A GLOBAL RANGE HEAVY TRANSPORT FOR GLOBAL MOBILITY

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INTRODUCTION

The United State's role in the post cold war era is that of the only remaining super power. The United State's tremendous economic and military power will demand that she act as a Global Relief valve with the ability to diffuse regional conflicts and military aggression. The ability to do this will require the United States to maintain a capability to rapidly project significant power that leaves would be aggressors little doubt as to the outcome. In the future there may be frequent, small scale, unaligned conflicts, particularly in South America, Africa, parts of Europe and Asia. This situation is concerning in light of the United State's reduced ability to project and sustain power due to military budget cut-backs in the form of over-seas military base closures and troop draw downs. A heavy lift global range transport has the potential to provide real force projection while reducing the need for overseas military bases.

The cost involved in developing and fielding a heavy lift global range transport aircraft will require a thorough understanding of the aircraft's capabilities and limitations. The primary mission for this aircraft will be force projection, however, other applications such as combined aerial surveillance / air-to-air refueling, nation building and disaster relief are missions for which this aircraft could excel. A heavy lift global range transport aircraft will require significant resources, however, these may be acceptable when compared to the costs of maintaining foreign bases for force projection.

Commercial interests are unclear. For passenger purposes, the need for an aircraft of this capacity is not projected in the near future and may never be required due to smaller capacity long range transports that will provide frequent delivery to more direct routes. For commercial cargo purposes, a global economy may make an aircraft in this category very attractive. Business strategies, such as "just in time" inventory and global manufacturing, will demand more responsive transport capabilities than those offered by the current shipping industry. A heavy lift global range transport aircraft has a tremendous potential to reduce the current shipping delivery lag which is caused by limited capacity aircraft, slow response sea lift assets, existing port off load capabilities, long overland transport requirements and limited cargo dependent flexibility. A heavy lift global range transport could improve the delivery response by providing rapid, direct delivery, however, the start up costs are significant.

GENERAL

This report is based on the Catholic University of America's entry in the 1992/1993 AIAA / McDonnell Douglas Graduate Team Aircraft Design Competition. The scope of the competition was very broad, providing only general mission requirements and design guidelines for the conceptual design and justification of a large aircraft capable of projecting a significant military force. The focus of this report is the development and justification of specific mission requirements, the concept development of an aircraft that will satisfy those requirements and estimates for the resources required to obtain this aircraft. The report is formulated about two phases. Phase I analyzes the competition requirements, formulates specific mission requirements, investigates three aircraft concepts and determines specific aircraft performance requirements for the most effective global range transport. Phase II develops an optimal conceptual design based on the phase I selected concept, estimates required resources and analyzes the potential benefits of future aerospace technology advances.

COMPETITION REQUIREMENTS

The Aircraft Design Competition was initiated via Request For Proposals (RFP) for "A Global Range Transport For Global Mobility". The Opportunity Description stated that, "the RFP is for the concept design and justification of a large aircraft capable of so "projecting" a significant military force without reliance on surface transportation". Summarized below are the competition requirements and, where necessary, significant sub-requirements.

1. A large aircraft capable of so "projecting" a significant military force without reliance on surface transportation.

This requirement identified an aircraft with a large payload and cargo area volume. This requirement was considered to eliminate a conventional Wing In Ground effect (WIG) or powered WIG from being considered since a WIG is limited to low altitudes and, therefore, could not reach far inland sites of opportunity that are surrounded by mountains or other high terrain.

2. Careful consideration must be given to affordability and technology risk.

The CST shall utilize historical lessons learned and capitalize on technology that has been fully proven by flight testing.

3. Potential for preplanned product extension to meet corollary military and commercial missions.
4. A Technology Availability Date (TAD) of 2010.
5. A Initial Operational Capability (IOC) of 2015.
6. The aircraft must meet all applicable Mil-Specs.
7. Consider FAR part 25 requirements (projected to IOC date, including noise regulations) that would limit conversion to civil applications.
8. At minimum cost provide maximum delivery in 72 hours from the continental United States (CONUS) without exceeding one aircraft present at the destination airfield.

Military planners have identified a 72 hour response time so that a military force may respond to a crisis before the potential enemy has time to "dig-in". The aircraft must operate from CONUS so that our military can still project deterrent force after many foreign military bases have been closed. One aircraft present at the destination airfield reduces vulnerability by minimizing potential losses and by providing a rapid evacuation capability. Also, this requirement identifies the requirement to be capable of rapidly loading and off loading cargo.

9. A minimum unfueled range of 6000 nm.
10. The minimum payload shall be 400,000 lbs.
11. Capable of withstanding a 2.5g maneuver load factor at full payload.
12. The aircraft must operate from existing domestic airbases and use existing airbases or sites of opportunity at the destination.
13. The mission profile;
    a. Warm-up and taxi for 15 minutes
    b. Takeoff and climb to best cruise altitude with max payload
    c. Cruise at best altitude and Mach to midpoint
    d. Descend on course and land
    e. Taxi/idle for 30 minutes, off load full payload
    f. Load 15 % of full payload, takeoff and climb to best cruise altitude
    g. Return at best cruise altitude and Mach
    h. Loiter 15 minutes (15 min. reserve fuel)
    i. Descend, land and taxi 10 minutes

PHASE I CONCEPT EXPLORATION

TECHNOLOGY PROJECTION

Because the competition only specified minimum range and payload, it was necessary to find a method to project what the maximum takeoff weight might be at the Technology Availability Date (TAD). This estimate would then be used as a maximum criteria when the concept weights were estimated for aircraft with increasing capabilities. To estimate the state-of-the-art takeoff weight at the required TAD (2010), takeoff weights for previously fielded aircraft, Table 1, were plotted versus the aircraft availability dates and then curves were fitted to the data (FIGURE 1). This method of projecting a feasible maximum weight was considered valid because of the multitude of known and unknown variables, other than purely technical, which would affect the final decision to produce the aircraft. For instance, an aircraft design must not only to be technologically feasible, but it also must be capable of being produced at an acceptable cost relative to the need and be capable of operating from airfields other than dry lake beds.

PROJECTED TAKE-OFF WEIGHT AT TAD = 2010

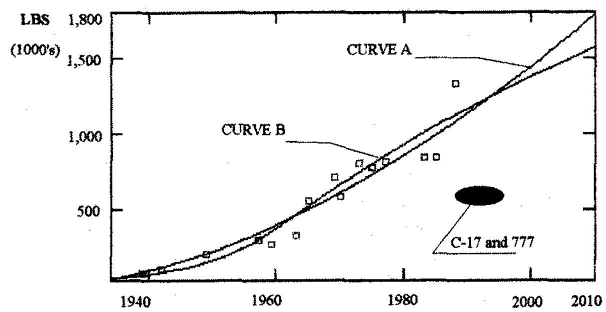


FIGURE 1

TABLE 1

AIRCRAFT	YEAR	TO WEIGHT(lbs)
1. JUNKERS JU.52	1934	23146
2. NAKAJIMA G5N1	1939	70768
3. MESSERSCHMIDT ME323D	1942	96050
4. DOUGLAS C-124C	1949	194500
5. DOUGLAS C-133A	1957	286000
6. BOEING 707	1959	258000
7. LOCKHEED C-141	1963	316600
8. ANTONOV AN-22	1965	551160
9. BOEING 747A	1969	710000
10. DC-10	1970	572000
11. BOEING 747C	1973	800000
12. LOCKHEED C-5	1975	769000
13. BOEING 747B	1977	810000
14. BOEING 747-300	1983	833000
15. LOCKHEED C-5B	1985	837000
16. ANTONOV AN-225	1988	1322750

Figure 1 includes two curves fit to the data and shows that maximum takeoff weights have been increasing with time in a relatively steady pattern. Curve A is the best case indicating a maximum takeoff weight of 1,800,000 lbs. This curve assumes that takeoff weights and aerospace technology will progress in a constant, predictable manner. Curve B is a step curve. The step curve gives a more conservative maximum takeoff weight estimate of 1,600,000 lbs. This curve suggests that increases in takeoff weight are a result of significant, identifiable advances in aerospace technology. For instance, the first step occurs in the late 1950's early 1960's. This step is probably the result of the tremendous data generated as a result of the post World War II technology boom and the development of the jet engine. The next noticeable step occurs in the mid 1960's and levels off in the late 1970's. This step may be rationalized due to events such as the rapid improvement of the jet engine, the space program and the development and increased use of the computer. The final step occurs as a result of the Antonov AN-225, which was available in 1988. This aircraft was cautiously included in this analysis, since, it is a one of a kind sample that was created by enlarging the AN-124 for carrying specialty cargo. Caution was exercised since, simply because the aircraft does exist, did not in itself justify the aircraft as a viable air transport. In fact, if the two most recent transport aircraft, the C-17 and 777 (expected to be the last all-new Western heavy transport to take flight this century [15]), were included in this graph, one might conclude that new aircraft development has hit a "maximum size wall" due to factors such as cost and support infrastructure barriers. These two aircraft are utilizing advances in technology to improve performance rather to push the size limit. In other words, one aircraft is hardly an indication of the start of a step or the magnitude of the step, however, it did provide a range of maximum takeoff weights to be investigated.

MISSION ANALYSIS

The RFP called for a global range aircraft that is available by the year 2015 at minimum cost and risk. The RFP definition of "Global" was only quantified by a minimum range of 6,000 nautical miles (nm). Obviously the longer the range the better, however, the selected range has to be consistent with the requirement to minimize cost and risks. The global mission will originate from CONUS and provide rapid delivery of both light and heavy contingency forces to geographic locations where U.S. military force projection has been reduced due to overseas base closures. Using a world globe and a metered string, ranges and mission radii required to provide force projection to potential worldwide flash points [13] were measured. It was immediately realized that to operate completely from CONUS would require the aircraft to have a mission radius capability over 7,000 nm. As will be seen later from the weight predictions, an aircraft with this capability would require much more advanced technology than that projected in the year 2010. The significance of this conclusion is that the aircraft will be required to incorporate staged missions or include air-to-air refueling equipment to have a global reach. From this point the investigation turned to finding the shortest range and mission radius required to perform the global mission.

The results of the investigation, as depicted in FIGURE 2, were that a 6,000 nm range capability and a 3,000 nm mission radius are the minimum distances acceptable for a Global Range Aircraft. The arrows in FIGURE 2 represent staging ranges at or below 6,000 nm, and the circles represent the reach possible with a 3,000 nm mission radius. As shown in FIGURE 2, these distances will allow staging in the Philippines, Turkey or the United Kingdom from CONUS, and provide sufficient mission radius capability to reach worldwide flash points and return to the staging base. This conclusion is also consistent with information found in the Strategic Plan for DoD Scientific and Technical Thrust 5, Advanced Land Combat report, which states that a realistic requirement for intertheater deployment is 3,300 nm.

GLOBAL COVERAGE

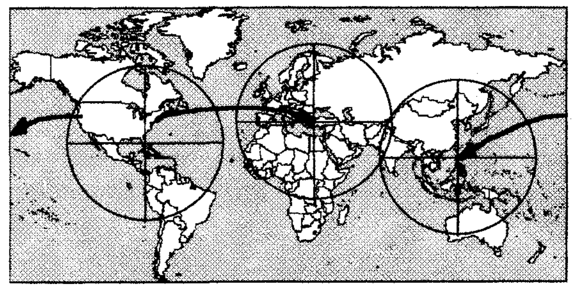


FIGURE 2

Under the assumption more is always better, mission radii of 4,000 and 5,000 nm were still investigated to quantify the ramifications.

CONCEPT DESIGNS

Concept designs of three aircraft categories were developed and competed to formulate the mission performance specifications for range, speed and payload that will maximize the amount of material that can be transported in 72 hours of continuous operation by a fleet of global transports based in the United States to any location in the world, while minimizing the cost of delivery. Aircraft concepts with medium subsonic speed, high subsonic speed, and one with supersonic speed were considered. These concepts were sufficiently developed to estimate the maximum takeoff weight, productivity and lifecycle costs. Performance parameters were selected by analyzing similar aircraft designs [5], from historical data [1,3,4] and engine data that was provided by AIAA. The performance parameters were conservatively selected to minimize the technological risks for meeting the Initial Operational Capability requirement. The parameters used and a description of each concept is described below.

Medium Subsonic The medium subsonic speed regime, M=0.6, was explored using a conventional, straight wing aircraft. This Mach number was considered the minimum accepted due to high vulnerability and slow response times. The aircraft has a high aspect ratio wing, AR=9, and is equipped with ultra high bypass ratio, BPR=20, turbofan engines. A cruise altitude of 17,000 ft was selected to maximize ground speed and provide the ability to clear a 15,000 ft mountain. Flying in this speed regime allowed this aircraft weight and cost advantages due mainly to the ability to fly with a straight wing. At the selected conditions, the engine specific fuel consumption (sfc), provided by AIAA, = 0.42 lb/lb/hr. A 10% benefit, compared to the C-5, was estimated for the empty weight ratio, We/Wo =.39, due to straight wing structural advantages. A target L/D of 18.5 was selected versus previous aircraft [1] due to improved aerodynamics and manufacturing capabilities and the high aspect ratio wing.

MEDIUM SUBSONIC CONCEPT

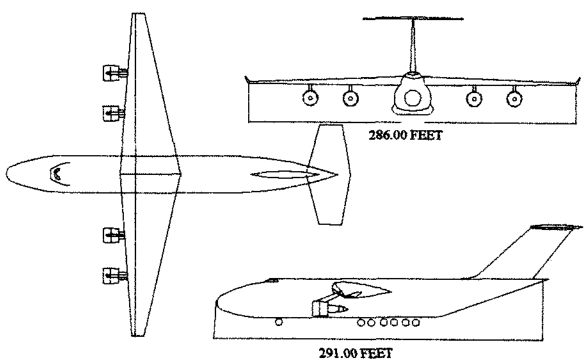


FIGURE 3

High Subsonic The High Subsonic speed regime, M=.85, was explored for this project by using a conventional, swept high wing configuration with a T-tail. The aircraft has a high aspect ratio wing, AR=9, and is equipped with ultra high bypass ratio, BPR=20, turbofan engines. A cruise altitude above 35,000 ft was selected to reduce drag and minimize fuel consumption. Advantages in developing this configuration were anticipated because of its similarity to current commercial and transport aircraft. At the selected conditions, the engine specific fuel consumption (sfc) is 0.48 lb/lb/hr. The empty weight ratio was estimated, based on C-5 data, and equals 0.43. A supercritical airfoil will be incorporated to achieve a reasonable L/D of 18.5.

HIGH SUBSONIC CONCEPT

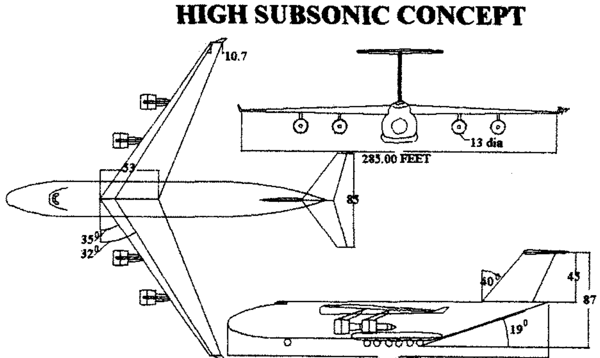


FIGURE 4

Supersonic The Supersonic speed regime was explored for this project by using an oblique wing, span loaded, Mach 2.5 aircraft. This high mach number was selected in an attempt to exploit Breguet's range equation. This configuration inherently reduced the wave drag without sacrificing payload capacity by incorporating the payload compartment into the wing, which can sweep from 0 to 70 degrees and eliminates the necessity for a fuselage. This unique concept allowed the wing thickness to cord ratio (t/c) to vary from .4, for low speeds, to .12 for high speeds, without complex mechanical sweeping systems. By incorporating the payload compartment into the wing the benefits from span loading are realized in reduced structural weight. This increases the manufacturability, decreases cost of materials and further reduces the wave drag by distributing the volume in a manner consistent with the Sears-Haack curve. At the selected conditions, the engine specific fuel consumption (sfc) is 1.2 lb/lb/hr. The empty weight ratio equals 0.4. The L/D is estimated to equal 10.

SUPERSONIC CONCEPT

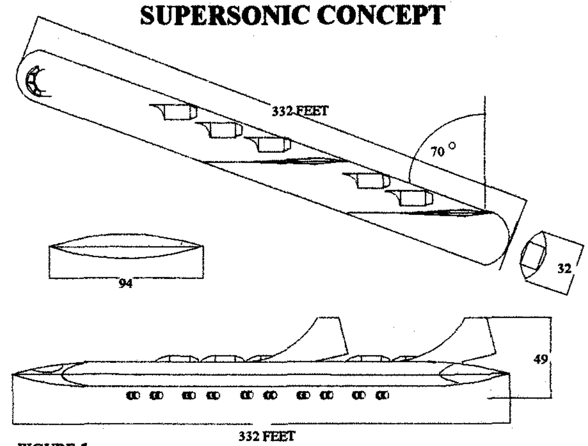


FIGURE 5

WEIGHT PREDICTIONS

Weight estimates for the three categories of aircraft were performed at mission profile radii of 3,000, 4,000 and 5,000 nm. For equal comparison, all aircraft were

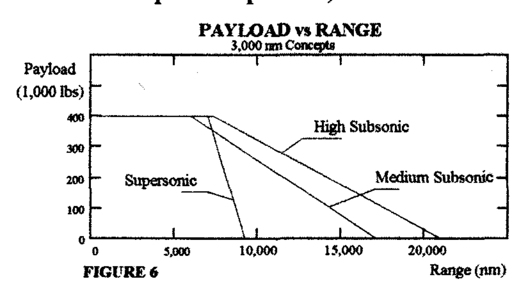
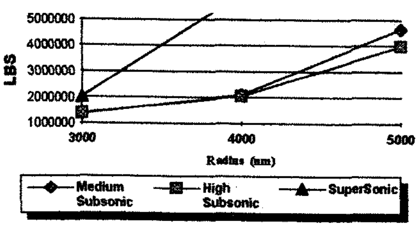


FIGURE 6

conceptually sized with equal payloads of 400,000 lbs. A takeoff weight estimating computer program was created for the mission profile that utilized Breguet's range equation and incorporated a 20% fuel reserve. The program allowed instant variation of the input parameters V_{cr}, L/D, sfc, Fuel Margin, We/Wo, Range and payload. The results are listed in Table 2 and plotted (Figure 7) for comparison of the three concept designs. Figure 7 demonstrates the advantage of higher speed for the long range requirement by showing that the medium subsonic concept, which has a 10 % structural weight advantage, is actually heavier than the high subsonic concept. It should be noted that in all cases, the fuel required for the mission radii condition was greater than that required for a range of 2 times the mission profile radius (the payload vs. range for the 3,000 nm

FIGURE 7



concepts are plotted in Figure 6). For example, the fuel required for the 3,000 nm mission radius was adequate to

TABLE 2

Radius	3000 nm	4000 nm	5000 nm
Medium Subsonic	1,366,563	2,115,306	4,657,316
High Subsonic	1,397,292	2,069,967	3,979,206
SuperSonic	2,037,115	5,999,387	92,810,216

provide a 6,000 nm straight through range with maximum payload. The results of the weight analysis indicate that the weight estimates for the 4,000 and 5,000 nm mission radii are significantly higher than the predicted maximum 2010 state-of-the-art maximum takeoff weight. Therefore, it was concluded that the CST had to have a maximum mission profile radius that is less than 4,000 nm to be available at the TAD.

PRODUCTIVITY

To determine the number of payloads that each class of aircraft could deliver and the number of aircraft required, a productivity analysis was performed. The aircraft were required to deliver 400,000 lb payloads at the destination radii, reload 60,000 lb (15%), and return without refueling as specified in the RFP. A maximum time of one hour was selected as a reasonable estimate for the time to load and fuel the aircraft and also for the time required to unload the cargo and then reload 15 % at the destination field. A limitation of one aircraft at the destination airfield was imposed by staggering takeoff by one hour (time at destination field). The results are listed in Table 3. As seen from the data, for the parameters selected, limiting the number of aircraft at the destination field strongly affects the size of the transport fleet and tends to equalize the productivity. For global force projection on two fronts, six fleets will be required. Two fleets will ferry payloads from CONUS to the staging field; and one will deliver them to the forward areas.

TABLE 3

Category	Med Subsonic			High Subsonic			Super Sonic		
Radii nm	3000	4000	5000	3000	4000	5000	3000	4000	5000
Sortie (min)	1125	1447	1769	895	1140	1384	405	487	568
Response Time	618	778	939	502	625	747	257	298	339
# of Payloads	62	59	56	64	62	60	68	67	66
# of AC	19	24	29	15	19	23	7	8	9

COST ESTIMATES

Lifecycle costs were estimated for each aircraft category. The total consist of the development, procurement and the operating and support costs. All costs were estimated using the Cost Estimating Relationships found in reference [1]. Using these CERs, a computer program was created which required the inputs; aircraft category, engine specific fuel consumption, maximum takeoff weight, number of aircraft, crew salary, maintenance labor rate, airframe weight, maximum speed, number of prototypes, and percent of composites. The total number of aircraft produced was based on having a six fleet operational capability. The rate of production was based on a five year production run. Four flight test aircraft were specified for each aircraft concept. The operating and support costs are based on 30 years of service. The costs are in constant 1993 dollars (<u>$ Billions</u>) and include a 10% profit. The results are listed in Tables 4,5,and 6.

Table 4 - 3,000 nm Mission Radius			
Category	Med	High	Super
#of AC	114	90	42
D&P costs	39.9	47.1	102
O&S costs	30.8	29.6	72.1
Lifecycle	70.7	76.8	174

Table 5 - 4,000 nm Mission Radius			
Category	Med	High	Super
#of AC	144	114	48
D&P costs	61.7	70.3	246
O&S costs	56.2	52.0	231
Lifecycle	118	122	477

Table 6 - 5,000 nm Mission Radius			
Category	Med	High	Super
#of AC	174	138	54
D&P costs	124	127	2,300
O&S costs	138	113	3,800
Lifecycle	263	240	6,000

SELECTION CRITERIA and SELECTED CONCEPT

Selection of the concept that would most satisfy the RFP was not based on any one performance capability or low cost. The RFP specified minimum range and payload capabilities that are required, and stressed affordability and technology risk. The technology projection and weight estimate analyses indicated that the medium and high subsonic concept designs could provide mission radius capabilities above 3,000 nm, however, the Mission analysis concluded that global projection, as possible at the specified technology availability date, does not require this. Therefore, a decision was made to discard the 4,000 nm and 5,000 nm mission radii concepts.

To assist in the evaluation of the three concepts a rating system was developed and employed. The primary factors involved are productivity, initial response time, technology risks, vulnerability, and life cycle cost. The initial response time was determined in the productivity analysis and is considered a separate factor because some National emergencies, such as coup attempts or humanitarian relief efforts, would require immediate response, rather than a three day build up, to defuse the situation . Vulnerability was considered a factor because the global transport will be exposed to radars, guns, guided missiles, exploding warheads and other enemy weapons while performing its mission. The primary characteristics which serve to reduce vulnerability for a global transport are altitude, speed, size and quantity.

Each evaluation factor was assigned a multiplier from 0 to 100, with the sum of the multipliers equal to 100. Each aircraft was then rated for each factor. The highest scoring concept received 100 points. The remaining two concepts received a corresponding percentage based on their score divided by the highest score. If the most favorable score for the evaluation factor was a minimum, then the lowest scoring concept received 100 points and the other concepts received corresponding percentages based on inverting their score. Table 7 shows the points for each factor and the total points each concept earned. The selected concept for the Catholic Super Transport is the High Subsonic platform with a 3,000 nm mission radius capability.

TABLE 7

Factor	Mult	Med Sub	High Sub	Super
Productivity	20	18.25	18.82	20
Response Time	15	6.25	7.67	15
Technical Risks	5	5	4.5	3
Vulnerability	10	5.55	10	10
Life Cycle Costs	50	50	46	20.3
Total Points		85.0	**88.0**	68.3

WING LOADING AND THRUST REQUIREMENTS

The final analysis of the Concept Exploration phase was performed to determine the wing loading and thrust requirements of the CST. For an aircraft of this size, a critical performance requirement is the field length from which it must operate. The RFP requires the CST to operate from existing domestic airbases and use existing airbases at the destination. According to the Design and Operations Division Office for the FAA, the most common airport size runway lengths are 5,000, 7,000 and 10,000 feet. Airfield environment specifications [10,11,14] state that Main Operating Bases (MOBs) and Deployment Operating Bases (DOBs) have runway lengths of 8,500 ft and Forward Operating Bases (FOBs) have runway lengths of 6,000 ft. Given that the CST is larger than any aircraft built to date, the goal was to ensure that the aircraft could operate from 10,000 ft, FAR 25 fieldlengths and MOBs at maximum weight conditions.

Takeoff, landing and cruise requirements were investigated to determine the thrust to weight ratio, wing loading and wing lift coefficients (c_l). Reasonable lift coefficients for transport aircraft ranging from 1.6 to 2.8 [3,4] were analyzed and 2.4 Triangle plots were created for five operating parameters which included:

1. FAR 25 takeoff requirements for Stof = 10,000 ft.
2. Landing curves.
3. FAR 25 climb requirements, OEI.
4. Mach 0.85 cruise requirement.
5. Four engine thrust to weight line.

The results of this analysis were that the CST required four engines (T/W =0.2857), a takeoff $c_{l_{max}}$ of 2.4 and a wing loading between 100 and 160 lb/sq. ft to satisfy all operating parameters. A takeoff $c_{l_{max}}$ = 2.4 was acceptable because it has been consistently achieved in similar aircraft without excessive or complex measures. To maximize range and minimize empty weight, a high wing loading of 155 lb/sq.ft was selected. Thrust is provided by ultra high bypass ratio, BPR=20, turbofan engines rated at 100,000 lb, for which data was supplied by AIAA.

Once the number of engines, takeoff $c_{l_{max}}$ and wing loading parameters were selected, the field length requirements were calculated for concrete runways for standard day conditions.

Sizing to Military Takeoff
Stog = 6,110 ft.
Sizing to FAR 25 Takeoff
Stofl = 9,973 ft.
Balanced Field Length
BFL = 7,052 ft.

Landing Field Length, FAR 25
Sfl = 8,977 ft (without thrust reversing)

The takeoff and landing conditions can be met for existing domestic airbases and sites of opportunity for the selected high subsonic speed configuration. To land at MOBs and DOBs with maximum payload, the CST will be required to have a thrust reversing capability. The CST will be capable of operating from FOBs at reduced operating weights.

SUMMARY OF REQUIREMENTS

The results of phase I have provided specific mission and aircraft performance requirements. Three aircraft categories were analyzed at three mission radii scenarios. The technology projection and mission profile analyses concluded that maximum takeoff weight limits will require the CST to perform staged missions to provide a Global force projection capability from CONUS. Further, the mission profile analysis concluded that global reach required the aircraft to have at least a 6,000 nm straight through range and a 3,000 nm mission profile radius, at maximum payload. The weight prediction analysis concluded that mission radii above 4,000 nm were beyond the state-of-the-art. The productivity analysis concluded that the one aircraft at the destination airfield limit determined the size of the transport fleet. Six fleets are required to provide a two front force projection capability. The selection criteria, which included productivity, response time, technical risks, vulnerability and Life Cycle Costs concluded that the high subsonic concept will provide the most effective CST. The wing loading and thrust requirements analysis concluded that the CST can operate from existing domestic airbases and sites of opportunity. A summary of the CST requirements are listed below.

1. Cruise at M = 0.85, at a 35,000 ft altitude
2. Maximum Payload 400,000 lbs
3. Minimum straight through range at maximum payload = 6,000 nm
4. Minimum Mission profile radius = 3,000 nm
5. Performance to operate from 10,000 ft FAR 25 fieldlengths, MOBs and DOBs.
6. FOB airfield compatibility
7. Capability to load 400,000 lbs of payload and maximum mission fuel in one hour.
8. Capability to offload full payload and reload 15% in less than one hour.
9. Be developed with proven technology that is available by the year 2010.
10. Deliver 90 aircraft (six fleets of 15) by the year 2015.

PHASE II CONCEPT DEVELOPMENT

CARGO SECTION SIZING

The Catholic Super Transport cargo volume and dimensions were determined by analyzing previous aircraft dimensions and dimensions of relevant military payloads. Considering that the CST can carry a maximum payload of 400,000 lbs, a large cargo volume is required for the large number of payload configurations envisioned. The C-5 and the Antonov -225 cargo capacities, 7.5 lb/cu ft and 9.2 lb/cu ft respectively, were analyzed to determine baseline estimates. The M-1 tank, Bradley Armored Vehicle, and the AH-1W helicopter were considered critical military assets for rapid response missions. The analysis concluded that the design payload would consists of two M-1 tanks, two Bradleys, four AH-1Ws and 88 troops. The required cargo area is 145.5 ft long, 24.5 ft wide, 15ft high. The cargo arrangement, shown in Figure 9, depicts the optimal loading pattern.

CARGO ARRANGEMENT

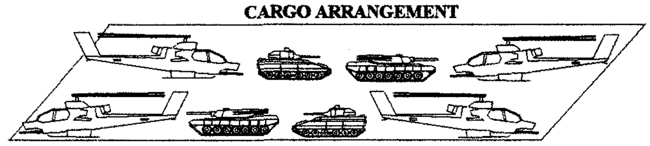


FIGURE 9

FUSELAGE SIZING

Once the cargo area requirement was determined the fuselage was dimensioned around it. The goals were to minimize wetted area and maintain a good aerodynamic shape that minimized weight, drag and production difficulty. To satisfy the requirement for rapid offloading in areas where ground support is minimal, forward and rear cargo openings and ramps were included. A double bubble fuselage cross section was selected to provide drive through loading and a simultaneous nominal troop transport capability while minimizing wetted area. Military cargo ramps must have ramp breakover angles less than 11 degrees. Therefore, an additional design criteria was to minimize the height of the cargo deck.

FUSELAGE CROSS SECTION

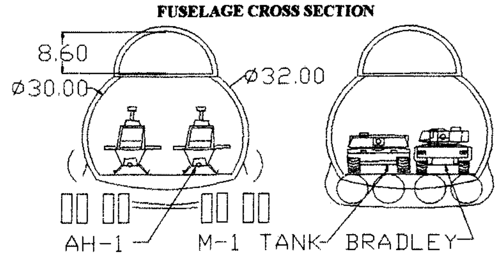


FIGURE 10

For each 1 foot that the cargo deck is lowered, the cargo ramps can be shortened by more than 5 feet. Since the cargo ramps are required to rapidly offload the M-1 tank, landing gear kneel and fold out adjustable ramp supports were included to reduce ramp structural weight.

Based on the cargo area requirements, a cargo inside diameter of 30 feet was selected, Figure 10. This diameter provided a final constant cargo area 25 ft wide, 16 ft high and 146 ft long. The cargo bay has a maximum height, 20.5 ft high, which will assure rapid offloading by eliminating the need for special dunnage for long awkward cargo, such as the AH-1, Figure 11. The final cargo bay payload density is less than 6.8 lb/cu ft, which will provide great flexibility in selecting mission payloads. The troop area inside diameter is 10 feet and oriented to provide a maximum head clearance greater than 8 feet. The bottom of the fuselage is clipped to minimize wetted area and to reduce the cargo deck height to 9 feet.

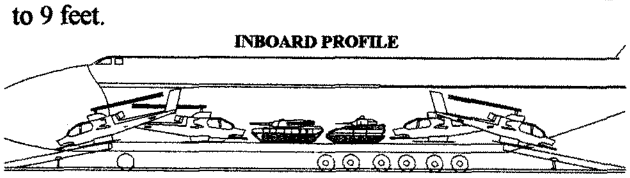


FIGURE 11

The CST nose cone has a length / diameter ratio of 1.5. To avoid large regions of boundary layer separation, the tail is smoothly tapered to a length three times the diameter of the fuselage[2]. The tail can have a maximum upsweep of 19 degrees for takeoff and landing rotation clearance.

LANDING GEAR

The number of wheels used on the landing gear was selected based on maximum landing weight so that the tire landing loads are similar to the C-5. The main landing gear are configured with six main legs each supporting eight wheels. The wheels retract into the fuselage fairing by swinging up and rotating 90 degrees The track width is 45 feet and the distance between the

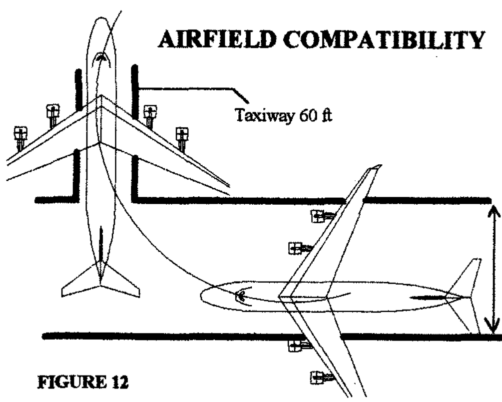


FIGURE 12

nose gear and the main gear turning point is 85 feet to meet the requirement to operate from FOBs. This configuration requires the rear legs to incorporate pneumatic lifting struts and the mid legs to rotate during taxiing to decrease the turning radius and reduce tire scuffing. As shown in Figure 12, this arrangement will allow the CST to turn off of a 150 ft runway onto a 60 ft taxiway (FOB condition) with minimal tire wear. Figure 12 also identifies the a requirement to have a reverse capability for turning the aircraft around. The thrust required for reverse is analyzed in the Aircraft Operational Performance section.

WING SIZING

To meet the required Lift to Drag ratio a high aspect ratio of 9 was required for the main wing. For the required wing loading of 155 lb/ft^2, the wing span was set at 285 feet. The wing has a taper ratio of 0.2 which sets the root cord at 52.8 ft and the tip cord at 10.6 ft. To enable cruise speeds at 0.85 Mach, the wing is swept 32 degrees at the quarter cord and has a supercritical 14 % airfoil. This arrangement provides an acceptable cantilever ratio of 23 [2].

A T tail wing configuration was selected to raise the horizontal tail clear of the main wing wash for buffet reduction. This configuration is advantageous, for very long range aircraft, to reduce fatigue on the aircraft structure and passengers by elevating the tail clear of the wing wash. The horizontal tail has a aspect ratio 5, taper ratio 0.3 and is swept 5 degrees more than the main wing to make the tail stall after the wing. The vertical tail has an aspect ratio of 1.2 and an acceptable taper ratio 0.63 [1] selected to equalize the vertical tail tip and horizontal root cords. The vertical tail is swept 40 degrees to assure a higher critical mach number than the main wing.

AIRCRAFT OPERATIONAL PERFORMANCE

As the concept design developed, more detailed design information was calculated and used to obtain more accurate performance capabilities. Sizing of the fuselage and wings allowed component weight and drag estimations to be calculated. This data was then used as input into several computer programs that were developed to analyze the mission profile, weight and balance, and payload versus range capabilities.

The component weights for the aircraft were estimated by the use of statistical equations based upon a sophisticated regression analysis [1]. The empty weight is 509,683 lbs, maximum fuel load is 396,000 lbs and the maximum payload is 400,000 lbs, which makes the maximum takeoff weight 1,306,533 lbs. Parasite drag estimates, Cdo cruise = 0.0211 & Cdo takeoff = 0.0782, were obtained by a component build up method [2] and related to the lift coefficients by the Oswald span efficiency method.

A detailed mission profile analysis was performed to determine the fuel required to perform a mission profile at a operational radius of 3,000 nm. The fuel required for the mission profile is 317,425 lbs. The fuel available at maximum takeoff weight is 396,000 lbs, which provides a 20 % fuel margin.

A weight and balance analysis was calculated to locate the aircraft Center of Gravity (C.G.) and determine the amount the C.G. would shift in operation. The individual weights were given horizontal and vertical stations and the aircraft was balanced to position the C.G. at 25 % of the Mean Aerodynamic Cord (MAC). The horizontal

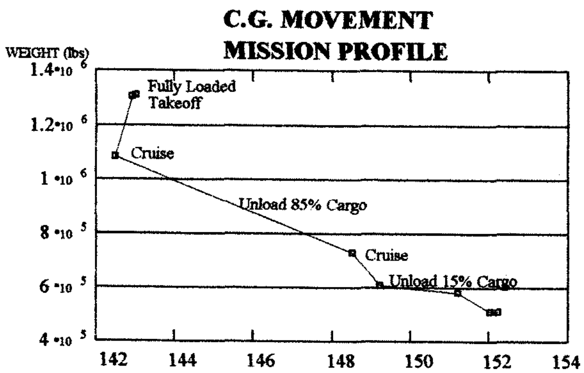


FIGURE 13

datum line is located 10 feet ahead of the nose of the aircraft, while the vertical datum is located 10 feet below ground level. The C.G. envelope was then determined for the mission profile (Figure 13). Due to the positioning of fuel and expendable cargo near the aircraft C.G., the C.G. location travels a distance of only 25 % of the MAC in the most extreme case.

Payload range curves were then plotted, Figure 14, to determine the range for a straight one way flight. The results are that the CST has a maximum range with full payload equal to 7,800 nm, with zero payload equal to 12,900 nm and 21,300 nm when the maximum cargo weight is substituted with fuel.

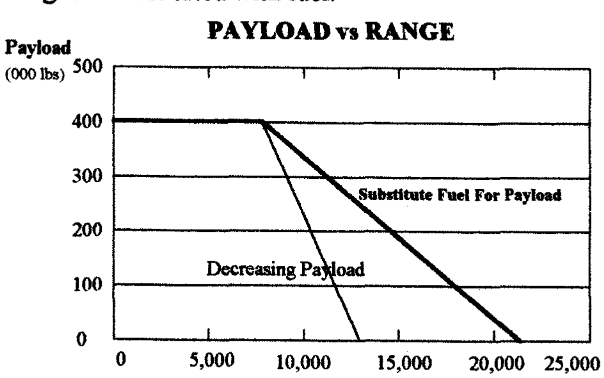


FIGURE 14

With the revised weight and drag estimates, the airfield takeoff and landing distances were recalculated. The revised weight calculation reduces the wing loading to 145 lbs/sq ft. Sizing to military takeoff rules, Stog = 5,715 ft. Sizing to FAR 25 takeoff, Stofl = 9,307 ft. The Balanced Field length equals 6,663 ft. Figure 12 identified the requirement to have reverse thrust for maneuvering the CST at FOBs. To provide a reverse capability over a 2% grade, the CST will require 52,400 lbs of reverse thrust. Having 52,400 lbs of reverse thrust enables the CST to land within 7,820 ft, according FAR 25 regulations. The flare and ground role distance is 4,691 ft. These distances will allow the CST to operate at full capacity from MOBs and DOBs and slightly reduced capacity (85,000 lbs less cargo or fuel) from FOBs.

FINAL COST ESTIMATE

A final cost analysis was performed for with the updated parameters of the CST. The development and procurement cost were calculated based on empty weight of 509,683 lbs, maximum speed = 490 knots, total number of aircraft produced = 94 (4 prototypes), 30 % composites, production rate = 1.5 aircraft per month, 4 turbofan engines rated at 100,000 lb thrust, avionics listing at $100,000 and a 10 % contractor profit. The operational and support costs were based on a max takeoff weight = 1,306,533 lbs, engine sfc = 0.48, lift to drag ratio = 18.5, 30 years of service, 3 man crews with an average salary = $75,000, and maintenance labor = $60 per hour. The results are as follows.

CATEGORY	HOURS	COST in 93 Dollars
Engineering	91,581,250	6,711,440,000
Development Support		802,084,300
Flight Test Operations		124,897,200
Tooling	52,149,610	3,925,197,000
Manufacturing Labor	139,859,400	8,688,624,000
Quality Control		1,328,925,000
Manufacturing Material and Equipment		5,653,550,000
Engines per Plane		26,375,180
Avionics per Plane		71,447

Total Investment required for Development = $11,563,620,000

Cost per Operational Plane $ 491,737,667

Total Contract Cost $ 44,256,390,000
Total O&S Cost $ 27,334,020,000

Lifecycle Cost $ 71,590,410,000

The cost estimating relationships utilized to predict the LCC are based on historical data and since the CST is significantly larger than any previous aircraft, the cost figures could include extrapolation errors. Some developmental cost realism is verified when compared to the estimated $10 Billion development cost for a very large commercial transport that would carry 550-800 passengers with a potential range of 7,000-10,000 nm [16] or the $4 Billion development cost for the 375 passenger, 4,000-5,000 nm range Boeing 777 [15]. Contract cost realism is offered by comparison with the $35.3 Billion contract cost for 120 C-17s [17]. Operating cost are based on an average annual usage of 1,400 hours per aircraft per year. Assuming that the aircraft is at cruise speed 75 % of that time, the operating cost of the CST is $19.67 per nm. This is low when compared with $32 per mile to operate a 747 [15].

TRADE STUDIES

Evolutionary improvements in aerospace technology will come at a premium price. Identifying those that would most significantly benefit the CST is the focus of this section. It is important to remember that the focus of this study was for a military aircraft contract. For the military to acquire this aircraft it would have to fund the development, testing, evaluation, procurement and O&S costs. Therefore, the results of this analysis are presented in terms of LCC and O&S costs.

From the cost analysis, the $71.6 Billion LCC is the sum of 16 % Development (includes testing), 46 % Procurement and 38% O&S. For a specified speed, payload and range, the development and procurement components of the LCC are mostly related to the aircraft weight. Technologies that improve the aerodynamics, structure and engine performance could all be applied to the CST for the purpose of reducing weight. When the aircraft weight is reduced, the complexity of the design and the manufacturing requirements are reduced and specific performance is increased, which reduces all of the LCC components.

To analyze the technologies that will reduce the LCC, performance characteristics that were specified by the RFP and those required by the mission profile; speed, payload, and range were held constant. The variables utilized are the empty weight ratio, lift to drag ratio, engine specific fuel consumption and maintenance requirements. These characteristics will be improved by advances in structures, aerodynamics, materials, aircraft controls, avionics, manufacturing capabilities and engineering analysis tools. Quantifying the advances is a matter of speculation, however, [9,18-23] indicate the following improvements in aerospace technology will be possible early next century.

Empty Weight. Advanced materials and the ability to design them with confidence in critical structures or at competitive costs will reduce the empty weight by 32%. Materials such as advanced aluminum alloys, titanium alloys, and metal-matrix composites will provide higher strength at lower weight for critical load carrying structure. Low cost composite manufacturing technologies, such as resin transfer molding, pultrusion tow placement, and compression molding will increase the components fabricated with light weight composite materials. Cocuring, consolidation, dual resin bonding and the use of textile preforms will reduce metal fasteners.

L/D Ratios. L/D ratios will be increased to values as high as 46 through advanced engineering design, and enabling manufacturing technologies. Advances in aerodynamic analysis capabilities will provide the ability to cost effectively develop optimized low drag designs. as well as, provide the capability to design devices to control flow characteristics. Aerodynamic efficiency will be increased by employing Active and Passive Laminar Flow Control (L/C) technologies. Examples include boundary layer suction and blowing, ribblets and low-drag laminar flow engine nacelles. Improved manufacturing technologies will enable production of clean surfaces that are aerodynamically optimized. Laser technology will improve quality control to hold tolerances under 0.001 of an inch.

Engine Specific Fuel Consumption. Engine specific fuel consumption will be reduced 40% by improved aerothermodynamics and higher temperature turbines. Improvements are well under way as a result of various armed services and NASA research programs. High temp materials, such as, titanium matrix composites, orthorhombic composites, ceramic matrix composites and intermetallics utilizing polycrystalline alumina fibers will allow future turbine engines to survive temperature ranges of 1,371-1650 C for 18,000 hours. That's longer than existing subsonic transport combustor liners.

Maintenance Costs. Maintenance costs will be reduced 25%. This improvement will be a result of improved maintainability design, automated inspection systems, and designs which reduce or eliminate sonic fatigue and corrosion problems. On board diagnostic computers will quickly identify malfunctioning equipment and optimize routine scheduled maintenance to reduce aircraft down time.

These improvements were then each separately introduced into a combined weight and cost estimating computer program to compare the relative reductions in LCC. The results are plotted in figure 15. As can be seen from the plot, the rate at which each technology characteristic reduced the LCC is the slope of the line. The plot shows that reducing the empty weight will provide the most rapid reduction in life cycle costs and the largest LCC reduction possibility. By reducing the empty weight 32%, the takeoff weight will decrease 36% and the LCC will decrease 37% (Point A). By reducing the engine specific fuel consumption 40%, the takeoff weight will decrease 28% and the LCC will decrease 27% (Point B). Increasing the Lift to Drag ratio to 46 will reduce the takeoff weight 37% and decrease the LCC 36% (Point C). Remember, these "LCC cost reductions" are estimated based on historical CERs and would only occur if the advanced technology existed and could be incorporated into the design for free. The actual reduction would depend on the resources required to implement the technology, above those historically required.

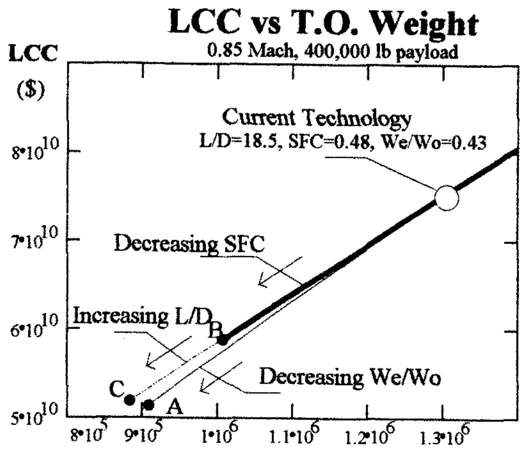


FIGURE 15 Takeoff Weight (lbs)

The operating and support cost were then broken out for comparison (Figure 16). The results are that the improvements in L/D will reduce the O&S costs by 47%. Reducing the empty weight ratio will reduce the O&S costs by 23% and reducing the specific fuel consumption will reduce the O&S costs 36%. Figure 16 demonstrates

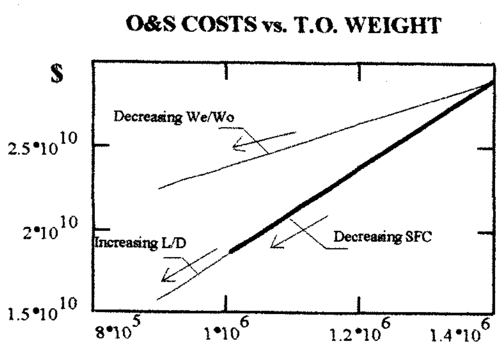


FIGURE 16

the double effect improvements in L/D and SFC have on O&S costs by reducing weight and improving performance. Again, the actual reductions realized would depend on the amount of any additional costs these technologies would require. For instance, the O&S costs reductions predicted by improving the L/D ratio, did not incorporate an increase in weight, specific fuel consumption or increased maintenance required for a Laminar Flow Control subsystem.

The goal of the analysis was not to quantify the actual dollar amounts each technology is worth, only to compare the relative benefits. The results of this analysis show that each technology has the potential to significantly reduce the cost of a CST, however, technology risks were not accounted for. For instance, composite technology is advancing at a rate slower than expected due factors such as maintenance costs and environmental hazards [16]. If the predicted technology improvements are correct, technologies which reduce the empty weight and improve the L/D will reduce the CST LCC the most.

The final calculation performed was to determine the LCC and takeoff weight reductions if all the advanced technologies could be synergistically incorporated into the CST. The results were that the takeoff weight would be reduced 52% and the LCC would decrease 51%.

DEVELOPMENT AND PRODUCTION REQUIREMENTS

The development, testing, evaluation and production of a project of this scale in an efficient and timely manner will require the expertise of an established airframe manufacture with highly efficient and responsive suppliers. Concurrent Engineering and Total Quality Management (TQM) practices will be required to assure that quality is defined and that all assets of the corporation are working towards achieving that goal. Automated design tools will be relied upon heavily for design, analysis and manufacturing efforts. These disciplines will be integrated through the use of internationally standard open systems, such as the Standard for the Exchange of Product model data (STEP) and Product Design Exchange using STEP (PDES), that will capture digitized geometric information and manufacturing production data that can be transferred to analysis software, manufacturing software, configuration control and other project functions with full data compatibility. Product testing and quality assurance problem solving teams should have active participation from members of all production teams as well as all the development engineering teams. A successful development, testing, evaluation and production program will be achieved by encouraging participation and cooperation in quality issues early and throughout the effort.

A manufacturing and assembly plant with one million square feet of floor space will be required for producing the CST at a rate of 1.5 aircraft per month. To assure quality and uniformity, the major components will be manufactured at this facility and then the entire aircraft will be assembled by moving through structured assembly stage locations from start to finish. Automation will be relied upon heavily to preclude design faults and minimize tolerances. Periodic inspections and check outs of critical part tools will be performed to ensure dimension control. Finally, tests will be conducted to ensure the assembly inspections have verified and satisfied the operational requirements in the assembly process.

Figure 17 shows the plan for implementing all stages of design and testing of the CST from the date of contract award to the beginning of full scale production. Approximately 7.5 years will elapse in this time making this a very quick and efficient life cycle. The concurrent engineering aspect of TQM makes this short life cycle possible by allowing prototype fabrication activities to begin before the detailed design is completed. The workforce required to accomplish this schedule is 2,440 Engineering employees, 1,806 Tooling employees and 6,000 Manufacturing employees.

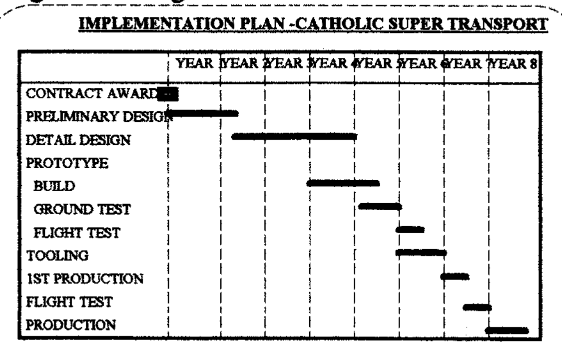


FIGURE 17

CONCLUSIONS

This report has reviewed the mission, operation, funding and schedule requirements for a possible heavy lift global range transport. The conclusion is that a conventional, high subsonic aircraft based on conservative technology projections will meet the minimum heavy lift global reach requirements as specified by the RFP. This report has identified major design considerations and develops a evolutionary conceptual design that meets those requirements. This conclusion is significant because development will not include the astronomical cost required to develop and prove a revolutionary new design. This conclusion is also significant in that the aircraft will "fit" existing world airfield infrastructure with the ability to operate from existing world airfields. Exotic concepts such as the span loader and multi body fuselage offer great promise, however, their ability to operate from existing world airfields is severely limited. I suspect that if the mission required a global reach from CONUS without staging, or, if the minimal payload was much higher, these exotic concepts would be required and their airfield constraints addressed.

The mission and operational analyses were performed under the constraints of the required 2010 Technology Availability Date. The technology available at this date will require a global range transport, with a minimum payload of 400,000 lbs, to perform staged missions from which it will deliver the payload to it's final destination. The minimum staging distance from CONUS is 6,000 nm and the minimum mission radius is 3,000 nm. The optimal design for this mission is a high subsonic, 0.85 Mach, conventional aircraft. Six fleets of 15 aircraft are required to provide a two front force projection capability. The aircraft will be required to operate from 10,000 ft FAR 25 field lengths and 8,500 ft military Main Operating Bases at maximum operating weights and from 6,000 ft Forward Operating Bases at reduced operating weights to provide maximum flexibility in selecting target destinations. The aircraft will be required to provide rapid cargo offloading in areas with minimal ground support and have reverse thrust capability for ground maneuverability.

The Catholic Super Transport meets all of these requirements. The CST has a cruise speed of 0.85 Mach, a straight through range of 7,800 nm and can perform the mission profile radius with a 20 % fuel reserve. The CST can operate at maximum operating weight from a 10,000 ft FAR 25 takeoff field length and a 8,500 ft military runway. Leaving 85,000 lbs of fuel or payload, the CST can operate from a 6,000 ft military runway. The CST has 52,400 lbs of reverse thrust for reverse capability over a 2% grade. The CST has a large, 6.8 lb/cu ft, cargo capacity and fore and aft ramps that will allow rapid unassisted loading and offloading of military cargo, such as the M-1 tank, Bradley fighting vehicle and AH-1 helicopter. The time required to deliver 90 aircraft, from contract award, is 12.5 years. Production of the CST will require a manufacturing and assembly plant with one million square feet of floor space. The cost to develop the CST is $11.5 billion. The cost of each CST is $492 million. The Operation and Support cost, for 90 aircraft over 30 years, is $27.3 Billion.

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