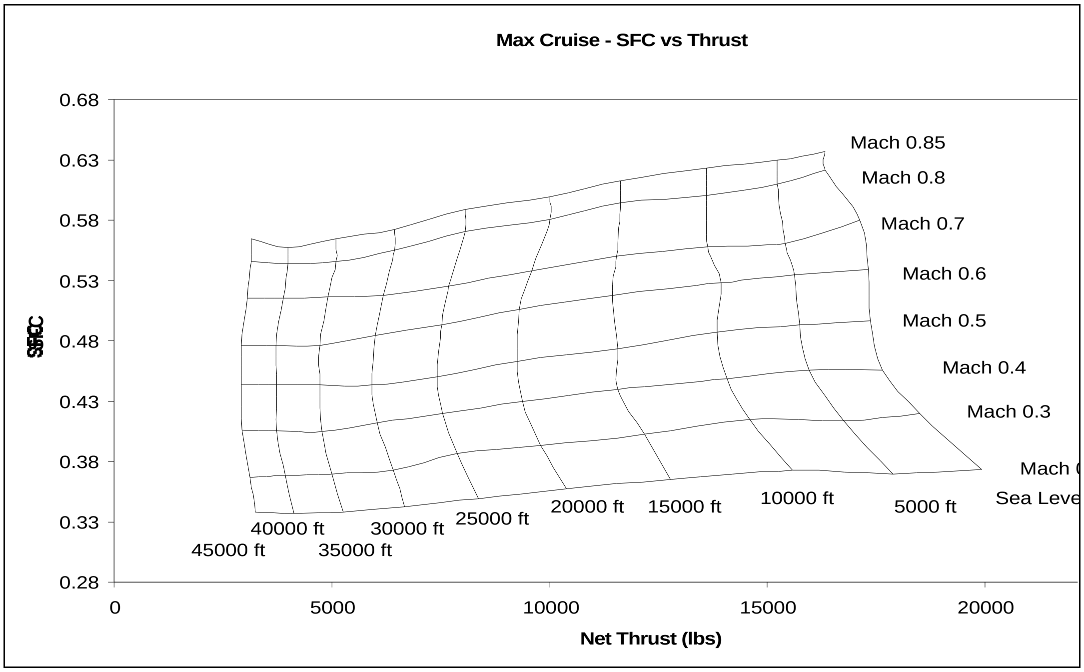


Figure 7-1 - Engine Performance at Maximum Cruise Condition (SFC vs Thrust)

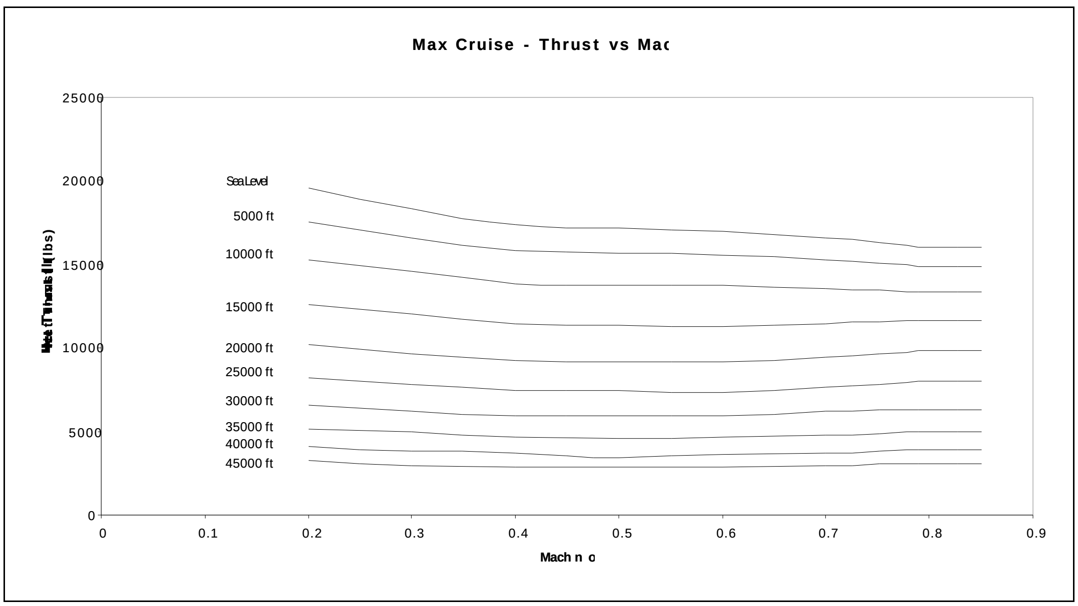


Figure 7-2 - Engine Performance at Maximum Cruise Condition (Thrust vs Mach Number)

7.4. Engine Installation

In the preceding sections only uninstalled engine thrust has been considered. Placing the engine within an airframe inevitably induces forces on the external services that increase the total drag; these must be overcome by the available thrust of the engine.

The object of the engine installation is to take the chosen powerplant enclose it in a nacelle and install it on the aircraft for the lowest possible weight, drag and cost.

7.4.1. Nacelle Design

There are many issues involved in designing a nacelle for an engine. These are presented in Table 7-4.

Table 7-4 - Nacelle Design Considerations

· Air must be delivered at conditions acceptable to the engine compressor system throughout the aircraft flight envelope. · The nacelle must efficiently separate the air, which passes through the engine from that which passes outside. · The external nacelle flow must be such as to minimise the external drag. · There must be an acceptable fairing down to the final exhaust nozzle. · Sufficient space must be provided for the engine mechanical systems and accessories while at the same time minimising the impact on the nacelle profile. · Provision is required for a thrust reverser and its associated systems. · Also, noise levels, maintenance accessibility and safety considerations are required.
(Reference 7.2)

The nacelles for the CFM56-5A3 engines are currently manufactured by Rohr Industries. It is a reasonable decision to use these off-the-shelf nacelles for our design. This will increase the cost effectiveness of our aircraft, and also simplify the design process as there is currently a satisfactory nacelle on the market and therefore, no longer a need for a specific nacelle to be designed, tested, and built.

7.4.2. Intake Design

The main requirement of an air intake is that, under all operating conditions, delivery of the air to the engine is achieved with the minimum loss of energy occurring through the duct.

The CFM56-5A3 engine nacelle, as most nacelles installed on subsonic aircraft, has a pitot-type circular intake.

A feature to be noted on the CFM nacelle design is the vortex generator fins, which improve the leading edge slat efficiency near the pylon.

7.4.3. Secondary Airflow and Cruise Bleed

In a jet propulsion system the engine takes the amount of air it wants, not what the inlet wants to give. The inlet capture area must be sized to provide sufficient air to the engine at all aircraft speeds. It must also provide “secondary air” for cooling and environmental control.

The required engine mass flow is provided by the engine manufacturer, and is a function of Mach number, altitude, and throttle setting. Usually the manufacturer's data should be increased by 3% to allow for manufacturing tolerances.

The secondary airflow requirements are accurately determined by an evaluation of the aircraft's subsystems. Raymer suggests an initial estimation of secondary airflow as a fraction of engine mass flow as shown in Table 7-5.

Table 7-5 - Secondary Airflow (Typical)

	Ms/Me
Engine	
Nacelle cooling	0-0.04
Oil cooling	0-0.01
Ejector nozzle air	0.04-0.20
Hydraulic system cooling	0-0.01
Environmental control system	
Cooling air (if taken from inlet)	0.02-0.05
Typical totals	
Fighter	0.2
Transport	0.03
(Reference 7.3)	

During cruise the engines will be required to provide the necessary extra services for the rest of the aircraft. Usually the cabin pressurization system takes a bleed from the compressor to power a turbine that draws in atmospheric air, which in turn cools the avionics equipment and pressurizes the fuel. The hydraulic system is pressurized by using a pump that is driven by the engine. Also, a small proportion of the compressor bleed is used to prevent ice formation.

Neither of these off-takes was considered in the engine performance section above. These factors will usually increase the specific fuel consumption by 4%. However, as discussed in the systems section, our aircraft is mainly electrical which will improve the sfc.

7.4.4. Engine Mounting

The engine needs to be mounted on the aircraft in such a way that allows the thrust forces to be transmitted to the aircraft main structure to allow the aircraft to be propelled, in addition to supporting the engine weight and carrying any flight loads. Figure 7-3 illustrates a typical engine-mounting technique.

On our aircraft the engines are to be mounted on underwing pylons, this helps to reduce the wing weight because the weight of the engines out along the wing provides a ‘span-loading’ effect. Also, flaps can direct the jet exhaust downward, which greatly increases lift for short take-off. However, this configuration can disturb the airflow on the wing and strut, increasing drag and reducing lift. This was one main concern, as well as the onset of shockwaves.

The position of the strut restricted the exact placement of the engine pod. It was decided the engine would have to be mounted far forward of the engine leading edge and with a very small gulley, to conform to the wing strut placement. The pylon dimensions were based on the A320 arrangement. Dimensional details can be seen on the drawing.

The optimum spanwise location of the engines from the fuselage centerline was found to be 0.32 of the wing semi span using the Virginia Tech MDO code.

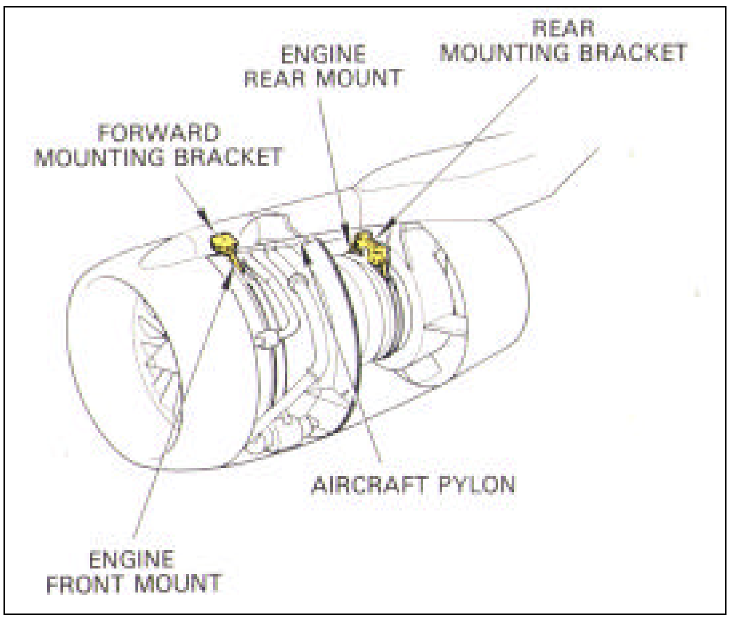


Figure 7-3 - Typical Engine Mounting (Reference 7.4)

The engine mounting beams connect the mounting points of the engine with the aircraft pylon. These beams are connected to the aircraft pylons via connecting links and thrust struts. The weight of the engine is supported via the connecting links, and the thrust of the engine is transmitted to the aircraft structure via the thrust strut.

7.5. Conclusions and Recommendations

The propulsion system has now been specified and designed sufficiently to progress onto the next stage of the design process.

The engine choice is a currently available off the shelf option and not a new design. This reduces the initial purchase cost for the aircraft manufacturer.

Using a combination of the available data for our engine (CFM56-5A3) and the AIAA generic engine performance curves it has been possible to predict the engine's performance through all stages of the flight envelope.

The engine installation and mounting has been considered. The chosen powerplant is offered as a total propulsion system, which significantly reduced the time and cost of the installation process.

8. Weights and Balances

In addition to the weights generated by the MDO code, the component weights of the Vulture were estimated using the procedures outlined Torenbeek (Reference 8.1) and the Class II Method in Roskam (Reference 8.2). In most cases, two (or more methods) were used to calculate the component weights and then averaged for the final weight approximation. When possible, systems weights were estimated using comparator aircraft systems found in *Jane's All the World's Aircraft* (Reference 8.3). After the component weights were tabulated and their individual centers of gravity estimated, Equations 8.1, 8.2, and 8.3 from Roskam were used to compute the overall center of gravity for the Vulture's design mission.

$$x_{CG} = \sum_{i=1}^{n}\left(W_i \ x_i\right) \Big/ \sum_{i=1}^{n} W_i \tag{8.1}$$

$$y_{CG} = \sum_{i=1}^{n}\left(W_i \ y_i\right) \Big/ \sum_{i=1}^{n} W_i \tag{8.2}$$

$$z_{CG} = \sum_{i=1}^{n}\left(W_i \ z_i\right) \Big/ \sum_{i=1}^{n} W_i \tag{8.3}$$

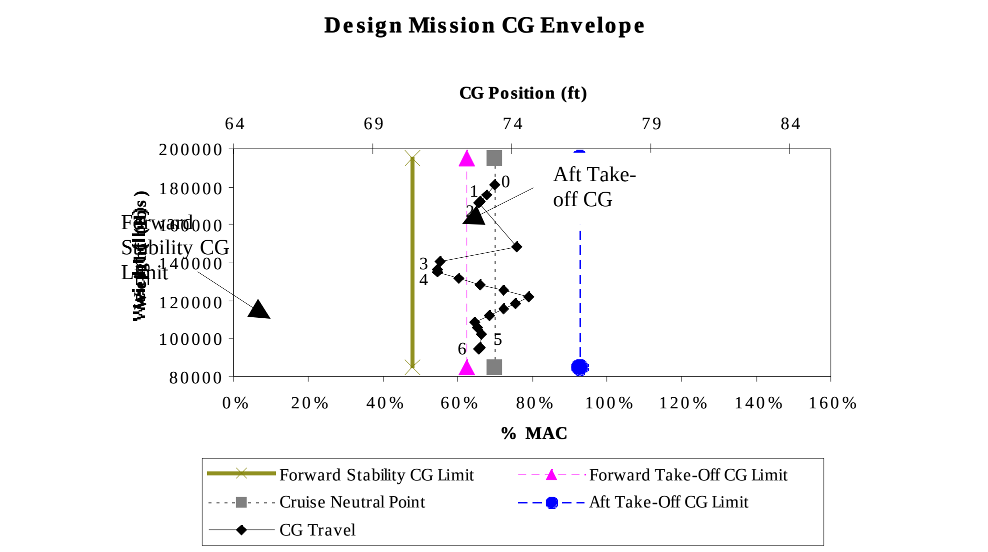
Table 8-1 - Component Weight Table

Component	Weight	x	W*x	z	W*z
wings	17288	68.7	1188042	13.5	233390
struts	908	68.3	62003	6.5	5902
horizontal tail	1631	137.6	224323	33.0	53814
vertical tail	1581	126.1	199326	10.7	16952
control surfaces	3080	63.7	196258	15.0	46200
fuselage	26229	65.0	1704873	6.8	177045
nose landing gear	1754	20.0	35080	1.5	2631
aft landing gear	5516	76.0	419216	1.8	9653
Structure Subtotal	**57986**	**69.5**	**4029120**	**9.4**	**545586**
CFM total engine system	15000	59.6	894300	8.5	127500
inflight refueling	78	4.8	371	9.0	702
fuel dumping system	57	130.0	7410	12.5	713
Engine Subsystem	**15135**	**59.6**	**902081**	**8.5**	**128915**
cockpit	2186	10.0	21857	9.0	19672
APU	2200	128.0	281600	5.0	11000
electrical system	1980	65.0	128700	8.0	15840
electrical power systems	2200	125.0	275000	6.0	13200
a/c generation	3300	125.0	412500	6.0	19800
looms & cabling	528	65.0	34320	8.0	4224
radar	132	3.0	395	2.0	263
RWR	112	120.0	13464	10.0	1122
ELINT	169	125.0	21175	18.0	3049
EW/ECM equipment	5000	125.0	625000	9.0	45000
countermeasures	389	128.0	49843	15.0	5841
chaff countermeasures	880	128.0	112640	18.0	15840
Fixed Equipment Subtotal	**19076**	**96.1**	**1833494**	**8.1**	**154851**
crew	360	15.0	5400	8.5	3060
missiles 1-4	3250	40.0	130000	1.3	4063
missiles 5-6	3250	68.5	222625	1.3	4063
missile 7-10	3250	90.5	294125	1.3	4063
Mission Payload	**32860**	**65.3**	**2147150**	**1.3**	**43685**
tail fuel tanks	8026	122.0	978931	10.7	86079
inboard wing fuel tanks	22785	63.5	1447092	13.5	307601
mid-wing fuel tanks	15408	73.6	1133418	13.5	208009
outboard wing fuel tanks	9924	83.3	826740	13.5	133969
Fuel	**56143**	**78.1**	**4386181**	**13.1**	**735658**
Empty Weight	**92198**	**72.6**	**6696500**	**9.0**	**829352**
Take-off Gross Weight	**181201**	**73.0**	**13229831**	**8.9**	**1608695**

The component weights and their centers of gravity are found in Table 8-1, and the operational centers of gravity are found in Table 8-2. The CG envelope was calculated for the Vulture's design mission. This is shown in Figure 8-1.

Table 8-2 – Operational Weights and Centers of Gravity

	Weight (lbs)	x-CG (ft)	y-CG (ft)	z-CG (ft)
Maximum TOGW	181,200	73.0	0.0	8.9
Take-off, Zero Stores	148,700	74.6	0.0	10.6
Landing, Zero Stores	94,600	72.4	0.0	9.1
Landing, Returned Stores	127,100	70.7	0.0	7.1
Operating Empty Weight	125,100	70.7	0.0	7.0
Empty	92,200	72.6	0.0	9.0



0-1 Warm-up and take-off.
1-2 Climb.
2-3 4000 NM cruise.
3-4 30-minute loiter.
4-5 CALCM launch.
5-6 1000 NM cruise.
6 Descend and land.

Figure 8-1 - Design Mission CG Envelope

After calculating the Vulture's center of gravity, the moments of inertia were calculated using Equations 8.4, 8.5, and 8.6, again from Roskam.

$$I_{xx} = \sum_{i=1}^{n} \left[m_i \left((y_i - y_{CG})^2 + (z_i - z_{CG})^2 \right) \right] \tag{8.4}$$

$$I_{yy} = \sum_{i=1}^{n} \left[m_i \left(\left(z_i - z_{CG}\right)^2 + \left(x_i - x_{CG}\right)^2 \right) \right] \tag{8.5}$$

$$I_{zz} = \sum_{i=1}^{n} \left[m_i \left(\left(x_i - x_{CG}\right)^2 + \left(y_i - y_{CG}\right)^2 \right) \right] \tag{8.6}$$

The calculated moments of inertia are presented in Table 8-3.

Table 8-3 - Moments of Inertias

	I_{xx} (slug-ft^2)	I_{yy} (slug-ft^2)	I_{zz} (slug-ft^2)
Take-off	3,013,800	3,522,600	6,221,100
Cruise	1,993,600	3,433,200	5,162,200
Landing	1,325,400	2,870,700	3,375,300
Empty	730,200	2,730,000	3,326,000

The moments of inertia play an important role in determining how to control the Vulture.

9. Aerodynamics

9.1. Wing Planform Selection

The principal criteria for wing planform design can be summarized into four main areas, shown in ??.

1. The wing must enable the aircraft to meet the performance requirements cited in the specification, whilst maintaining maximum operational flexibility and economic performance.
2. The wing must provide acceptable flight characteristics at all flight speeds, altitudes, and configurations.
3. The wing must be structurally feasible, in terms of manufacture, weight, rigidity, strength and cost.
4. Sufficient space for storage of fuel and systems.

(Reference 9.1)

Clearly these design criteria are not solely dependent on aerodynamic considerations, and compromise between control, stability, and structural requirements proved necessary in planform design process.

The principal performance requirement to be considered in this case was the requirement for long-range cruise at Mach 0.85. This requirement was considered the prime consideration; therefore an efficient planform for transonic cruise was produced prior to consideration of the high-lift configuration

The problems of aerodynamic design for this aircraft would hence entail achieving a high critical mach number for efficiency of cruise at high speed, whilst avoiding undesirable characteristics at low speed and off-design flight conditions.

Initial sizing and performance analysis indicated a wing area of 1790 ft^2. This value was further refined by analysis of the altitude requirements imposed on Vulture in the cruise portion of the design mission, to ensure compliance with minimum/maximum altitude constraints.

The selection of wing aspect ratio has direct implications upon aircraft performance. Induced drag is inversely proportional to aspect ratio, as detailed in Reference 9.2; hence aerodynamically speaking it was desirable to maximize the aspect ratio of the wing in order to minimize induced drag.

Structural factors limit the maximum achievable aspect ratio hence the final selection of aspect ratio cannot be decided purely on the basis of aerodynamic efficiency, but results from a compromise between conflicting aerodynamic and structural requirements. For the wing area and mission requirements given for this design, the airport gate-width requirement would not impose a constraint upon the aspect ratio.

For conventional cantilever aircraft, the range of acceptable cantilever ratios structurally limits maximum achievable aspect ratio. Analysis of comparator aircraft, listed in Table 9-1, indicated a range of wing aspect ratios of 8.5 to 10.1

Table 9-1 - Comparator Aircraft Aspect Ratios

Aircraft	Aspect Ratio
Airbus A320	9.5
B52	8.6
Boeing 767	9.3
Boeing 777	8.7
Airbus A340	10.1
Vulture	***12.3***

The strut-braced wing configuration of Vulture provided a considerable advantage over such designs. As explained in Reference 9.3, the inclusion of the strut to the wing structure creates bending moment discontinuity along the span. This allows increased aspect ratio without wing structural penalty.

Taking this load alleviation into account, it was estimated initially that the maximum allowable aspect ratio would be approximately 12.5. This was considered to be the maximum aspect ratio achievable with strut bracing, without experiencing structural penalties upon the wing, and provided a considerable increase in aspect ratio over comparator aircraft

Analysis with the multidisciplinary design optimization (MDO) code based at Virginia Tech closely confirmed this estimation, also indicating an aspect ratio of 12.5. This value was adopted as the basis for wing planform sizing and implied a wingspan of 149.6 ft

The selection of appropriate combination of sweep and thickness ratio for the wing was of utmost importance in determining the high-speed cruise performance of the aircraft. The main aim when choosing these geometric criteria was to increase the critical Mach number for the wing to a value where wave drag was reduced – resulting in efficient cruise. It was hence clear from the requirement to cruise at Mach 0.85 that sweep would be required

The advantages of strut braced configurations include the capability to reduce wing thickness/chord ratio without weight penalty, which in turn permits some reduction of sweep of the wing without drag penalty (Reference 9.4). It was clear that the aerodynamic design of the wing should exploit this possibility, which would permit weight saving and hence cost saving over conventional, cantilever wing designs.

In order to evaluate the effectiveness of different combinations of sweep and thickness to chord ratio, an analysis of cruise wave drag contributions for Vulture and comparator aircraft was carried out using the Korn Equation modified to account for wing sweep (References 9.2, 9.3, 9.4).

$$M_{dd} = \frac{K_a}{\cos\Lambda} - \frac{t/c}{\cos^2\Lambda} - \frac{C_l}{\cos^3\Lambda} \tag{9.1}$$

$$M_{crit} = M_{dd} - \left(\frac{0.1}{80}\right)^{1/3} \tag{9.2}$$

This allowed divergence drag mach number to be calculated as a function of wing sweep , section lift coefficient, Aerofoil technology factor, and thickness-to-chord ratio. From this value, the critical mach number and hence the wave drag coefficient was found;

$$C_{d_{wave}} = 20(M - M_{crit})^4 . \frac{S_{strip}}{S_{ref}} \tag{9.3}$$

Aerofoil technology factor was taken as 0.95 for a supercritical wing section, and section C_l was estimated at 0.5 for purposes of analysis

The results of this analysis allowed a section quarter-chord sweep of 25.5° and mean thickness-to-chord ratio of 9.5% to be selected. This design point produced a wave drag coefficient of 0.0006 at mach 0.85, which proved lower than that of the Boeing 777 and Airbus A340 at their respective cruise velocities, whilst fully embracing the benefits of reducing sweep. This process was also used to determine the sweep and thickness-to-chord ratios of the tail and strut surfaces. For all surfaces, the results from this analysis concurred with those from the MDO code.

Wing taper was desirable to achieve near-elliptical spanwise lift distribution hence minimizing induced drag. When deciding taper ratio for Vulture, constraints based on structural height required for ailerons were taken into account. A taper ratio of 0.3 was selected as an initial design taper, based on analysis of comparator aircraft, and information in Reference 9.6. Vulture possesses straight taper from root to tip, whereas cantilever wings generally possess compound taper, with reduced inboard trailing-edge sweep. This was not an aerodynamic decision, but was done in order to allow the MDO code to be used for aerodynamic evaluation, since it was based upon a straight tapered planform. (A compound taper planform would be preferable for flap performance.)

In combination with the selected taper ratio, a design linear wing twist of 3° was added in order to provide reasonable tip stall characteristics. The incidence of the wing should facilitate straight-and-level cabin angle during cruise.

Without experimental data, it is difficult to predict whether these parameters provide the optimum lift distribution and stall characteristics for Vulture. However such further analysis and testing is outside the scope of this treatment.

9.2. Aerofoil Selection

The prime criteria for airfoil selection were as follows in Table 9-2:

Table 9-2 - Airfoil Selection Criteria

· Section Cl of around 0.7, as is suitable for advanced-technology transonic transport aircraft (Reference 9.7)
· Thickness-to-chord ratio of 9.5% as determined by wave-drag analysis.
· Optimum drag characteristics for design cruise at Mach 0.85.

These criteria led to the selection of section NASA SC(2)- 0710, as shown in Figure 9-1.

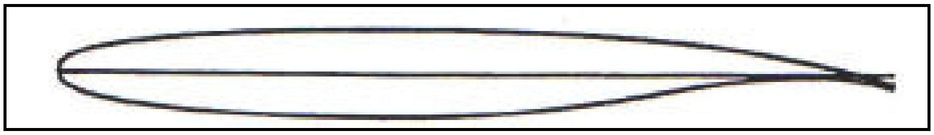


Figure 9-1 - NASA SC(2)-0710 Supercritical Airfoil

This aerofoil is a supercritical section. Supercritical airfoils delay the onset of the divergence-drag mach number by up to 15% compared to similar classical airfoils, as detailed in Reference 9.8. Test data from Harris (1990) indicates a drag bucket for a freestream mach range of 0.74-0.78. Accounting for sweep effects, this corresponds directly to the required cruise regime of Vulture. At mach 0.85 cruise, a section C_d of 0.008 was obtained. The pressure distribution of the SC(2)-0710 section is illustrated in Figure 9-2.

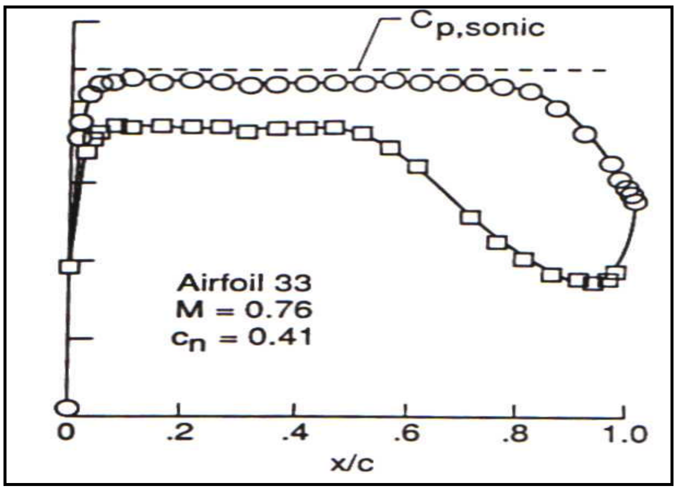
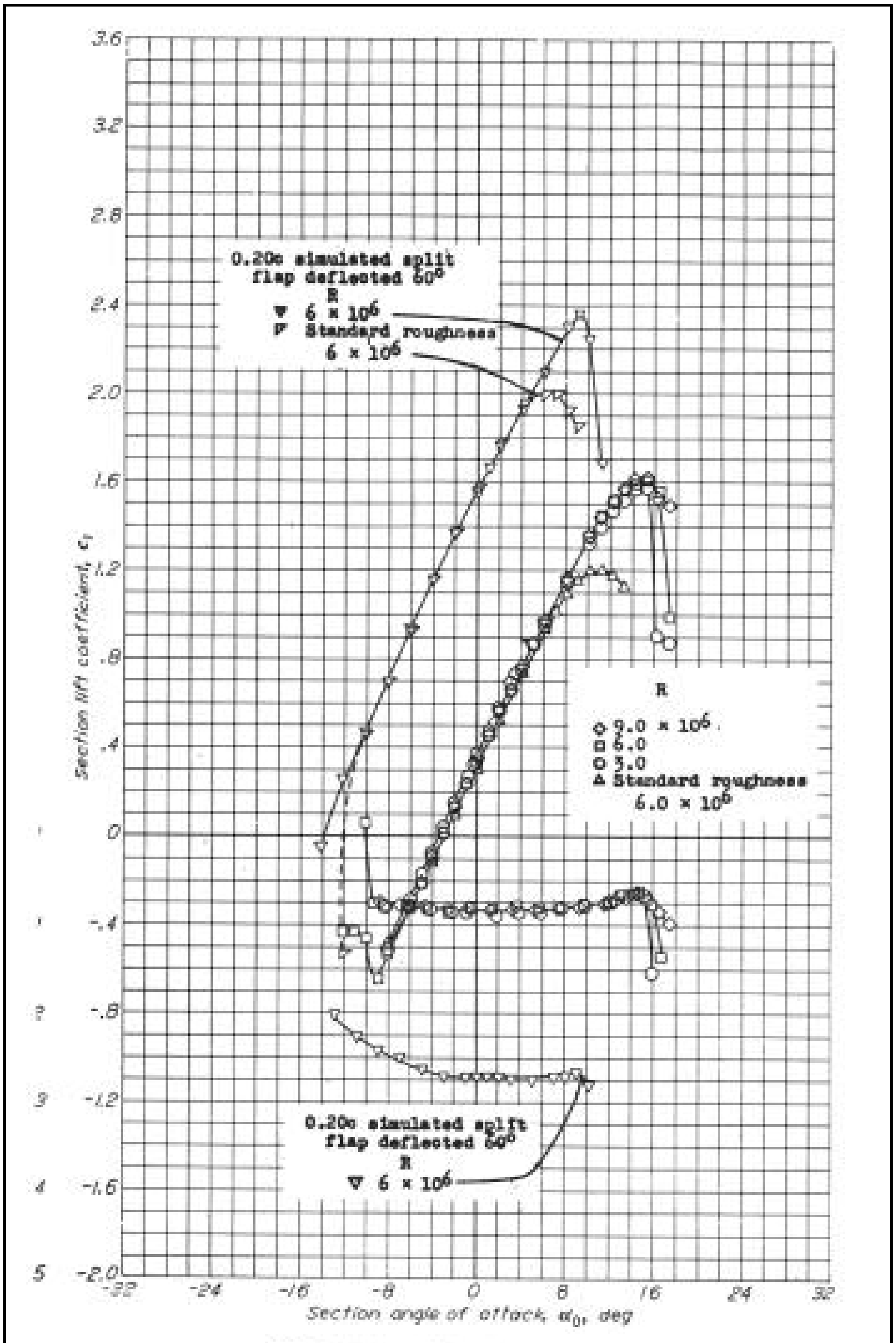


Figure 9-2 - SC(2)-0710 Pressure distribution

Information on the lift-curve slope, $dC_l/d\alpha$ and maximum section lift coefficient, C_{lmax}, was not available for this aerofoil. Because of this, information on these parameters was taken from the geometrically similar NACA 64A410, for a Reynolds number of 6×10^6 shown in Figure 9-3 and the C_{lmax} modified by a factor of 30% to account for the increased performance of the supercritical section, as suggested by Reference 9.8.



NACA 64A410 Wing Section

Figure 9-3 - Lift Curve Slope of NACA 64A410 Section

The section characteristics for the selected aerofoil are summarized in Table 9-3.

Table 9-3 - SC(2)-0710 Section Characteristics

C_l	0.7
C_d	0.008
C_m	-0.15
Cl_{max}	1.625

The section selected does not precisely meet the requirement for the wing section. In particular, it possesses a t/c of 10%, which is higher than the 9.5% design requirement. However, this does not preclude the use of this section as a design starting point for Vulture. Optimization packages, such as the BVGK package that utilizes the 'CODAS' multivariate optimization process, can be applied to supercritical airfoil to minimize C_d at a given Mach number by varying several parameters for the aerofoil. ESDU data item 99019 summarizes the optimization process, whilst ESDU data items 99020 and 99021 show cases where the process has been applied to the RAE 2822 supercritical section (References 9.9, 9.10, 9.11).

9.2.1. Aircraft Lift

The wing maximum lift coefficient, C_{Lmax}, was calculated from the aerofoil C_{lmax} using the method from Reference 9.6.

This equation gave a C_{Lmax} value of 1.31, although it should be noted that maximum lift cannot be predicted with great accuracy through this method. Even with refined wind-tunnel tests, such a prediction cannot easily be made.

In order to achieve the required C_{Lmax} values for takeoff and landing both leading edge slats and double-slotted trailing edge flaps were incorporated into the wing planform. The addition of these devices gave a maximum achievable lift coefficient of 2.72 with both sets fully deployed. This margin for error was included to take account of any limitation on the calculation method used for C_{Lmax}. The flap and slat deployments for takeoff and landing, calculated according to Reference 9.6, are detailed in Table 9-4.

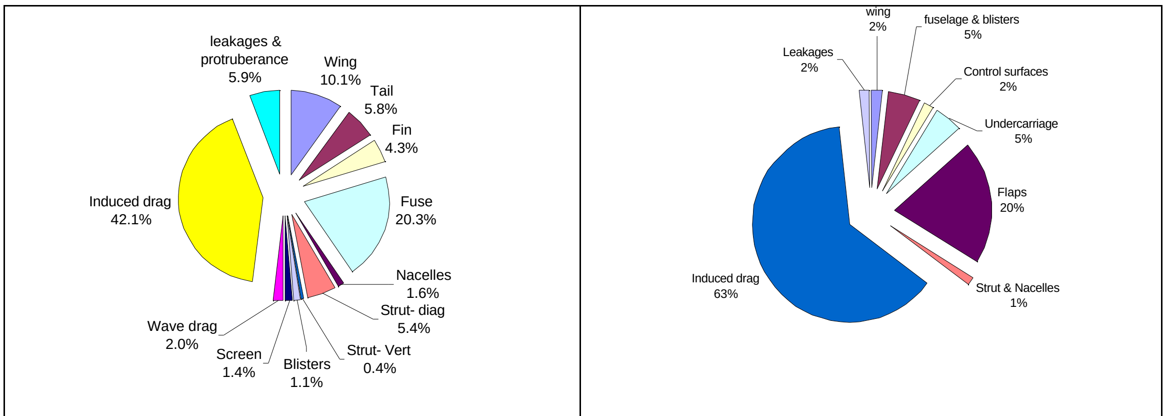
Table 9-4 - Flap/slat Configurations for Takeoff and Landing

Flight Condition	Required C_{Lmax}	Flap Deflection	Slat Deployment
Cruise	1.3	0	0
Takeoff	2.2	32	100%
Landing	2.6	54	100%

9.3. *Aircraft Drag*

The process of aircraft drag estimation was carried out using two principal methods. The Multidisciplinary Design Optimization code was used to give a breakdown of drag components, and to estimate wave drag and induced drag at various flight conditions. In order to validate the output of the MDO code, an estimation of aircraft drag based on flat-plate approximations modified to approximate the component form. This process combines techniques recommended by References 9.1, 9.6, and 9.12 and accounted for laminar, transition and turbulent flow regions on the main wing.

Results of the drag estimation process for takeoff and cruise configurations are shown in Figure 9-4, Figure 9-5, Figure 9-6.



a) cruise config. C_{D0}=0.0166,C_D=0.029& | b) takeoff config., C_{D0} = 0.0503,C_D =0.1360

Figure 9-4 - Drag Breakdown for Takeoff and Cruise configurations

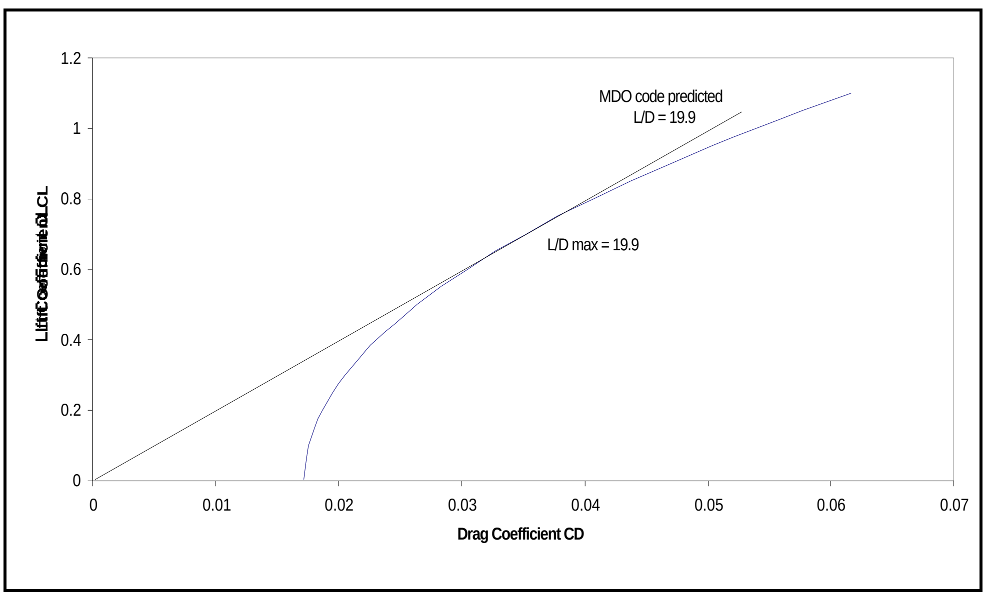


Figure 9-5 - Drag Polar for Cruise Configuration

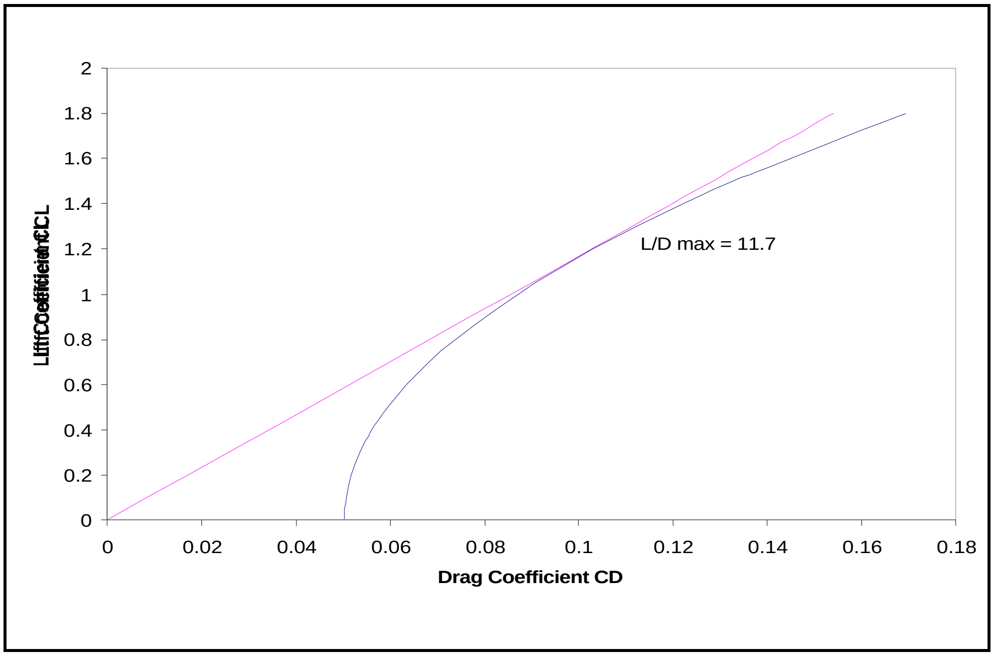


Figure 9-6 - Drag Polar for Takeoff configuration

Figure 7.5 shows that for cruise configuration, Vulture has maximum lift-to-drag ratio of 19.9. This compares favorably with current civil transport aircraft, such as the Boeing 777. In addition to this, Figure 9-5 and Figure 9-6, show that the MDO code results exactly match those produced from the Flat-plate estimation method undertaken, both providing the same cruise L/D ratio. This concurrency provides a high degree of confidence in the techniques used and answers gained from the drag estimation process. Although the MDO code assumes a circular fuselage, whereas flat-plate method accounts for the increase in fuselage drag due to bomb-bay blisters, Figure 9-5 shows that the increase in drag due to these blisters is only 1% of the overall drag of Vulture, and as such not significant enough to cause any major discrepancies.

10. Performance

10.1. Overview and Mission Statement

The calculation for the performance for this aircraft was carried out with the aid of spreadsheets and the generation of Visual Basic code. All calculations are based on the performance equations derived and outlined in References 10.1 thru 10.5. The Visual Basic codes make use of iterative techniques to give an improvement in accuracy.

The Request for Proposal gives a detailed description of the design mission (Figure 1-2) and the point performance requirements for the aircraft. These are:

Table 10-1 - Design Point Performance Requirements

1.	Cruise Speed of 0.85M
2.	An initial cruise altitude of 30000ft rising up to a maximum of 45000ft
3.	The Balanced Field Length shall be less than 8000ft
4.	The landing distance at max landing under ISA conditions shall be less than 6000ft
5.	The landing distance at end mission shall be less than 6000ft under ISA conditions and less than 6800ft under USAF tropical conditions.

All point performance requirements are met by this design.

10.2. Cruise Performance

The Request for Proposal defines several performance requirements for cruise. As time goes on the aircraft burns fuel or drops payload, becoming lighter. To maintain the optimum cruise C_L at a constant mach number the altitude must be increased with time. Figure 10-1 shows the optimum cruise altitude for a given weight when flying at 0.85M.

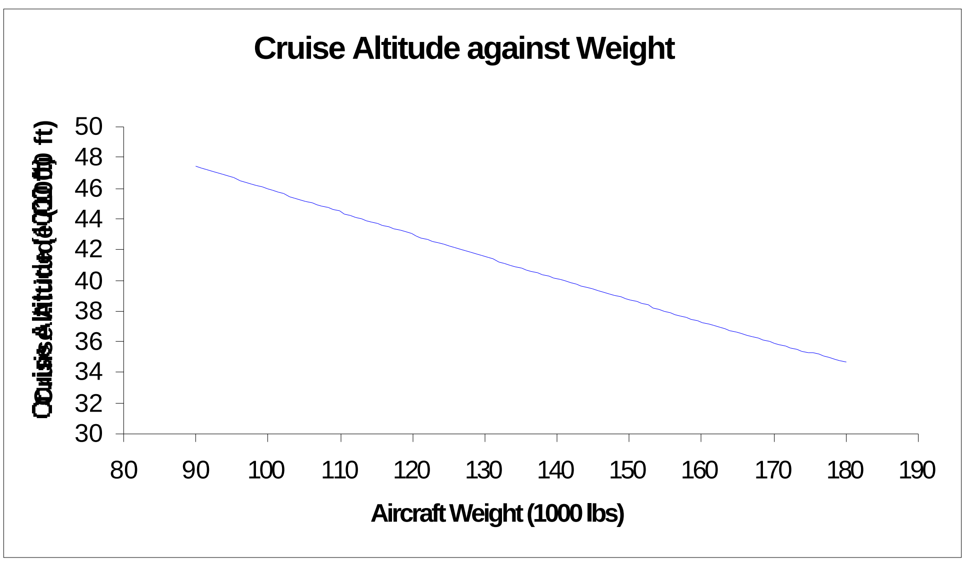


Figure 10-1 - Cruise Altitude against Weight

It is clear from Figure 10-1 that at weights below 105000lbs the optimum cruise altitude is greater than 45000ft. Capping the altitude to 45000ft means that the aircraft must cruise slightly off optimum below this weight; this will have a slightly detrimental effect on fuel efficiency.

FAA regulations state that an aircraft may not fly higher than the altitude at which it can achieve a minimum climb rate of 300ft/min. The aircraft has a significant surplus in climb rate at any point during cruise.

The design mission calls for a range of 5000nm plus the loiter and other phases of flight. The design payload of the aircraft is 40000lbs; this comprises of 10 cruise missiles and a 5000lbs allowance for protective countermeasures. It is possible that in times of high tension the aircraft will be dispatched on a mission so as to be in the required position if force becomes necessary. In this case the aircraft will be required to hold at its loiter point until authorization is given; if a peaceful resolution is achieved then the aircraft will be recalled. As the aircraft will consume more fuel on the return leg if the payload has not been dropped then it seems prudent to ensure the design mission can be flown without dropping the missiles. Range calculations have been carried out with this in mind. Figure 10-2 shows a Payload Range Diagram for the aircraft.

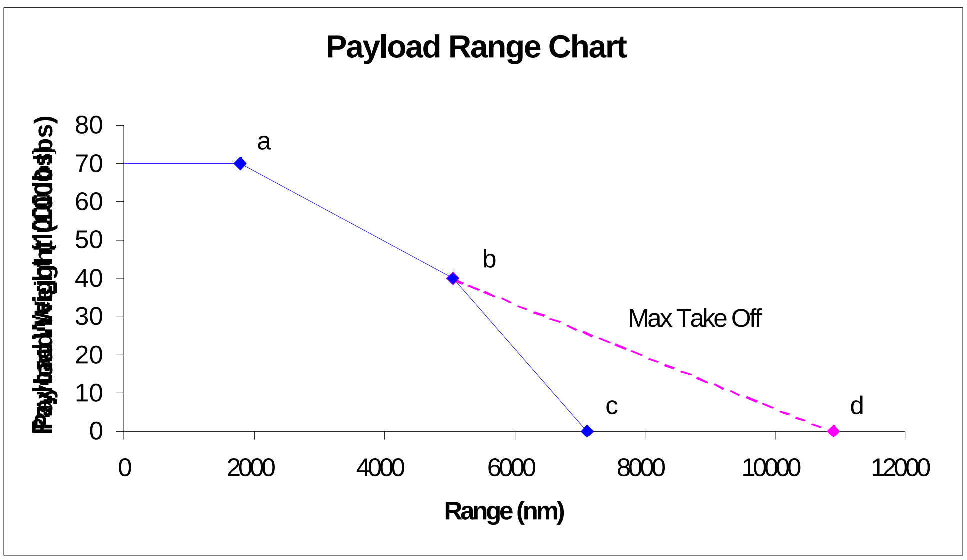


Figure 10-2 - Payload Range Diagram

The Payload Range Diagram is comprised of 4 points. Point (a) is the range for the aircraft flying with maximum payload. In this case fuel has been replaced with cargo until it reaches it max payload weight. This is most relevant with configurations such as the cargo variant where the mission is not constrained by the number of missiles that can be dropped. It is also possible that the cruise missile variant could fly with extra cruise missiles within the fuselage so that it could be re-armed at its final destination. Point (b) coincides with the design mission. In this case the aircraft is flying with the design payload and with design/maximum-standard fuel weight. As you progress towards point (c) the aircraft is flying with the design/maximum-standard fuel weight, but a decreasing payload weight. At point (c) the aircraft is flying with no payload. Moving toward point (d) involves installing extra/temporary fuel tanks within the aircraft and replacing payload with fuel so as to keep the aircraft at Max Take Off Weight until, at point (d) the entire payload has been swapped for fuel. In any military role this last option is unlikely to be implemented as the aircraft is capable of in-flight refueling.

10.3. Take Off Performance

For a two engine aircraft the balanced field length is generally the limiting factor for take off. The balanced field Length is the distance at which the field length for an aborted take off is the same as the distance for take off to 35ft after an engine failure at the critical speed. The balanced field length is a function of aircraft weight and Figure 10-3 clearly shows that at the aircraft's maximum takeoff weight the balanced field length is well within the 8000ft limit specified by the Request for Proposal. Table 10-2shows the critical speeds for the design case.

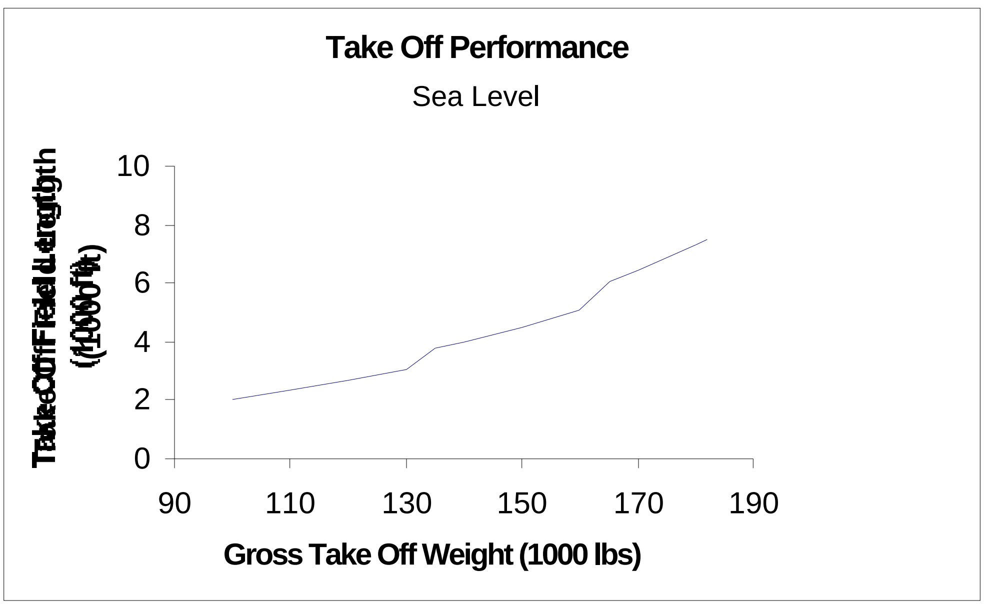


Figure 10-3 - Take Off Performance

Table 10-2 - Critical Velocities

Critical Velocities		
V1	(ms^{-1})	71.12
	(kts)	138.25
V2	(ms^{-1})	84.13
	(kts)	163.55

With one engine out, the second segment climb can become a critical phase of flight. The second segment climb is that from 35-400ft where the flaps are in the take off configuration but the undercarriage is raised. FAA airworthiness requirements say that the aircraft must be capable of a climb gradient of 2.4% at V_2. Calculations show that the aircraft is not capable of the required climb gradient with one engine in-operative. To give the necessary climb gradient the aircraft overspeeds on take-off. The flaps are retracted to give a lower C_{Lmax}, hence a greater required velocity for takeoff. The kinks in graph 8-4 show the weights at which the flaps must be retracted.

10.4. Landing Performance

The Request for Proposal states three landing performance requirements that must be met. Figure 10-4 shows the landing performance for the aircraft under both ISA conditions and USAF tropical conditions. The graphs show that the aircraft is comfortably within the limits set in the Request for Proposal at both maximum landing weight (156000lbs) and end mission weight (86000lbs).

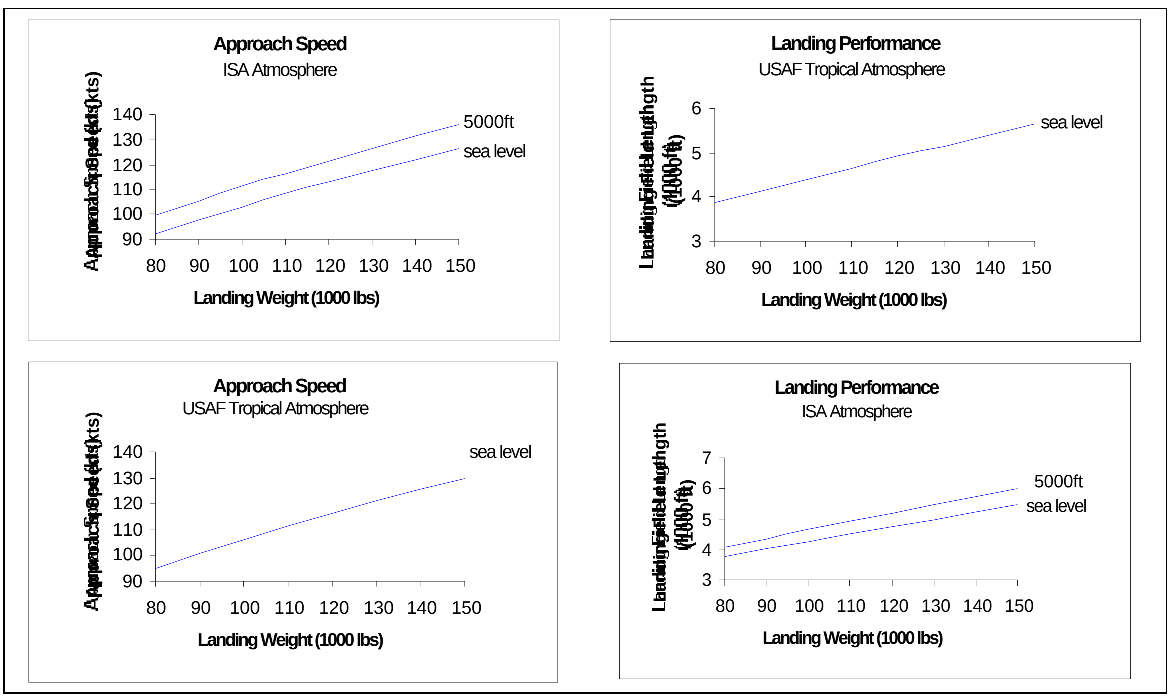


Figure 10-4 - Landing Performance for ISA and USAF Tropical Conditions

10.5. Conclusion

This section shows that the aircraft can satisfy the Request for Proposal requirements; comfortably in most cases.

11. Structure

The major load bearing structure of the Vulture can be divided into three major categories. The fuselage, wing, and tail structure. The major aspects of each will be discussed in further detail in this section.

11.1. Wing Structure

The wingbox of the Vulture consists of three spars with six corresponding spar caps. The two main spars were positioned at 15% and 65% of the wing chord, creating a standard half chord wing box. A third spar was added at 40% of the wing chord extending only to the wing strut intersection. The purpose of the additional spar was to handle the compressive load on the wing that is induced by the strut under positive loading. Ribs were spaced equally at three-foot intervals along the span of the wing. The ribs were placed perpendicular to the wingbox to allow for ease of manufacturing as well as a reduction of materials and therefore weight. The major ribs are shown below Figure 11-1.

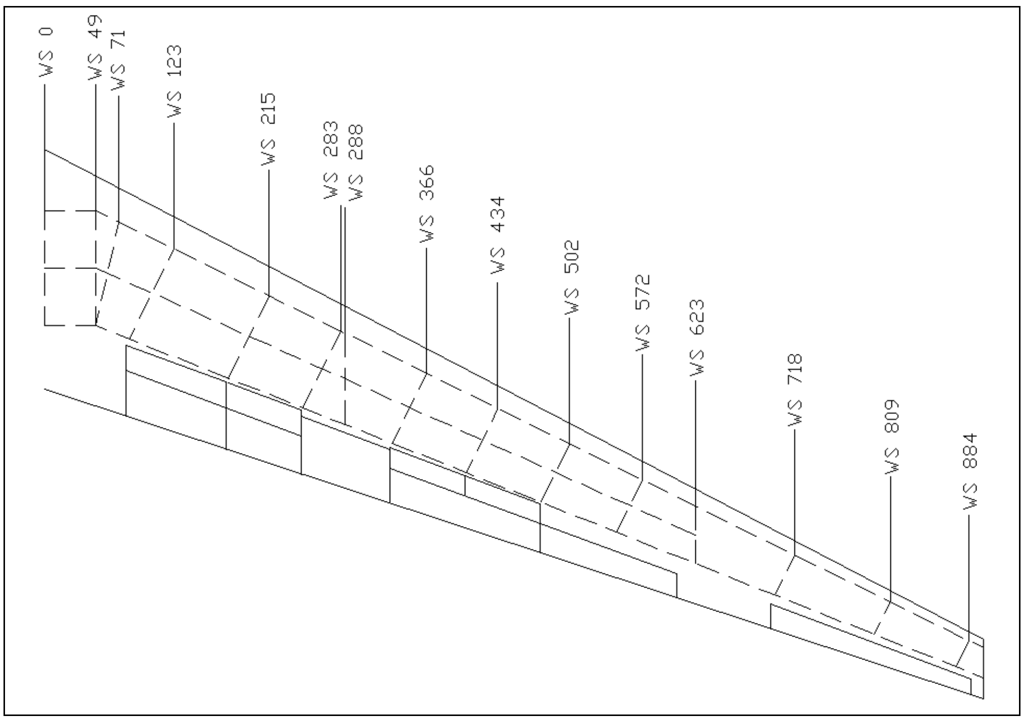


Figure 11-1 - Rib Placement (Only Major Ribs Shown)

The ribs were positioned in relation to the flaps allowing for at least two hinge points for each flap section. Additional ribs were placed to provide a secure connection for the engines and wing strut intersection. These ribs were placed parallel to the stream flow to align with structural constraints of these components ensuring a solid attachment for these critical installations.

The one drawback of the strut braced wing concept is the possibility of strut buckling under compressive loads. Since strut compression presents a problem in negative wing loading and taxi bump conditions, a damping system was devised to take these compressive loads and ensure that the strut is failsafe. When the strut is sent into compression, the damping system compresses with the strut, and prevents the strut from absorbing the negative loads and eliminates the chance of buckling. A picture of the damping system used on the Vulture can be seen in Figure 11-2. With this damping system in place it is possible to assume that the strut is inactive in compression.

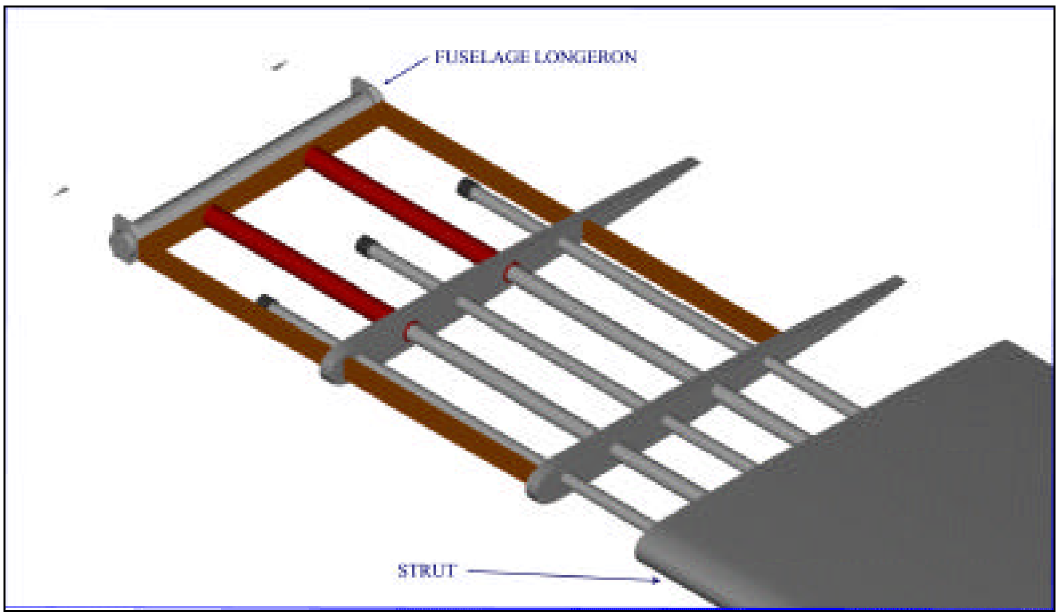


Figure 11-2 - Shock Damping System on The Vulture

One of the main advantages of the strut-braced wing on the Vulture is the reduction in shear force and bending moment along the span. With the strut attachment, shear and bending forces are reduced, allowing a reduction in wing thickness and therefore reduction in structural weight. Shear force and bending moment diagrams for the Vulture can be seen below in Figure 11-3 and Figure 11-4.

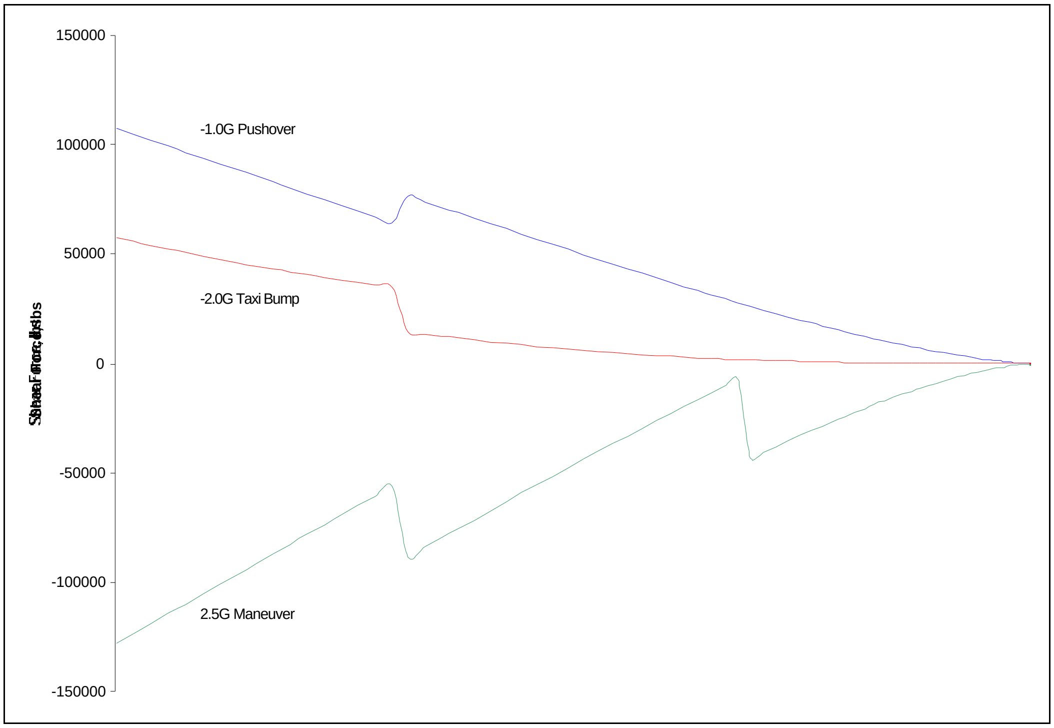


Figure 11-3 - Vulture Shear Force Diagram

As seen in Figure 11-3, discontinuities are created in the shear force diagram as a result of the engine weight acting downwards as well as the strut force. In the negative g cases, there is no discontinuity shown at the wing strut intersection as a result of the assumption that the strut is inactive in compression.

The Vulture's SBW design allows for a large reduction in the bending moment of the wing as seen in Figure 11-4. A discontinuity is created at the wing strut intersection, due to the vertical offset of the strut. The vertical offset is the result of tradeoff studies between aerodynamics and structures. From a structures perspective the strut intersection should be at the wing, while from an aerodynamics perspective, the strut should be connected to the wing at a 90 degree angle. The vertical offset distance is determined so that the end result is a structurally and aerodynamically efficient aircraft. Once again the negative g loading cases do not present this discontinuity, because the strut is considered to be inactive in compression.

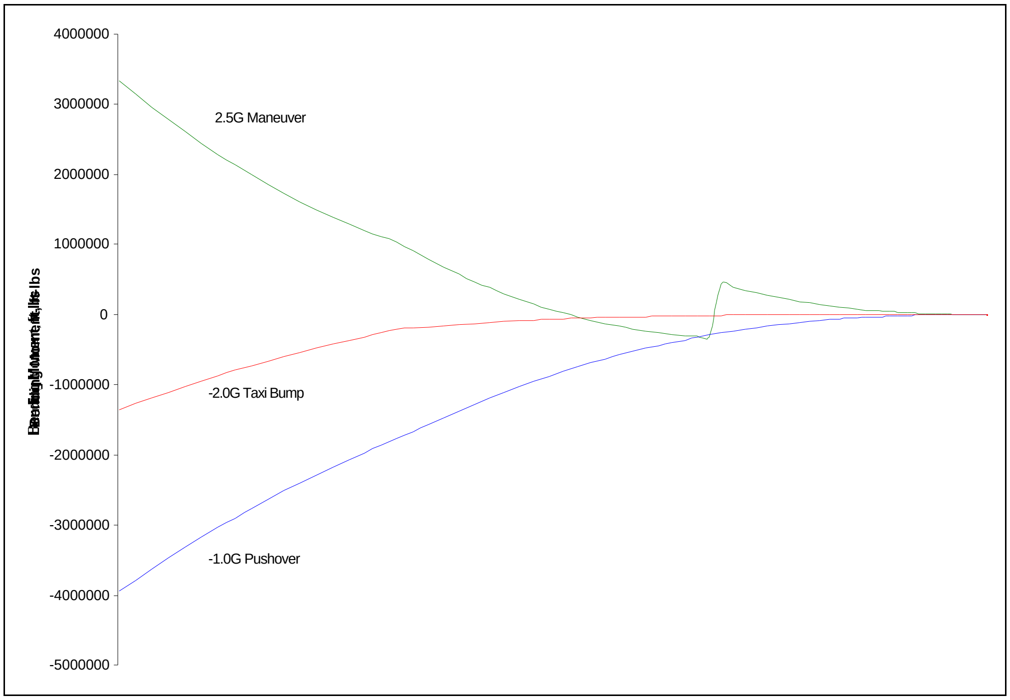


Figure 11-4 - Vulture Bending Moment Diagram

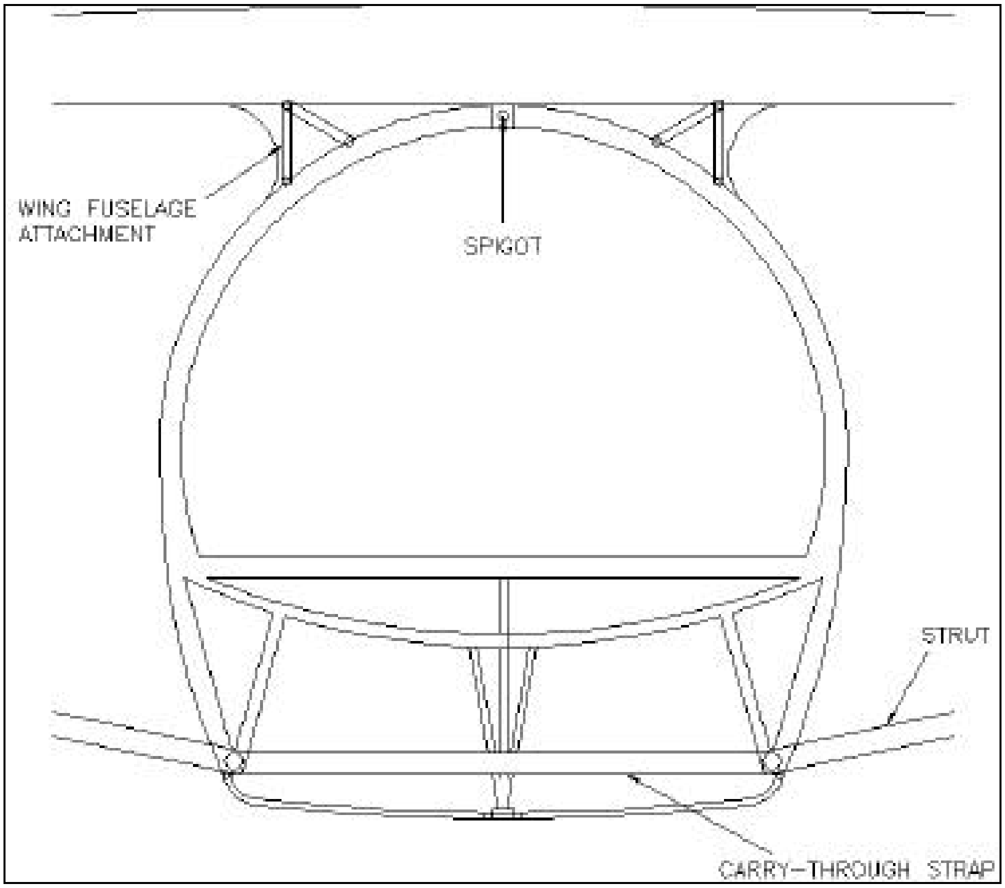
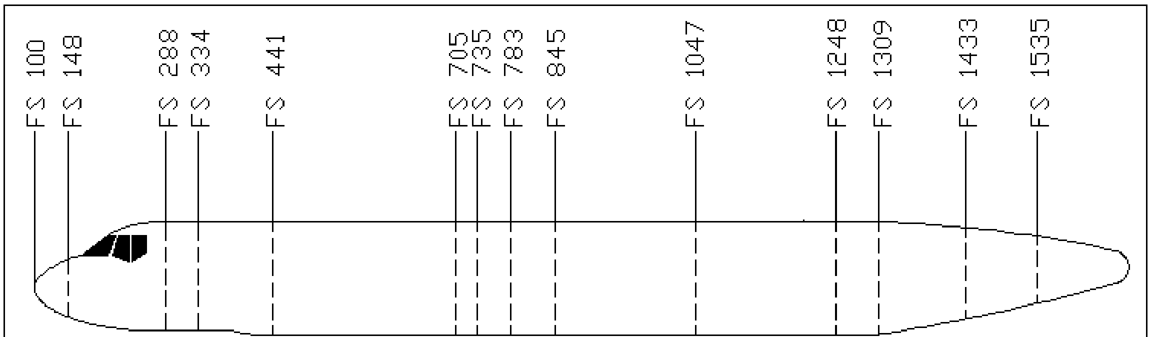


Figure 11-5 - Wing–Fuselage and Strut-Fuselage Connection

The wing is connected to the fuselage by four metal linkages at the fore and aft spar attachment bulkheads as well and to the locating spigot at the top quadrant of the fuselage. The strut is attached to the frame of the aircraft and is connected by a carry through strap to prevent the strut force from forcing the bottom of the plane apart. Figure 11-5 shows the connection of both the wing and the strut.

11.2. Fuselage Structure

The fuselage structure of the Vulture consists of thirteen major bulkheads as well as frames spaced at equal intervals of two feet. Figure 11-6 shows the placement of the bulkheads and the corresponding fuselage station. One keel is located beneath the floor at the centerline. The keel helps to resist the bending loads from the main cargo floor.



	Function	Fuselage Station (in)
Bulkhead 1	Forward Pressure Bulkhead	148
Bulkhead 2	Door Support, Nose Gear	288
Bulkhead 3	Door Support	334
Bulkhead 4	Missile Rack	441
Bulkhead 5	Missile Rack, Strut Carry through	705
Bulkhead 6	Wing Spar	735
Bulkhead 7	Missile Rack, Strut Carry through	783
Bulkhead 8	Wing Spar	845
Bulkhead 9	Missile Rack, Main Gear	1047
Bulkhead 10	Door Support	1248
Bulkhead 11	Missile Rack, Door Support	1309
Bulkhead 12	Vertical Tail Spar	1433
Bulkhead 13	Vertical Tail Spar, Horizontal Tail Pivot	1535

Figure 11-6 - Vulture Bulkhead Placement

11.3. Tail Structure

Since the Vulture has a T-tail, it is necessary to take special care in placing the structural elements to ensure structural integrity. The root chord of the horizontal tail is equivalent to the tip chord of the vertical tail in order to ensure a structurally sound connection. In both horizontal and vertical tails, spars were placed at 15% and 65% of the chord. The rear spar also serves as a pivot point for the horizontal tail so that is able to change its angle of incidence for trim. Ribs were spaced at an interval of three feet, and once again were place perpendicular to the spars. The major ribs can be seen in Figure 11-7. A kick rib was placed at the base of the vertical tail to help transfer loads to the fuselage.

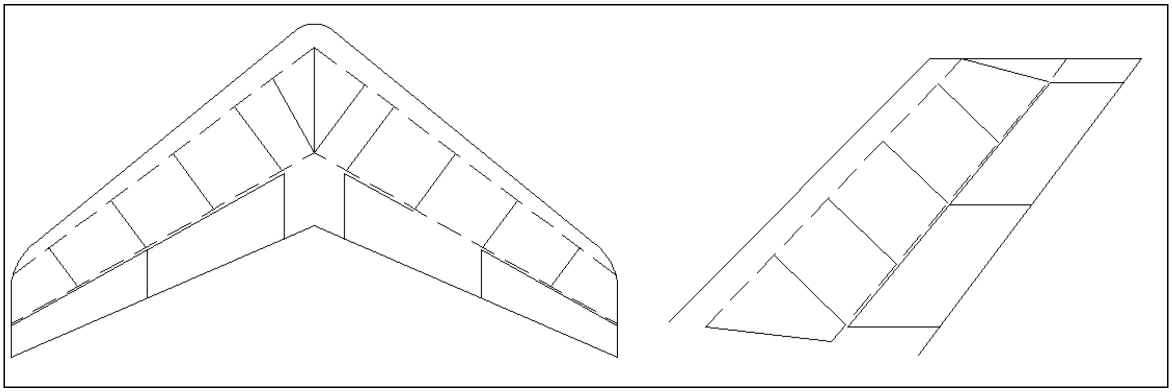


Figure 11-7 - Vulture Rib Placement In The Tails (Only Major Ribs Shown)

11.4. V-n Diagram

The V-n diagram, Figure 11-8, shows the maneuver envelopes and gust envelope for the Vulture. The maneuver envelope indicates the limits of speed after which structural limits of the aircraft will be exceeded. The maneuver envelope was determined using the stall parameters, maximum design loading, and the dive speed. To ensure the aircraft can withstand loads from turbulent air in flight the gust envelope was created. This envelope was created using dive speed gust of 18.3 ft/sec, high speed gusts of 36.52 ft/sec, and rough air gusts of 50.65 ft/sec. These gusts velocities are determined from FAR-25 Airworthiness Standards-Transport Requirements.

The stall line for the maneuver envelope was determined by finding the level flight stall speed, and correcting for accelerated flight using the following equation:

$$n = \frac{C_{L_{max}} q}{W / S} \tag{14.1}$$

The RFP stated the maximum positive load factor was 2.5 until the dive speed was reached. The maximum negative load factor for this aircraft is –1 until the equivalent cruise speed was reached and then varied linearly to zero load factor at the dive speed.

The gust envelope is created by gust lines which were found using the gust equation shown below:

$$n = 1 \pm \frac{K_g U_{de} V_e C_{L_{\alpha\alpha}}}{498(W/S)} \tag{14.2}$$

Here K_g is the gust alleviation factor, U_{de} is the equivalent gust velocity, V_e is the equivalent velocity of the aircraft, and W/S is the wing loading.

The gust alleviation factor is shown below:

$$K_g = \frac{0.88\mu}{5.3 + \mu} \tag{14.3}$$

Here the mass factor,μ, is:

$$\mu = \frac{2(W/S)}{gc\rho C_{L_{\alpha\alpha}}} \tag{14.4}$$

The combined V-n diagram is shown in Figure 11-8.

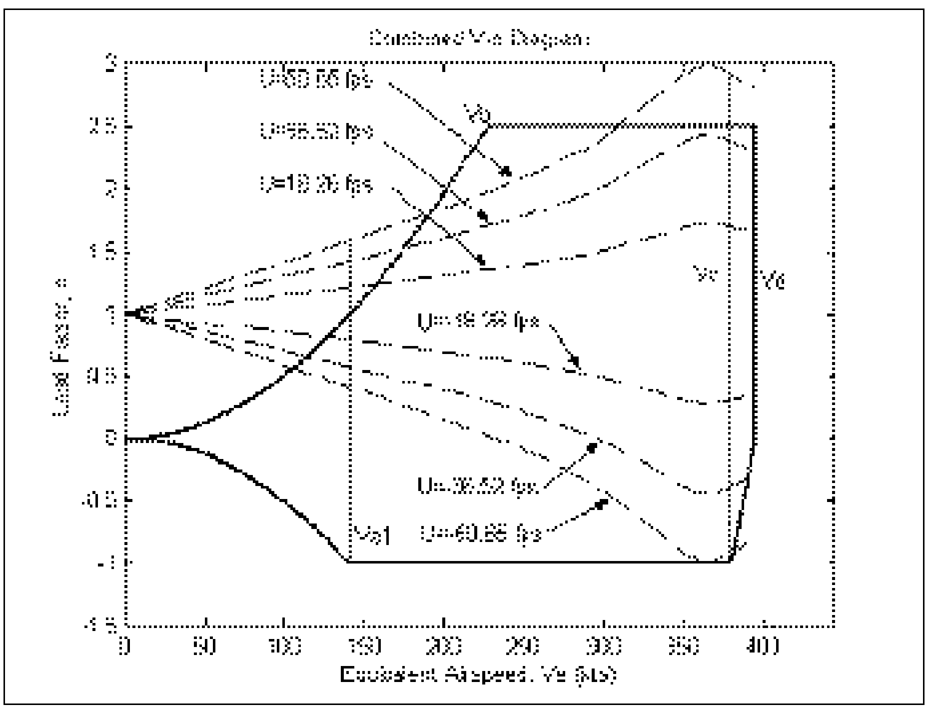


Figure 11-8 - Combined V-n Diagram

12. Undercarriage

12.1. Introduction

Considering the landing gear is only used for a small part of the operational mission, it represents a significant proportion of the overall cost and weight of the aircraft. For this aircraft a nose-wheel undercarriage configuration was chosen.

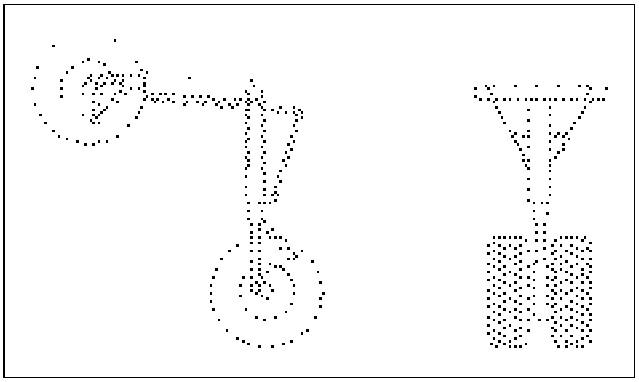


Figure 12-1 - Forward Undercarriage

The front nose gear is telescopic as opposed to offset, and benefits from not having any side loads and the added security of a second tire. Although this system will be considerable heavier than an offset design, it is more suitable for this aircraft, due to the greater mass of the aircraft compared to that more commonly associated with the offset design e.g. more typically fighters. The front landing gear is designed to retract forwards. The benefits of this configuration are twofold; the airflow past the gear can aide the lowering of the undercarriage if an emergency low hydraulic pressure situation arises, and advantage can be made of the bulkhead at the front of the first missile bomb-bay as a means of attachment. This design offers a more stable aircraft due to the position of the main undercarriage pivot, which is relatively close to the center of gravity (CoG), and therefore the nose-over condition is eliminated. Good maneuverability and pilot visibility on the ground due to attitude are also achieved. Wheels are stored in wells where, 3-inch gaps have been left around the wheel to allow for expansion of the tire after take-off and also to allow cooling air to lower the temperature of the tire. A cooling method employed on the trident, where deflected air channeled from vanes at the center of the wheel, deflects air as the wheel spins while the undercarriage is raised has been adopted. The tire will feature temperature monitoring and in the event of the tire temperature increasing too much, a plug will blowout on the tire and reduce the tire pressure, and prevent the tire exploding.

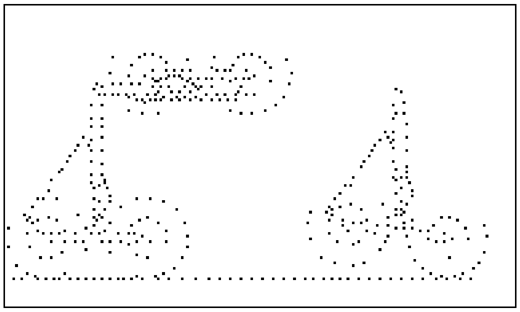


Figure 12-2 - Rear Undercarriage -

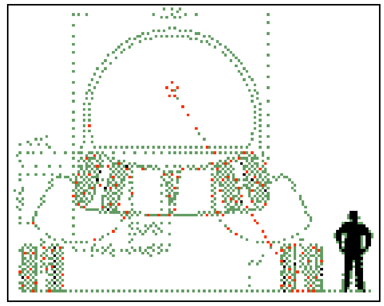


Figure 12-3 - Undercarriage Storage

The rear undercarriage closely resembles that used upon the Tupolev–134 and Lockheed C-141 Starlifter. It features a double bogey arrangement consisting of eight wheels, and incorporates an "anti-hop" damper mechanism. This is designed to dampen the rotation of the main undercarriage after the rear set of wheels hit the ground upon landing and the front pairs of the rear bogies begin to swivel around and hit the ground. A turnover angle of 60° has been used at the center of gravity, and 57°, above the pivot point of the main undercarriage, this compares favorable with the C-141 and other high-wing aircraft, which display turnover angles of between 61° and 63°.

12.2. Undercarriage Struts

The undercarriage has an oleo-hydraulic actuator on the strut that will be fixed onto a bulkhead at the front of the rear missile compartment, which is "shrinkable" when the undercarriage is retracted into the aircraft to reduce the length of the landing gear strut. This saves weight by removing the need for a dedicated bulkhead for the undercarriage. The undercarriage was found to be 3.30m long and has a 0.61m oleo-hydraulic cylinder that can be compressed as it is stowed away in the aircraft by 0.51m. The mechanism is designed to be simple and low cost. The actuators use a design where, oil is forced through orifices during compression, which offers good recoil damping and is simple and reliable, although a weight penalty is associated with these benefits. The landing gear is required to deal with 2g landings, however, the tires will withstand a landing of up to 2.44g. The Strut Compression for a 2g landing would be 0.102m, and if a 2.5g landing load was placed upon the strut, it would compress 0.23m, which means that there is plenty of compressible length available. At this stage in the design process, the nose gear is assumed to act the same in terms of compression required.

12.3. Tire Selection

The main undercarriage of the aircraft has been designed so that it should withstand 92% of the aircraft's weight and a Type VII 36"(dia.) x 11"(width) tire. For the nose gear, there were only two appropriate tires, but the model above was employed. There were a number of issues to deal with: There would be an 38.2 Kg weight penalty if the larger wheels were used and there would also be a loss of space. However, this is a large aircraft and as opposed to a fighter, the space is not at such a premium, and the proportion of the difference in weight over the aircraft is small. These points had to be traded-off against cost and logistics. If only one sort of tire is used, then the "House keeping" costs for the aircraft can be reduced, as the stores only have to look after one sort of wheel, and there would always be a wheel easily available for any part of the airplane. The wheel also has a commercial code TSO-C62b, which means that any company can manufacture it, which is desirable for the Armed Forces who would like to be assured that there is always going to be a supplier.

12.4. Braking

The landing gear will have an Electronic Anti-lock/Anti-skid system, and will meet current military and civilian requirements. The brakes will be carbon fiber discs.

13. Materials Selection

Material selection for individual components of the aircraft were chosen on there ability to reduce weight and cost, as well as structural integrity and facilitating reparability. To choose the most effective material each for component, material strength to weight and strength to cost ratios were calculated. Composite materials offer the best strength and stiffness to weight ratios and are competitive with conventional materials in cost. The major drawback associated with these materials is their damage tolerance, reparability, and defect detection. Composite materials have traditionally been used for doors and control surfaces, which are thin and relatively small. Large surfaces such as the wing and fuselage have never been made on an aircraft of this size, with one exception, the B-2, which utilized these materials for there radar absorption properties. Currently, research in creating large surfaces of composite materials is being performed, but the most cost effective material is still conventional aluminum. Using conventional metals for these large surfaces would allow the use of existing manufacturing techniques and tooling which would reduce the manufacturing cost over composite materials. Thus areas of the aircraft prone to debris strike will be made of conventional metals, which are less costly to repair. Composite materials were mainly utilized for the flaps, control surfaces, and weapons bay and landing gear doors.

The structures carrying the primary loads such as the bulkheads, keel, heavily loaded longerons, and wing spars will be made of Al-Li 2020, which is has 10% greater stiffness and strength, and a 15% reduction in density over conventional aluminum. This material was used only for structures carrying primary loads because the cost increase over conventional aluminum would not merit its usage in less critical areas. Al 2024 was used for the ribs, longerons, frames, weapons bay, and wing skins because of it prevalence in the aerospace industry which reduces cost of manufacture and maintenance.

Graphite epoxy will be used for the vertical and horizontal tail, strut, engine cowling, flaps and all control surfaces. The radome will be made of E-glass because of its favorable dielectric properties for the radar equipment. Graphite epoxy was selected for its relatively low cost and weight reduction for objects in this size range Figure 13-1 shows a cost based justification for the use of composite materials on the vertical and horizontal tail of the aircraft. This figure shows the wide range of cost savings by using composite materials on the vertical tail on the L-1011.

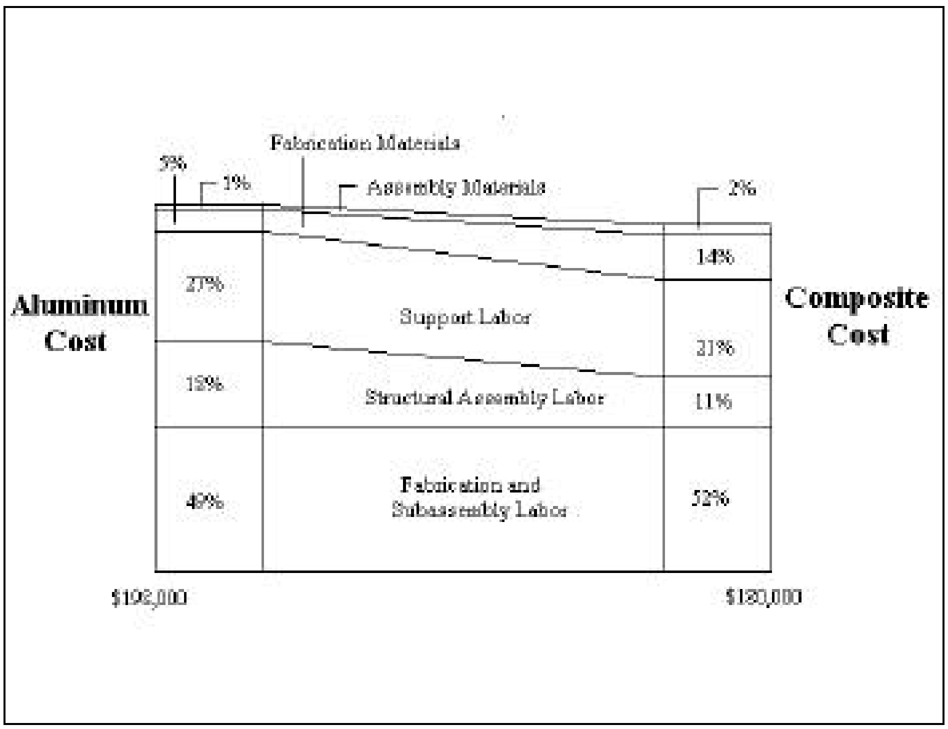


Figure 13-1 – Cost savings on the L-1011 composite vertical tail (Reference 13.1)

14. Stability

The Vulture was designed to deliver its payload efficiently and safely. The aircraft must be controllable in spite of CG travel resulting from its missions as both a cruise missile carrier and transport. Relaxed static stability with its low attendant trim loads is a key feature of the Vulture and allows the Vulture to attain low induced drag at cruise but also forces the Vulture use stability augmentation to maintain longitudinal stability.

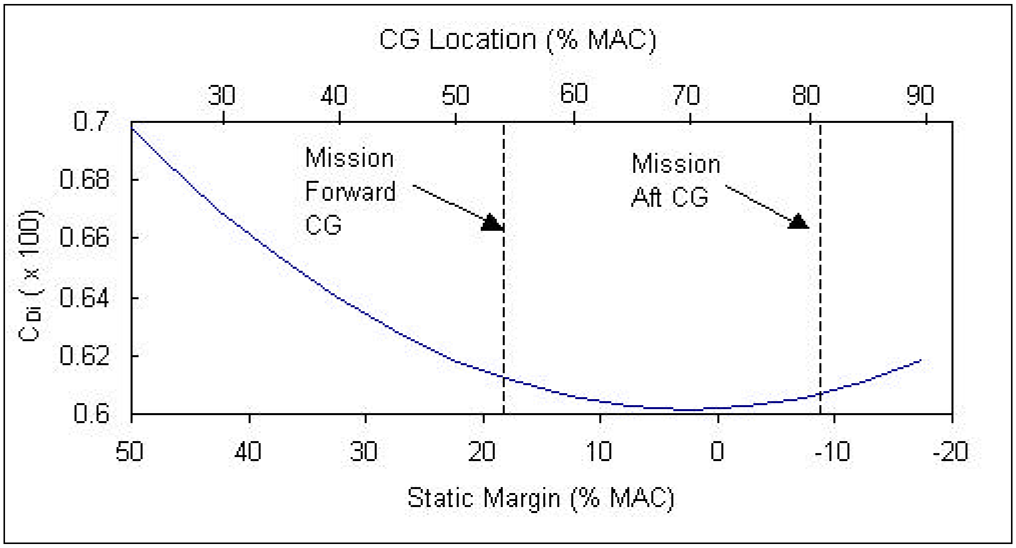


Figure 14-1 - Drag vs. Static Margin

14.1. Design Philosophy

Minimizing drag is an important design goal for an aircraft with a 5,000 nautical mile range. Placing the CG at an optimal location reduces the induced drag of an aircraft trimmed for cruise (Ref 14.3). As the CG is moved aft on an aft-tail vehicle, the download on the horizontal tail is reduced or may even become positive. The reduced tail download reduces the lift that must be produced by the main wing and therefore reduces induced drag. Lutze (Reference 14.1) presents a technique for estimating the CG location that results in the in the minimal induced drag and his technique was used to produce the curve in Figure 14-1. Figure 14-1 shows the induced drag verses CG location for the Vulture at the cruise conditions of Mach 0.85 and 40,000 ft. The vertical lines in Figure 14-1. represent the most forward and reward CG locations encountered over the Vulture's mission. The Vulture was designed so that its CG transverses the drag curve at a minimum. To achieve a minimum drag design, the CG was allowed to travel aft of the neutral point. The Vulture's stability margin is largest after loiter, equal to 18.3%, and is most unstable during missile drop, equal to –9.1%. A longitudinal Stability Augmentation System (SAS) was used to keep the Vulture stable. It is also important to note that the Vulture's CG location lies aft on the MAC; the aft location results placing the CG near the aircraft's neutral point of 72.6% MAC.

14.2. Stability and Trim

The Vulture's longitudinal control system controls the 425 ft^2 horizontal tail and a 120 ft^2 elevator. The elevator has an operational deflection range of 20 deg down to 20 deg up and produces the primary pitch control moments. The elevator is partitioned into two independent sections to increase redundancy. Due to the Vulture's high transonic cruise speed, shock waves can appear on the elevator and increase the aircraft's wave drag. To alleviate this problem, the Vulture incorporates a variable incidence horizontal tail to trim the aircraft at cruise. The variable incidence horizontal tail is also trimmed for takeoff to reduce the Vulture's takeoff roll. The horizontal tail can rotate nose-down to 5 degrees and nose-up to 10 degrees. The control surfaces are connected to the pilot input through a computerized SAS. Important longitudinal stability derivatives were obtained from Roskam and are listed in Table 14-1 for a take off static margin of 0.6% and a cruise static margin of 3.2% (References 14.2, 14.3).

Table 14-1 - Longitudinal Stability and Control Derivatives

Flight Conditions	Take-off	Cruise
Altitude (ft)	0	40000
Density (slugs/ft^2)	0.002377	0.00058727
Speed (ft/s)	211	823
Initial Attitude (deg)	10	3.6
Weight (lg)	181000	181000
Steady State Coefficients		
C_{L1}	1.9110	0.5084
C_{D1}	0.339	0.0022
C_{Tx1}	0.339	0.0022
Longitudinal Derivatives		
$C_{L\alpha}$	5.7696	8.1347
C_{Lq}	6.4355	0.2612
$C_{m\alpha}$	0.0359	4.3572
C_{mq}	-37.7720	-44.7010
$C_{m\dot{\alpha}}$	-8.3231	-9.8346
Elevator		
$C_{L\delta e}$	0.9361	0.9361
$C_{m\delta e}$	-4.3109	-4.2579

The aft CG limit is limited by the control system so that the feedback gain of elevator input per change in angle of attack does not exceed 5 deg/deg. The critical value was specified by Roskam (Reference 14.4) as the maximum feedback gain for safe operation. The aft limit is defined by a static margin of –15%.

14.3. Longitudinal Dynamics

Roskam's stability analysis techniques were used to evaluate the Vulture's stability. The Vulture meets or exceeds MIL-F-8785C requirements for phugoid oscillations (Reference 14.4). Table 14-2 shows the military specifications for the phugoid motion and the Vulture's characteristics. The Vulture is well damped in all flight phases but exhibits minimum damping value at take off.

Table 14-2 - Longitudinal Handling Qualities

	Phugoid	Short Period	
	ξ_p	$(\omega_{sp})^2/n/\alpha$	ξ_{sp}
Mil-Spec	$0.04 < \xi_p$	$0.15 < (\omega_{sp})^2/n/\alpha < 3.6$	$0.35 < \xi_{sp} < 1.4$
Vulture Take Off	0.05756	2.5	0.8
Vulture Cruise	0.1954	2.5	0.8
Vulture Landing	0.1201	2.5	0.8

Without its SAS, the Vulture is unable to meet MIL-F-8785C's requirements for short period motion because of its marginally stable design. The SAS employed by the Vulture uses both angle of attack (Kα) and pitch rate feedback (Kq) to modify the short period oscillation. The gain values are estimated by Roskam's techniques (Reference 14.4). Ka values range from 0.94 to 2.83 and Kq values range from –0.081 to –0.25 and both depend on the dynamic pressure. The gains are scheduled to produce the same acceptable, short period characteristics throughout the Vulture's flight entire envelope.

The Vulture's longitudinal control system employs an angle-of-attack limiter to prevent the aircraft from entering a locked-in deep stall. The Vulture's T-tail design possesses locked in-deep stall potential because the wing and fuselage disrupt the airflow over the horizontal tail at very high angles of attack (Reference 14.5). The aircraft can become locked into the stall when the horizontal tail can no longer produce a moment to pitch the nose down and break out.

14.4. Lateral-Directional Stability and Trim

The lateral-directional control system consists of a rudder, inboard and outboard ailerons, and wing-mounted spoilers. The rudder is 92.6 ft2 of the 325 ft2 vertical tail. Like the elevator, the rudder is split into two sections for redundancy. Two high-speed ailerons of a total area 62.9ft2 located behind the engines and two tip located low speed ailerons of total area 51.9 ft2 generate rolling moments. The low speed aileron doubles as a flap for landings and takeoffs but is limited to 15 degrees down deflection due to engine exhaust. Additional roll control is provided by 57.6 ft2 of inboard spoilers and 51.9 ft2 of outboard spoilers; each spoiler has a maximum deflection of 60 degrees. The lateral directional stability derivatives are listed in Table 14-3 for a take off static margin of 0.6% and a cruise static margin of 3.2%.

An important requirement in for the Vulture is that lateral-directional control be maintained in the asymmetric thrust condition resulting from the loss of an engine. FAR 25.149 requires that the aircraft maintain straight and level flight at 1.2 times the stall speed with one engine wind-milling and the other at full power during which the bank angle cannot exceed the maximum value of 5 degrees (Reference 14.6). Reference 14.6 was used to estimate the drag produced by a wind-milling engine so that an accurate yawing moment could be estimated. The Vulture met the requirements for a failed engine at the critical conditions of take off and cruise. At a sea level takeoff, while flying fully loaded at 130 knots, the Vulture requires a rudder deflection of 17 degrees and an outboard aileron deflection of 17 degrees and an inboard deflection of 15 degrees. At cruise while flying at 40,000 ft and Mach 0.85, the Vulture requires a rudder deflection of 18 degrees and minimal aileron deflection. The large rudder deflection offset the large yawing moment generated by the drag from the wind-milling engine. Both deflections are under the maximum rudder deflection, maximum inboard aileron deflection, and maximum outboard aileron deflection.

14.5. Lateral/Directional Dynamics

The Vulture has acceptable spiral stability as determined using Roskam's methods (Reference 14.4). The Vulture possesses a Dutch roll mode that meets the specifications laid down by MIL-F8785C. Table 14-4 displays the frequency and damping of the Vulture's Dutch roll oscillation. Although the damping during takeoff and cruise does meet specs, it does so with a small margin. Roll- and yaw-dampers remain an option to improve the Dutch roll response of the Vulture, but preliminary calculations show that they are not needed.

Table 14-3 - Lateral-Directional Stability Derivatives

Flight Conditions	Take-off	Cruise
Altitude (ft)	0	40000
Density (slugs/ft^2)	0.002377	0.00058727
Speed (ft/s)	211	823
Initial Attitude (deg)	10	3.6
Weight (lg)	181000	181000
Steady State Coefficients		
C_{L1}	1.9110	0.5084
C_{D1}	0.339	0.0022
C_{Tx1}	0.339	0.0022
Lateral-Directional Derivatives		
C_{lp}	-0.4451	-0.6963
C_{np}	-0.2407	-0.2340
C_{yr}	0.5556	0.5684
C_{lr}	0.1530	0.0156
C_{nr}	-0.1841	-0.1957
$C_{y\beta}$	-0.6981	-0.7163
$C_{l\beta}$	-0.1236	-0.1310
$C_{n\beta}$	0.1123	0.1316
Inboard Aileron		
$C_{y\delta a}$	0.0000	0.0000
$C_{l\delta a}$	0.0318	0.0318
$C_{n\delta a}$	-0.0287	-0.0287
Outboard Aileron		
$C_{y\delta a}$	0.0000	0.0000
$C_{l\delta a}$	0.0446	0.0446
$C_{n\delta a}$	-0.0067	-0.0067
Rudder		
$C_{y\delta r}$	0.2120	0.1986
$C_{l\delta r}$	0.0188	0.0134
$C_{n\delta r}$	-0.1051	-0.0991

Table 14-4 - Lateral-Directional Handling Qualities

	Dutch Roll		**Roll**	
	ξ_{dr}	ω_{dr} (rad/sec)	t_{30deg} (sec)	τ_r (/sec)
Mil-Spec	$0.4 < \xi_{dr}$	$0.4 < \omega_{dr}$	$t_{30deg} < 2.3$	$\tau_r < 1.4$
Vulture Take Off	0.2007	0.5461	1.49	1.281
Vulture Cruise	0.09767	1.082	0.85	0.5711
Vulture Landing	0.2747	0.8198	1.193	0.567

Military specifications place demands on the roll mode which the Vulture meets throughout its flight envelope. Table 14-4 shows the required time to bank values with the roll mode time constants in comparison with the Vulture's performance. The Vulture was able to achieve the specified roll rate even at the critical conditions of take off and landing.

15. Systems

15.1. Introduction

The aircraft Systems should allow the aircraft to do the following:

Table 15-1 - System Requirements

- Be aware of aircraft within 300 miles
- Navigate within 50 m enemy airspace, 20 m friendly airspace
- Combat all electronic threats the aircraft could expect to encounter on a typical mission.
- Allow the aircraft to function effectively and reliably as required.

15.2. Avionics

Table 15-2 - Avionics Demands

- A high-speed, low cost architecture with low latency
- Provide instant diagnostic information of the aircraft and its systems, which can be sent to ground stations via a data link and supports real time computing
- Support for both message passing and shared memory computing
- System are scalable and will accommodate expanded duties
- Support for a Serial and Parallel connections and, Electrical and Optical physical sources.
- Systems that are highly fault tolerant and relatively insensitivity to distance from sources.
- Support for real time computing

Scalability is required to allow numerous new functions to be added and an increase in performance technology, which will occur throughout the life of the aircraft.

15.2.1. Avionics Components

Many "standard" avionics will be on-board, but particularly attention is drawn to those in Table 15-3.

Table 15-3 - Key Avionics Components

- Ring Laser Gyro - Inertial Navigation System
- Global Positioning System
- Stores Management, incorporating Center of Gravity calculator for Cruise Missiles and Cargo
- Electronic Counter Measures, including: Radar Warning Receivers and Laser Warning Receivers
- FLIR vision enhancement system
- Secure Voice & Jam resistant (UNF/VHF/HF)
- Wireless intercom system
- Identification Friend or Foe/Selective ID Feature
- The Nemesis anti-missile defensive system.

Where possible, the aircraft will make use of “off-the-shelf” components. The Radar, for example, is a fully coherent Pulse-Radar by Northrop Grumman, with capabilities shown in Table 15-4.

Table 15-4 - Radar Capabilities

· 300 miles coverage
· Five modes including:
· Weather/turbulence
· Predictive windshear
· High resolution Ground map

It includes integration with the collision avoidance system TCAS. As the radar is not used for programming missiles, its cost is reduced. This system is also used on the Hercules C-130J.

15.2.2. Data Links

Aircraft will most likely make use of real-time communications to bases using satellite communications. In hostile airspace, this made not be used, as, it could perhaps be used as a homing signal. Therefore, the aircraft must be able to operate independently of ground based systems in hostile territory. As more information is outputted, the need to recover the “Black box” will diminish, as all aircraft data and voice data recordings will be sent back to ground stations, thereby eliminating the huge costs sometimes associated with recovering the “Black boxes,” in all but the most extreme cases.

15.3. Flight Control System (FCS)

FCS in Vulture is advanced fibre optic Fly-by light system. FBL offers advantages over conventional FBW in that it is lighter and less effected by Electro-magnetic interference and EM weaponry. Also it removes need for heavy EMI shielding material. The system is designed for multiple redundancy during data transmission stages. Two flight computers carry out primary signal processing. One located behind the cockpit and another in the rear of the aircraft. Which convert the commands of the pilots into an optical signal that is sent in the form of a laser though various fibre-optic cables to the control surfaces. Additional redundancy is designed into the airframe in the form of multiple control surfaces. Each large surface is divided into smaller surfaces, which continue to provide powered control if one or more surfaces receive damaged signals. For actuators the electricity comes by direct conversion of the optical signal to electricity via photo voltaic cells located at the actuators. This eliminates the need for electrical routing through the aircraft and further enhances immunity to EMI. We are using a digital FBL system, with a reliability criterion of 10^{-9} per flight. All electric aircraft are going to be of greater advantage on FBL aircraft such as our SBW.

15.4. "All Electric Aircraft"

15.4.1. Actuation

Actuation will be required for each control surface, the bomb bay doors and the undercarriage. Electrical actuators will be used on control surfaces. They offer great advantages for large aircraft, due to removal of the piping required to reach the actuators, and the size quantities of fluid required to move these control surfaces, these advantages increase with the size of aircraft. Hydraulic systems will be used for the undercarriage and bomb-bay doors. The actuators are of a linear or rotary type and the system will operate at a pressure of approximately 270 bar (4000 psi) as used on current aircraft.

15.4.2. Secondary Power Systems

The aircraft requires a source of power other than the primary need for propulsion. The aircraft will be self-sufficient, with autonomous starting of the engine and reduced maintenance. The aircraft's systems will be able to be report without running the primary engine or using ground equipment.

Secondary Power System can be positioned away from the engines to reduce drag. There is an increased demand in power output of from avionics, but with a reduction in ownership cost the elimination of seals and clutches, helps pro-long the operational life and reduce life-cycle costs. For an integrated system to supply the power requirements of the aircraft drive must be provided by either.

Table 15-5 - Benefits of "All-Electric Aircraft" to Maintenance

· Improved aircraft operations and maintenance
· Increased efficiency in power generation
· Flexibility in launching and maintenance
· Operational readiness on short notice
· Remote airbase operation
· Reduced hazard of accidents using ground support equipment
· Fewer personnel required for operation
· Life cycle cost reduction
· Less on-ground main engine operation, improving life and fuel consumption
· Reduced maintenance cost of an APU when compared to ground support equipment

Table 15-6 - Power Sources

· The engine
· The auxiliary Power Unit (APU)
· Ground Power Source
· Emergency Power

Electrical Secondary Power system involves an electric starter mounted in the accessory gearbox, supplied from APU generator, but it is now possible to have a dual starter/generator, which increases the feasibility of this power system. The advantages of this are that, the APU is placed away from the main engines, and the systems are increasingly integrated and simplified. The number of ground carts required to service the aircraft are reduced, thereby significantly reducing the ground support (maintenance/logistic) costs and number of personnel and their related costs (where ground equipment is associated with multiple power sources is replaced with electric power). Reduction of ground support is one of the main aims in designing this aircraft, the system must be designed to be self-sufficient.

The high starter power requirements have only been recently been feasible to supply, courtesy of Samarium Cobalt motors (SmCo), and these are integrated into the engine. The emergence of these rare earth magnets allows SmCo motor-actuators for primary flight controls. SmCo motors are not only highly efficient and reliable, their power/weight ratio was the technology breakthrough needed for this Power System to be considered.

Advantages over other Secondary Power Systems include: Hydraulic system would be complicated as ground support would have to interact with the hydraulic system. Large routing of ducts, and heavy connections in mechanical links. It is difficult to compare systems on the basis of weight and size, but there are strong arguments for the “All-Electric Aircraft”. Research by NASA/Lockheed assigned a weight penalty to the electric actuator, but more recent weight projections suggest that advanced SmCo actuators may be lighter than their hydraulic counterparts. Boeing have actually found in some cases that their electric actuators are lighter.. Electrically driven motors supply powers to the hydraulics in the undercarriage and bomb bays. The main advantage for this aircraft is that piping fluids to the control surfaces can cause problems with routing, and also there is an increased weight penalty. Maintenance is a lot easier, as a high system fluid pressure does not have to be maintained. All-electric aircraft allow one or more electric generators to achieve a more simple and reliable secondary power systems. Recent NASA/Lockheed studies have identified the all-electric aircraft as an energy efficient transport that eliminates labour-intensive systems such as high-pressure hydraulics, engine bleed air, pneumatics and the electric engine start systems.

The practice of bleeding air from a high-bypass ratio engine has become an aspect of major concern because of the thrust/specific fuel consumption penalties. Bleed air demand impacts adversely on the engine's fuel consumption and its aerodynamic design. This design also simplifies the mechanical complexity of the engine. Weight savings for reduction of bleed air systems (ports etc) from NASA/Lockheed study) show savings of 1300lb (in three engines) and savings of 2540 lb when the stainless steel power ducting was removed from the engines, pylons, wings and empennage. This simplification results in a reduction in the time it takes to install and remove engines.

15.4.3. Environmental Control System

This is mainly to do with thermal management of Advanced Avionics equipment. Large military transport and large commercial transport should achieve major benefits in performance, production and maintenance support costs. Pratt and Whitney design increases SFC 3.2%. ECS is a labor-intensive system since a significant amount of customized stainless steel ducting is routed from the engines and the APU, to the fuselage distributing ducting. The major production gain in the all-electric airplane is therefore the elimination of all the power ducting and the associated control valves/mounting hardware in the engine. The ECS plays a major role in the sizing of the sizing of the electrical power.

In replacing a bleed air powered ECS with an all-electric ECS (AEECS), cabin pressurization must be achieved through the use of dedicated motor-driven compressors. This could be implemented by a motor-driven bootstrap or motor-driven Freon type-compressor system, which has been chosen in previous systems for large aircraft. NB Freon is now not allowed for such systems, due to environmental concerns, so an alternative would have to be adopted.

Engine bleed out elimination is a major objective, the other main advantages of the all-electric ECS are that the electric turbo-machinery can be conveniently located in a below fuselage compartment and the airlines do not have to supply pressurized air.

15.4.4. Combined EPU/EPU

Carrying two power system son the Vulture to carry the same job is a waste. It is desirable for APU and EPU systems to bed combined. Using methods proposed by Friedrich and Schaper (see references), the APU can be integrated with the EPU. The main advantage of this would be to lower the number of components on the aircraft's inventory and increase the maintenance time turnover. Associated benefits should include decreased weight. Weight saving on propellant fuel. Tank and expulsion system and EPU Components (up to 15%).

The major use of the auxiliary power unit is to start engines, and be able to restart in case of a flameout. For the secondary power system to be able, to do this, it must be able to operate over the full envelope of the aircraft. Difficult for APU at altitude due to low pressure. APUs cannot easily be made to start in all conditions, and Emergency Power unit (EPU) needs to be installed, especially given engine out installation.

Table 15-7 - APU/EPU Integration

· Improved Specific Fuel Consumption due to reduction in bleed air
· Increased power density and specific weight to reduced installation penalties
· Better Reliability and maintenance
· Increased altitude operation, to restart after flameout.
· Reduced run-uptime for in-flight starting
· Reduced power losses (up to 65% on ground and 25% in flight
· Maintained flight safety

To improve the SFC and the specific power, the turbine inlet temperature must be increased as much as possible. The balance must be made to allow for reliability and cost at the chose compressor pressure ratio. It is obviously desirable to minimise the SFC and the specific power must be maximised to keep the APU inlet airflow low. This results in a reduction in the weight and size of the APU.

15.5. Fuel Systems

The fuel will be carried in eight tanks, with three of these tanks in each side of the wing, and alongside the fuselage mount. With the exception of the forward tank, these tanks are integral with reticulated foam inside to inhibit fire and explosion. The forward fuselage tank is a self-healing and tear resistant bag to minimize the risks associated with combat damage. The system will be pressurized to prevent the fuel boiling at altitude.

Provision for an in-flight refueling receptacle has been made on the top of the aircraft behind the cockpit similar to current stealth aircraft. To allow for other nations to use a probe/drogue system, provisions for a retractable flight probe has been placed to the right of the cockpit.

16. Costs

16.1. Fly-Away Costs

Fly-away costs were estimated for a production run of 140 CMC versions and 460 cargo/tanker versions. The fly-away cost was estimated using the cost algorithm described by Roskam (Reference 16.1). This method used historical data and combines them with scaling factors that allow the data to be analyzed for future time spans. The fly-away cost is made up of two main sub-categories: research, development, testing and evaluation (RDT&E) costs and manufacturing costs. The RDT&E cost is calculated for both versions simultaneously due to their near identical airframe configuration while the manufacturing cost was calculated for both the CMC version and the cargo version because of their differences. The RDT&E cost is divided into the following categories: airframe engineering and design, development support and testing, flight test aircraft, flight test operations, test and simulations facilities, RDTE profit, and cost to finance. Table 16-1 shows the cost breakdown of the RDT&E costs for the entire production run of 600 aircraft.

Table 16-1 - Breakdown of the RDT&E Costs

Category	Cost
Engineering and Design	704.7
Development Support & Testing	121.7
Flight Test Aircraft	945.7
Flight Test Operations	54.8
Test & Simulations Facilities	608.9
RDTE Profit	304
Cost to Finance	532
Total RDT&E Costs	**3,045**

*All cost in millions of year 2000 dollars

Table 16-2 - Breakdown of the Manufacturing Costs for the CMC Version

Category	Cost
Tooling Cost	1,132
Aircraft Production Cost	21,927
Acquisition Finance Cost	4,153
Acquisition Profit	2,769
Acquisition Cost	30,453
Acquisition Cost / Unit	**59.22**

* All costs in millions of year 2000 US dollars

The manufacturing cost can also be broken down into several categories, which include: airframe tooling, aircraft production, production flight testing operations, and cost to finance. The manufacturing cost breakdown for the CMC version is shown in Table 16-2 and for the tanker/cargo version in **Error! Not a valid bookmark self-**

reference.. The fly-away costs were calculated below by adding the manufacturing cost and the RDT&E cost proportional to the number of aircraft being produced. Table 16-4 shows the flyaway cost for both the CMC and cargo versions.

Table 16-3 - Breakdown of the Manufacturing Costs for the Tanker/Cargo Version

Category	Cost
Tooling Cost	1,078
Aircraft Production Cost	20,883
Acquisition Finance Cost	3,955
Acquisition Profit	2,637
Acquisition Cost	29,003
Acquisition Cost / Unit	**56.4**

* All costs in millions of year 2000 US dollars

Table 16-4 - Fly-Away Cost for CMC and Cargo Versions

Type	Fly-Away Cost	Fly-Away Cost / Aircraft
CMC	9,424	**67.32**
Cargo	29,674	**64.51**
*All cost in millions of year 2000 dollars		

The flyaway costs for the two versions vary by nearly four million dollars mainly because of the additional systems incorporated in the CMC version as well as the additional armament installations required. Modifieds for commercial operations, the tanker/cargo version has fly-away cost of 64.51 million dollars which is very competitive with a Boeing 757-200 which has an acquisition cost of 61-68 million dollars depending on the options included. The Vulture's increased cost, compared to the 757, is justified because this aircraft has a range increase of nearly 1,000 miles and a higher cruise speed, increasing it's productivity and work capacity. These attributes make this aircraft more appealing to delivery companies such as Fed-Ex or UPS.

16.2. Operational Costs

The operational costs for the Vulture were calculated for the CMC version and the commercial cargo version of the aircraft. The operating cost for the CMC version was based solely on the direct operating cost (DOC) since it is a military aircraft. The operating cost of the commercial cargo aircraft can be split into two categories: DOC and indirect operating cost (IOC). Direct operating costs are a result flight operations such as: fuel, crew, maintenance, insurance, and depreciation costs. Indirect operating cost are considered to be any cost that did not result from flying the aircraft such as administrative, sales, and aircraft handling. The IOC cannot be accurately approximated because it is varies widely with each purchaser of the aircraft. For this reason the IOC was assumed to be 60% of the DOC for commercial cargo.

16.2.1. Direct Operating Costs (Cruise Missile Carrier)

The DOC was calculated for the CMC role using the algorithm provided by Roskam. This estimation assumed an operational life of 25 years, and 800 annual flight hours per aircraft which is similar to current aircraft (GAO). The costs used in the estimation were direct personnel, indirect personnel, fuel and lubricant, maintenance support, training, spares and depots. The DOC was calculated per nautical mile based on the design range of 5000 NM. The cost was calculated for a production run of 140 aircraft, which would replace the current fleet of B-52s and B-1Bs. The operating cost per hour for the Vulture was calculated as $9,364/hr. The operating cost break down is shown in Table 16-5.

Table 16-5 - Breakdown of the Total Operational Costs for CMC Version (800 flight hrs/yr)

Category	Cost
Direct Personnel Cost	37,871
Indirect Personnel Cost	13,421
Fuel, Oil and Lubricants	4,571
Maintenance Materials	2,475
Spares	17,511
Depots	17,792
Total Operational Costs	93,640
Operational Cost per Hour	**9,364**[1]

* All costs in millions of year 2000 US dollars

[1] Cost in thousands of dollars

Table 16-6 shows the operational cost breakdown if the flight utilization of the aircraft is increased to 2000 hours per year.

Table 16-6 - Breakdown of the Total Operational Costs for CMC (2000 flight hrs/yr)

Category	Cost
Direct Personnel Cost	57,366
Indirect Personnel Cost	23,926
Fuel, Oil and Lubricants	17,139
Maintenance Materials	5,567
Spares	31,217
Depots	31,717
Total Operational Costs	166,936
Operational Cost per Hour	**7,419**[1]

* All costs in millions of year 2000 US dollars

[1] Cost in thousands of dollars

16.2.2. Direct Operating Costs (Tanker/Cargo Version)

The DOC was calculated for commercial cargo role was again calculated using the algorithm provided by Roskam. This estimation assumed an operational life of 25 years, 1400 annual flight hours per aircraft. The costs analyzed for this version were the same as those used in the CMC version. The IOC was calculated as 60% of the DOC and added to find the operational cost per nautical mile. The costs were analyzed for a production run of 460 aircraft, which would be used to fill the military tanker/cargo, civilian cargo roles. These values are shown in Table 16-7.

Table 16-7 - Breakdown of the Operational Costs Per NM for Commercial Cargo Aircraft

Category	$/nm
Maintenance	1.1744
Fuel	2.4957
Crew	1.101
Interest	1.5414
Insurance	0.1556
Depriciation	1.7616
Total DOC	**8.231**
Total IOC	4.938
Operating Cost	**13.169**

* All costs in millions of year 2000 US dollars

Table 16-8 - DOC for Competitor Aircraft

Aircraft	DOC ($/nm)
737-300	5.56
737-400	5.93
737-500	5.14
757-200	7.70
767-200	9.48
767-300	10.48

The operating cost for the commercial version of this aircraft shows that the DOC is competitive with current aircraft. Table 16-8 shows DOC for competitor aircraft.

16.3. Cost Effectiveness

One of the most important design drivers for this aircraft is cost effectiveness. The most accurate method of judging cost effectiveness is based on the life cycle cost (LCC) which is very difficult to accurately estimate for a new design. The flyaway costs were used to judge cost effectiveness for this design because it is a general indicator of the life cycle cost and it is more accurately estimated. The most important performance characteristics which define an aircraft's effectiveness are: the range, cruise speed, and the payload weight. Using these characteristics and the estimated flyaway cost an effectiveness parameter was defined:

$$Effectiveness = \frac{Range \;\; PayloadWeight \;\; CruiseMach}{FlyawayCost} \tag{16.1}$$

Using this parameter, Figure 16-1 was created which compares the effectiveness of existing aircraft with the Vulture.

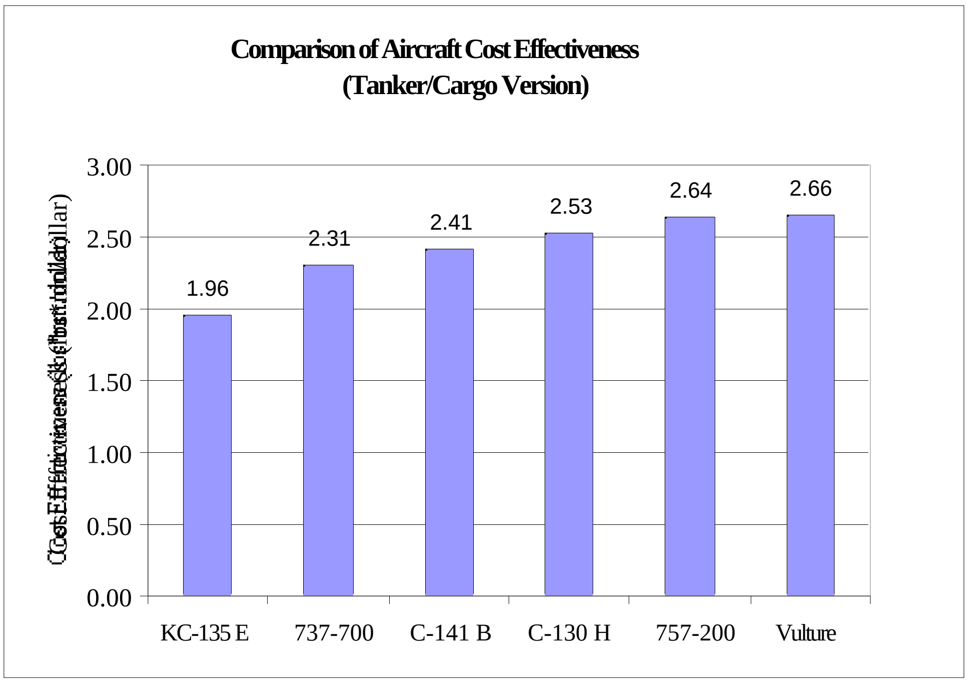


Figure 16-1 – Comparison of Aircraft Cost Effectiveness for Tanker/Cargo Version

Figure 16-1 shows that the Vulture is the most cost effective, based on flyaway cost, compared to its military and commercial competitors. The life cycle cost of the Vulture will be reduced compared it commercial and military competitors because it's strut-braced design allows more efficient cruising than a conventional cantilever wing which will reduce the fuel cost. With a production run of 600 aircraft, which perform many different roles, this airframe will become standard. This will reduce the maintenance cost of Vulture over other aircraft which are less prevalent. For these reasons the cost effectiveness of the aircraft based on the life cycle cost of the aircraft will favor the Vulture over conventional cantilever wing aircraft.

17. Conclusion

The Vulture's strut braced wing provides superior performance compared to a conventional cantilever wing. This increased efficiency decreases the operating cost of the aircraft, which makes it a cost effective solution for the RFP requirements. The Vulture also incorporates multirole capabilities which decreases the cost of both the CMC version and the tanker/cargo version over separate single use aircraft. The Vulture provides the best solution to the RFP requests.

Table 17-1 shows that the Vulture meets or exceeds all of the requirements set forth by the RFP.

Table 17-1 - RFP Requirement checklist

RFP Requirement	Requirement Met
Capable of transporting and launching 10 cruise missiles	Yes
Acquisition and life cycle cost kept as low as possible	Yes
Cruise Mach number no less than 0.85	Yes
Mission range of 5,000 nm	Yes
Military version capable of/for in-flight refueling	Yes
Minimum cruise ceiling of 30,000 ft	Yes
Landing distance should be equal to or less than 6,000 ft	Yes
Balanced field length of 8,000 ft on an ICAO std day	Yes

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