# “Cronus”
# A Response to the American Helicopter Society 2005 Request For Proposal
## Undergraduate Category

*Submitted for AERO2362 – Design Project*

*Commercial – In – Confidence*


***This report describes the conceptual design of a heavy-lift VTOL rotorcraft, intended for military use. Several systems are described in detail, and basic schematic drawings are included. This report is submitted both as an AHS 2005 Design Competition entry and as the major assessment for the RMIT AERO2362 course.***


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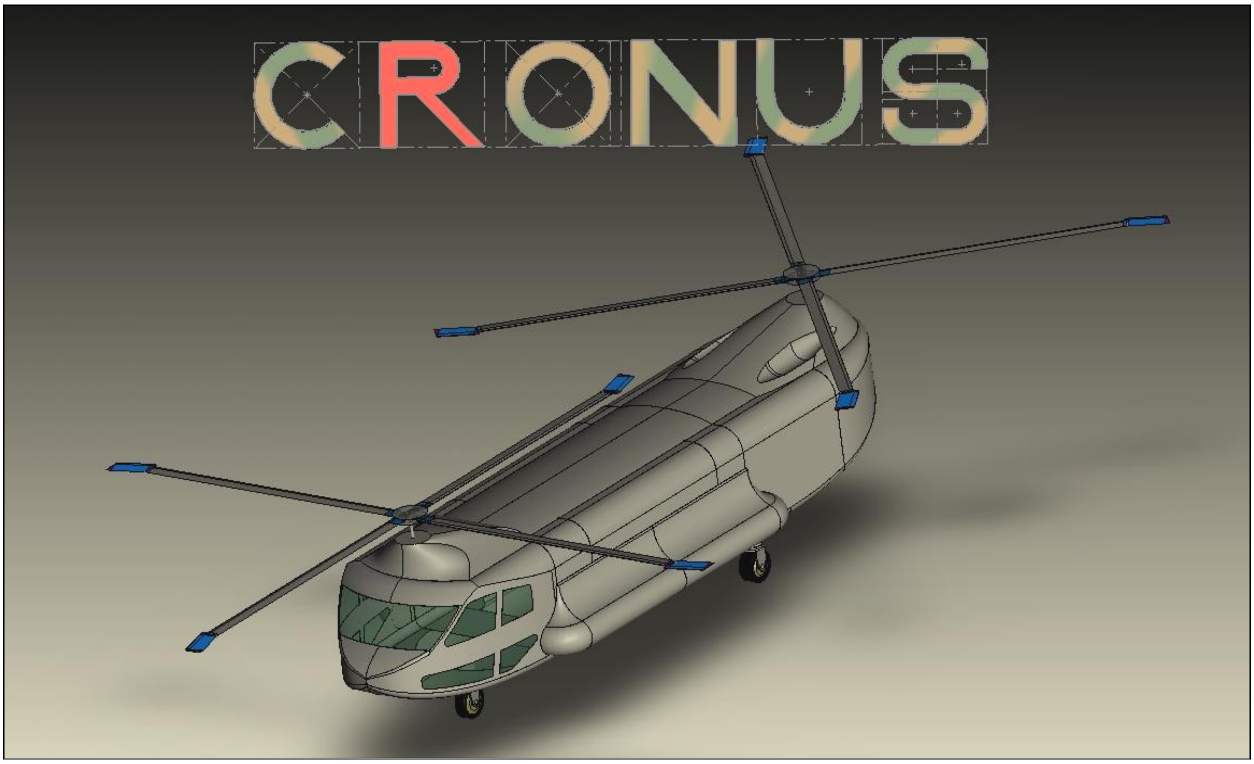
CRONUS

## *Executive Summary*

In response to the American Helicopter Society's 2005 Request For Proposal, an RMIT University (Melbourne, Australia) undergraduate design team has completed the conceptual design of a heavy-lift military helicopter, known as the "Cronus".

Trade-off studies of similar systems were conducted, and a configuration similar to that of the existing CH-47 Chinook was chosen. The tandem configuration is the most favourable for heavy-lift operations, when considering issues of stability, size constraints, and adaptability.

A nine stage design process, provided by Dr Arvind Sinha, was then followed. This iterative design method provides basic information, such as weight, power required, rotor dimensions, and so on. The values obtained were more than adequate for conceptual design.

These basic values were then used to carry out a more detailed performance analysis, which gave a refined engine weight, and allowed the maximum speed (governed by compressibility limits) to be determined. The performance of the rotorcraft was found to be satisfactory for the mission requirements.

A study of rotorcraft material selection was also carried out, with particular consideration for military requirements, such as ballistic protection. It was decided that composite materials would make up the majority of the structure, with certain exceptions, such as the undercarriage. The undercarriage and cargo handling systems were also analysed in detail. Other studies made include: vibration reduction and control; onboard survivability and observation systems; and designing for maintenance and reliability.

Finally, computer models were made of the entire structure, including the fuselage, cockpit, cargo bay, undercarriage, and rotors. Particular attention was paid to the design of the rotors, using experimental high-speed tips.

It was concluded that the overall conceptual design of the Cronus is feasible and worthwhile. The design process is expected to continue into next semester, and recommendations were made regarding future work possibilities.

## *Acknowledgements*

The student design team wishes to thank those individuals who have made recommendations, given advice, and have caused the project to proceed where it might otherwise have not.

- Doctor Arvind Sinha
- Associate Professor Peter Hoffmann

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# 1 INTRODUCTION

This submission is a response to the request for proposal (RFP) for the annual AHS International Student Design Competition (2005). The RFP concerns the conceptual design of a heavy lift VTOL vehicle, to be based on an existing naval ship, either the L-class or the larger CVN. The vehicle is to be integrated into evolving military operations, designed primarily to transport the US Army's Future Combat Systems (FCS) vehicles, which is a 20 ton maximum load.

Such an design does not yet exist, because of the many design challenges. Deck and elevator size is limited on naval ships, extreme levels of engine power and efficiency are required, and heavy lift rotorcraft tend to be slow and ungainly. These are some of the problems addressed in this report.

Despite these design difficulties, the successful implementation of military VTOL transport holds great benefits. A reliable rotorcraft will greatly increase the mobility and operational effectiveness of the armed forces, because it eliminates the necessity to appropriate a safe and usable runway in hostile or unfamiliar territory. This vehicle is designed to be compatible with the net-centric future combat environment.

It has been assumed that emerging technologies are available and reliable, as the first operational rotorcraft are scheduled for the year 2018. In particular, advanced materials and engine capabilities will be necessary. This is perfectly feasible, especially considering the military nature of the project, where cost concerns take a backseat to desire for performance.

It is anticipated that the design will lead to initial operation capability in 2018, and a subsequent fleet of 200 units will be delivered over a 15 year period. This high-technology combat supplement system, dubbed the "Cronus", is in keeping with the evolving attitude of the military; to provide a flexible, reliable and increasingly mobile combat force.

# 2 MISSION PROFILE

The exact mission profile requirements are listed below in their chronological order.

1. Rotorcraft is stored and maintained onboard the CVN-class carrier.
2. Rotorcraft is brought to the main deck on the elevator.
3. 10 minute warm up, including pre-flight checks.
4. Rotorcraft travels 75 nm to L-class assault ship, where FCS vehicles are kept.
5. Possible loiter (maximum 2 minutes) while waiting for clearance to land.
6. Rotorcraft lands on ship; FCS vehicle is brought on board and secured. FCS vehicle crew travel in the troop section of the rotorcraft (not in their vehicle).
7. Forward climb to 3000 ft altitude at best climb rate.
8. Maximum speed (130kts) to landing zone (at most 125 nm from the ship).
9. 15 minute loiter at 3000 ft near landing zone, wait for mission cue.
10. Descend to landing zone; FCS vehicle and crew disembark.
11. Return to L-class ship, flying at altitude if ground threats exist.
12. Possible loiter (maximum 2 minutes) while waiting for clearance to land.
13. Land with 20 minutes of loiter fuel reserve.
14. Simultaneously refuel and load next FCS vehicle and crew.
15. Repeat stages 7 to 14 until four vehicles have been delivered.

Figure 1 shows a visual summary of the mission.

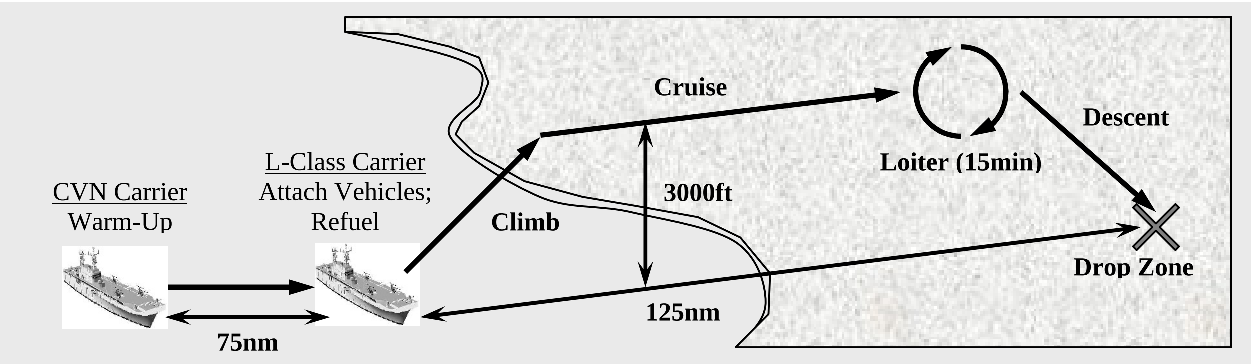


*Figure 1: Mission profile*

The successful completion of the mission will require and advanced integration of avionics, military tactical awareness, and rotorcraft performance. The primary measure of merit is the time taken to deliver four FCS vehicles to the landing zone, versus acquisition cost. An example of an FCS vehicle is shown in Figure 2.

*Figure 2: Possible FCS vehicle*

Particularly when in the vicinity of the landing zone, caution must be taken against the threat of hostile forces. Methods of reducing mission risk are detailed in Sections 4.1 and 4.2.

Note that the choice has been made for the Cronus to live onboard the larger CVN carrier, but operate from the L-class. This decision was made so that the maximum advantage was gained: living on the CVN means a greater rotorcraft size; operating from the L-class means that 150 nm of round-trip distance is eliminated for every vehicle deployment. Unfortunately, this means that L-class ships must always be paired with CVNs. The benefits of a shortened mission are considered worth the extra logistics.

# 3 TRADE-OFF STUDIES

When designing any helicopter, trade-off studies are necessary to determine the best possible configuration. In particular, the configuration of the rotor(s) and any anti-torque devices is a key design concept that should be discussed early in the process.

There are several types of rotor configurations that might be used for our heavy lift VTOL. As well as the standard main/tail rotor configuration, there are self-balancing rotor configurations which cancel out rotor torque effects. These include the following arrangements: tandem; side-by-side; coaxial; in combinations of 3 or 4; or even configurations which use tip-jet power.

Each type of configuration was investigated in order to choose the most suitable one for the proposed mission requirements. The most important design requirement is that the helicopter is capable of carrying a 20-tonne payload over intermediate distances.

## 3.1 Tandem Rotor Configuration

Tandem rotor configurations, shown in Figure 3, are mainly associated with large heavy lift helicopters. The tandem configuration consists of two rotors turning in opposite directions. The two rotors are driven by the same drive system, to ensure that they turn at the same and do not strike each other. The torque of each rotor is neutralised because of the opposite rotation of the other rotor. The control system is much more complicated, however, compared to a conventional helicopter configuration with a tail rotor.

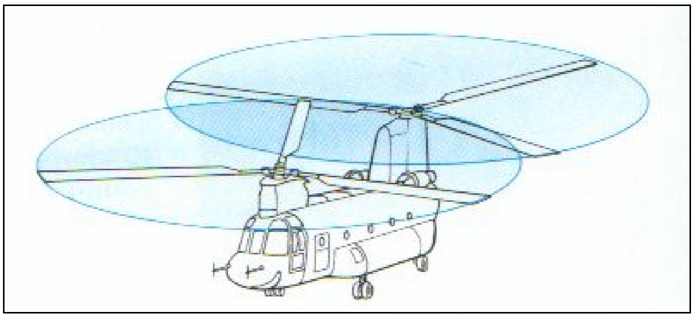

*Figure 3: Tandem rotor configuration (image courtesy of Jackson, 2001)*

The control along the vertical axis during hover flight is achieved by changing the differential collective pitch to change the relative thrust of the two rotors (bending the rotor discs against each other). A trim control for longitudinal cyclic pitch adjusts the tilt of the tip-path plane to reduce rotor flapping. This minimises oscillatory forces coming from the rotor hubs, as well as assuring fuselage-blade clearance.

The roll of the helicopter can be achieved by lateral cyclic pitch, tilting the two tip-path planes in the same direction to produce a rolling moment about the center of gravity. Directional control can be obtained by tilting the two tip planes in the opposite directions, which in turn produces a yawing moment. The use of tilt rotors gives the pilot plenty of flexibility when making turns (Prouty, 2002).

The tandem configuration is superior to the conventional configuration in terms of lift capability, because the tail rotor absorbs 10-20% of the engine power that would otherwise be used by the main rotor. Hence full engine power may be used to produce lift.

Another major advantage of using tandem rotors is that a lighter transmission system can be used. The smaller diameter rotors on the tandem turn at a higher rate than the single main rotor on the conventional design, with the same amount of lift capability. This means that less speed reduction is necessary between engine and rotor; hence a smaller gearbox is required (Gnaegi, 2005).

The conventional configuration also has trouble handling big shifts in the position of the centre of gravity (CG). The pilot can trim the helicopter in steady flight only when the CG is almost under the main rotor hub. The tandem design, however, has the natural capacity to unevenly divide the lifting chore between the aft and forward rotors, thereby handling a wider range of CG locations. A tandem configuration also makes the design process simpler, as only one rotor need be designed, and then used for the other rotor.

The main disadvantage of the tandem configuration is that it does not have the stabilizing benefit of a tail rotor; also, its aft end is not as far behind the CG, hence the fin effect of the aft pylon is comparatively weak. This means that marginal directional stability is a problem in tandem. Directional control is also sluggish because of the high moment of inertia about the vertical axis. This is due to the high weight at the ends of the fuselage. This lack in directional stability make the tandem configuration insensitive to the wind direction while landing on a ship (Prouty, 2002).

Another disadvantage is that the rotors are working on the same relatively small air mass and must impart more total energy to it to produce more lift, which results in an increase in induced power. A tandem configuration in sideward flight works with a larger mass of air, and requires less induced power than a conventional configuration. This can be used to obtain better climb performance by flying sideward at speeds below which fuselage drag begins to eat up the advantages. There are also vibration problems associated with tandem rotors.

## 3.2 Coaxial Rotor Configuration

This configuration, shown in Figure 4, consists of two main rotors mounted on a single axis. The main benefit of the coaxial configuration is that the two rotors rotate in opposite directions. This creates directional stability eliminating the need for an anti-torque device (tail rotor). This configuration is therefore not affected by crosswinds that can reduce a tail rotor’s effectiveness. It allows for allows for ultra-low level operations around obstacles. If a target is being tracked by a rocket or missile, higher efficiencies are achieved, due to the reduced level of correction needed from gusts during the tracking.

It should be noted that these rotors may not be the same size or even spin at the same speed, but they must both absorb the same torque. Advantages of the coaxial configuration includes the power saving due to no anti-torque device, and also having the overall dimensions defined only by the rotor diameter. There is an increased pressure differential over the rotor system – increased thrust; higher efficiency for increase in thrust; and reduction in rotor diameter for given thrust. The coaxial configuration is both dimensionally more compact and aerodynamically more efficient that that of a conventional rotor layout. There is also no dynamic roll and a high level of yaw authority (Zed, 2001).

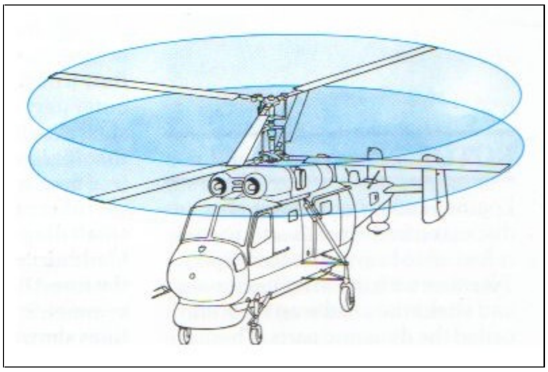

*Figure 4: Coaxial rotor configuration (image courtesy of Jackson, 2001)*

The disadvantages of this coaxial configuration include more complex rotor hubs and controls; and an increase of weight of rotors. Another disadvantage is that ample separation between the rotors must be ensured in order to prevent blades striking each other. This causes significant drag penalties and poor efficiency in high speed forward flight. There are also issues with maintenance, as access requires disassembly of large portions of main rotor gear and added complexity of the linkages required to operate pitching control of the rotor systems. The complexity and level of articulation of components results in high maintenance time and lubrication issues.

Other disadvantages include: inter-disc wash interference; increase in audio signature; alpha considerations for the lower disc; reduced efficiency of the lower disc; additional weight and complexity of main gear drive; and high running costs. Some flight restrictions include operation restrictions in high speed and certain maneuvering conditions, g-restriction for blade contact due to phase lag of differential rotor set-up, blade contact during RBS, issues of stability of main rotor shaft due to length, and much higher rotor drag (Zed, 2001).

## 3.3 Conventional Configuration (Tail Rotor)

This configuration, shown in Figure 5, is the most common, and is capable of carrying relatively heavy loads. It consists of one main rotor and an anti-torque device, usually a tail rotor.

*Figure 5: Conventional configuration (image courtesy of Jackson, 2001)*

The tail rotor has three main functions:

- To counteract the torque produced by the main rotor.
- To provide control for the vertical axis of the helicopter during hover.
- To provide directional stability.

This configuration has the advantage of being relatively simple since there is only one rotor, requiring one set of controls and one main transmission. The tail rotor in this setup uses approximately 8-10% of engine power in hover and 3-4% in forward flight. The simplicity of this configuration and the weight saving compared to other anti-torque devices compensate for the loss in power.

A distinct disadvantage when encompassing a tail rotor in the design is the danger to ground personell that is caused by the rotating blades. The main disadvantage of having a tail rotor for the Cronus is its fragility, making it a target.

These disadvantages, however, may be minimised by using alternative ways to counteract torque. The following anti-torque systems should be considered if the design team decides that a conventional aircraft is the most suitable design for the mission objectives (Prouty, 2002).

### 3.3.1 Fenstron

This type of anti-torque rotor is found on helicopters manufactured by Aerospatial and Eurocopter, and also the Sikorsky RAH Comanche helicopter. This anti-torque system uses a series of rotating blades that are located within the vertical tail. The blades rotate within a circular duct, making it less likely to come in contact with people or objects (ref 30).

### 3.3.2 NOTAR (No TAil Rotor)

MD developed this system, demonstrated in Figure 6. Helicopters use low-pressure air that is forced into the tail boom by a fan mounted at the bas of the tail boom. This air is then made to exit through horizontal slots located on the side of the tail boom. The pilot has control over the amount of air leaving through the slots in order to gain directional control and anti-torque.

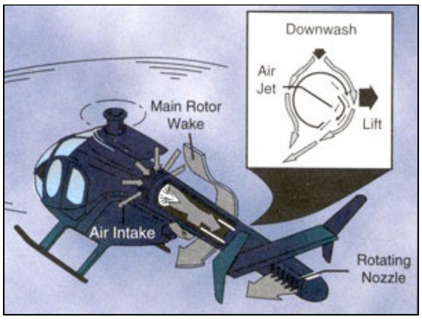


*Figure 6: NOTAR anti-torque system (image courtesy of HS, 2005)*

## 3.4 Side-By-Side Rotor Configuration

The main advantage of this configuration, shown in Figure 7, is the reduced power required to produce lift in forward flight. This advantage is quite relevant for the Cronus, since it becomes more effective in large multi-engine helicopters. This configuration also allows for considerable efficiency in high-speed flight.

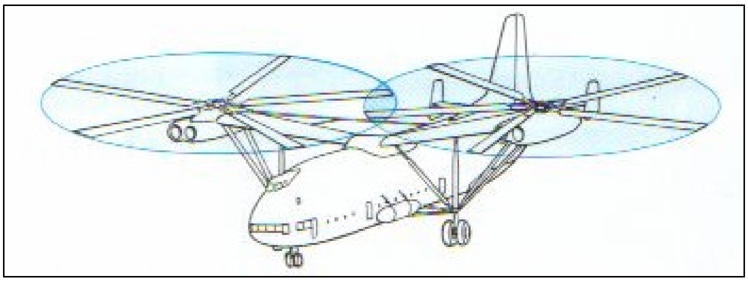

*Figure 7: Side-by-side rotor configuration (image courtesy of Jackson, 2001)*

The main disadvantage is that there is either high fuselage parasite drag or high structural weight. Also, when compared to the conventional configuration, the side-by-side configuration has the disadvantage of requiring relatively complex gearing and shafting. It should also be noted that this configuration has greater dimensions than the conventional configuration.

### 3.5 Synchropter Configuration

This configuration, shown in Figure 8, also uses side-by-side intermeshing rotors. The rotor shafts are close together, but arranged such that they are at a significant outward angle with overlapping rotors turning in opposite directions (meaning no anti-torque device is required). A gearing system ensures exact phasing of the rotors so that they do not strike each other. This symmetry reduces the demands on the pilot in coordinating the cyclic, collective and pedals. In addition, it results in a power to weight ratio that is approximately 15% better than the single rotor. This configuration was used in the first helicopter that demonstrated transition into autorotation and then back again into powered flight. Some other advantages include:

- Blade to ground clearance is equal to or greater than that of a comparable single-rotor craft.
- The lateral distance between the rotor hubs is decreased, which will reduce parasitic drag.
- A significant reduction of the pitch torque cross-coupling.
- The lift of both rotors is very close to vertical.
- The stability of the craft is enhanced.

*Figure 8: Synchropter configuration (image courtesy of Jackson, 2001)*

### *3.6 Tilt Rotor Configuration*

The wing tilts the rotors between airplane and helicopter modes and generates lift in the airplane mode. The conversion only takes a few seconds. This configuration, shown in Figure 9, uses a set of highly loaded main rotors to produce lift for vertical flight, the rotors then becoming propellers for cruise flight. No anti torque device is required as the rotors turn in opposite directions. The tilt rotor can take off, hover and land like a helicopter, making it more versatile than a conventional aircraft.

The main advantages of this system are that it has a greater high speed and a longer range than a conventional helicopter. It also has high cruise efficiency at high cruise speeds. In a helicopter the maximum forward speed is defined by the speed that the rotor turns at, however, with the tiltrotor this problem is avoided, because the rotors are perpendicular to the motion in the high-speed portions of the flight regime. This means the tiltrotor has a relatively high maximum speed.

*Figure 9: Tilt rotor configuration (image courtesy of TD, 2005)*

Tilt rotors require about 50% more installed power than a helicopter of the same lifting capacity, meaning that the tiltrotor requires larger engines for a given mission. This increases its cost and reduces the time it can hover, meaning tiltrotors carry about half the payload of typical helicopters. Other disadvantages are poor weight efficiency, no autorotation, and poor low speed efficiency and hovering capabilities (TD, 2005).

Furthermore, tiltrotors have a greater number of expensive components and structure than conventional helicopters. Their two rotors require all the fundamental parts of a twin rotor helicopter, a full set of airplane controls, and also a critical tilt mechanism that slews the lifting rotors (while carrying flight loads). This means that the typical tiltrotor has about three times the number of flight-critical components, adding to its cost and complexity (TD, 2005).

## 3.7 Final Trade-Off Study

The tandem configuration was chosen, as it is very effective in heavy lift helicopters. The tandem configuration is superior to the conventional configuration in terms of lift capability, as the tail rotor absorbs 10-20 % of the engine power that otherwise would be used by the main rotor. It also uses a lighter transmission system and smaller diameter rotors. Additionally, the tandem design has the natural capacity to unevenly divide the lifting chore between the aft and forward rotors, thereby handling wider range of CG locations. Using tandem makes the design process simpler, as only one rotor design is required, to be used for both.

The coaxial configuration was the second choice. The reason it wasn't used was because of the complex rotor hubs and controls, and an increase of rotor weight. Another disadvantage is that ample separation between the rotors must be ensured in order to prevent blades striking each other. The height limit of the Navy ship elevators meant that this was not a likely design.

The conventional configuration was not used because of the tail rotor. A tail rotor absorbs the power that is so valuable for a heavy lift helicopter, and it is not as sustainable as the other configurations because of its vulnerability. The synchropter was not chosen because of its complexity, and lower lift capacity (it is commonly used on small helicopters). The tiltrotor design was also dismissed as it requires about 50% more installed power as a more conventional helicopter of the same lifting capacity.

# 4 MISSION SYSTEMS

Mission systems pertains to systems that contribute directly to mission accomplishments. They are often of an avionic and optronic nature; the design of which for installation/operation may be on mechanical and avionics principles. Mission systems aboard this rotorcraft are the systems that provide capabilities in reconnaissance, surveillance and survivability. Included are systems that aid in observation and provide protection from adversaries. These systems can be classified as passive or active depending on their functional attributes (Sinha, 2004).

## *4.1 Survivability Systems*

### 4.1.1 Low Detection

When considering survivability, the goal is to fly undetected by any adversary, and to protect the crew and aircraft from enemy fire. Radar detection can be reduced using signature control, such as anti-reflective and radar absorbing materials (composites), as well as exhaust suppression. Exhaust ducts at the rear of the engines cool the hot exhaust gases, reducing the heat signature of the aircraft. The inclusion of these infrared suppressors comes with a loss of power (discussed in Section 7.1.3), but is necessary (Sinha, 2004).

### 4.1.2 Ballistic Protection

Survivability is also increased by the use of ballistic tolerant materials, which provides protection should a hit occur. Areas of vulnerability such as the rotor hub, fuel tanks, engine, transmission and the crew compartment are protected by armoured plating and by duplication of functionally critical systems.

### 4.1.3 Threat Detection

Threats are detected by (Sinha, 2004):

- **Radar warning receivers:** provide crew with data on hostile surveillance radar.
- **Laser warning receivers:** provide hostile laser range finders, illuminators and guidance systems.

- **Missile approach warners:** detect launch and approach of attacking missiles.

- **Point chemical detectors:** passively samples the air for the presence of chemical agents. A nuclear biological chemical (NBC) aircraft mask can be used to protect the crew from any such threat.

- **Infrared sensors:** detect invisible heat emissions that may be linked to threats.

- **Radar frequency jammer:** deny threat radar systems the information required to develop tracking and targeting information. They provide automatic or manual electronic jamming of radar threats. This system simply "outshouts" or masks the electronic echo from the aircraft.

- **Infrared jammer:** actively breaks the lock of infrared missiles. It projects modulated, infrared energy away from the helicopter. The projected energy is several times greater than the aircraft's heat signature. A secondary infrared source at an alternate position has the effect of confusing the missile's homing system as to the true location of the helicopter.

- **Chaff and flare dispenser:** deploys infrared and electromagnetic decoys. The chaff diverts the radar guided missiles and the flare protects the helicopter against infrared homing missiles. In the event of a heat seeking missile attack, chaff/flares will be automatically ejected from the rear of the helicopter.

- **IFF system:** assists automatic identification of the helicopter when interrogated by outside forces. Also permits a friendly helicopter to identify itself before approaching.

- **Laser radar:** provides navigational cues to avoid wire and obstacle strikes during nap of earth flying, at night, and in adverse weather conditions. Direct detection diode radar generates a very dense amount of light on a target. Pumped diode laser radar presents data as a video superimposed upon the pilot’s imagery.

## 4.2 Observation Systems

The observation system consists of a mission equipment suite, including navigation, sensors, communication gear, etc., suitable to perform flight operations in adverse weather conditions and night operations.

Because the mission must be satisfactorily and safely performed at any time and in all weather conditions, the continuos display of tactical and terrain information is required. Rather than individual sensors, multi-spectral capability is desirable for a greater operational advantage. The sensors include forward-looking infrared, thermal imager, millimetre wave radar and $CO_2$ laser radar. These sensors will be roof mounted, with a fully integrated electro-optical sighting system with visible and infrared capability. The information from these sights is sent to the flight crew to provide them with a comprehensive visual display for mission decisions (Sinha, 2004).

The various displays include the following:

- Head-up/head-down display, which displays attack, navigation and landing information.
- Head level display, which is used by the co-pilot.
- Multi-function display, which is located in front of each crew member, displaying flight information, mission status and subsystem monitoring.
- Integrated helmet display consists of helmet display and night vision goggles. The helmet mounted display is similar to the head-up display, except that the critical flight information is always in line of sight, and provides all information to the pilot required for flight. Night vision enables the pilot to fly head-up with the added comfort of night vision sensors.
- Mission Planning System (MPS) includes terrain following and avoidance radar, weather planning, collaborative planning, PGM planner, LO planning, portable flight planning system, which includes mapping applications that display various types of maps and geographically referenced overlays.
- Airborne warning and control system (AWACS).

# 5 DESIGN REQUIREMENTS

The preliminary design of the Cronus rotorcraft involves co-ordination with different disciplines, including aerodynamics, structures and systems. The primary objective of the preliminary design phase in the life cycle is to meet both the parametric requirements and constraints given.

In addition to Figures 10 and 11 in the next two sections, some other requirements are given in the RFP:

- Missile warning systems and countermeasures (RF & IR) must be included.
- Mission equipment suite suitable to perform flight operations in adverse weather conditions and night operations. These include navigation, sensors and communication gear.
- Basic aircraft maintenance must be facilitated. This includes access for inspection and rapid repair/replacement of all aircraft components (engines, transmissions, avionics, flight controls, hydraulic and electrical systems etc).

## 5.1 Design Parameters

The set of critical design parameters that govern the preliminary design are presented in Figure 10 below.

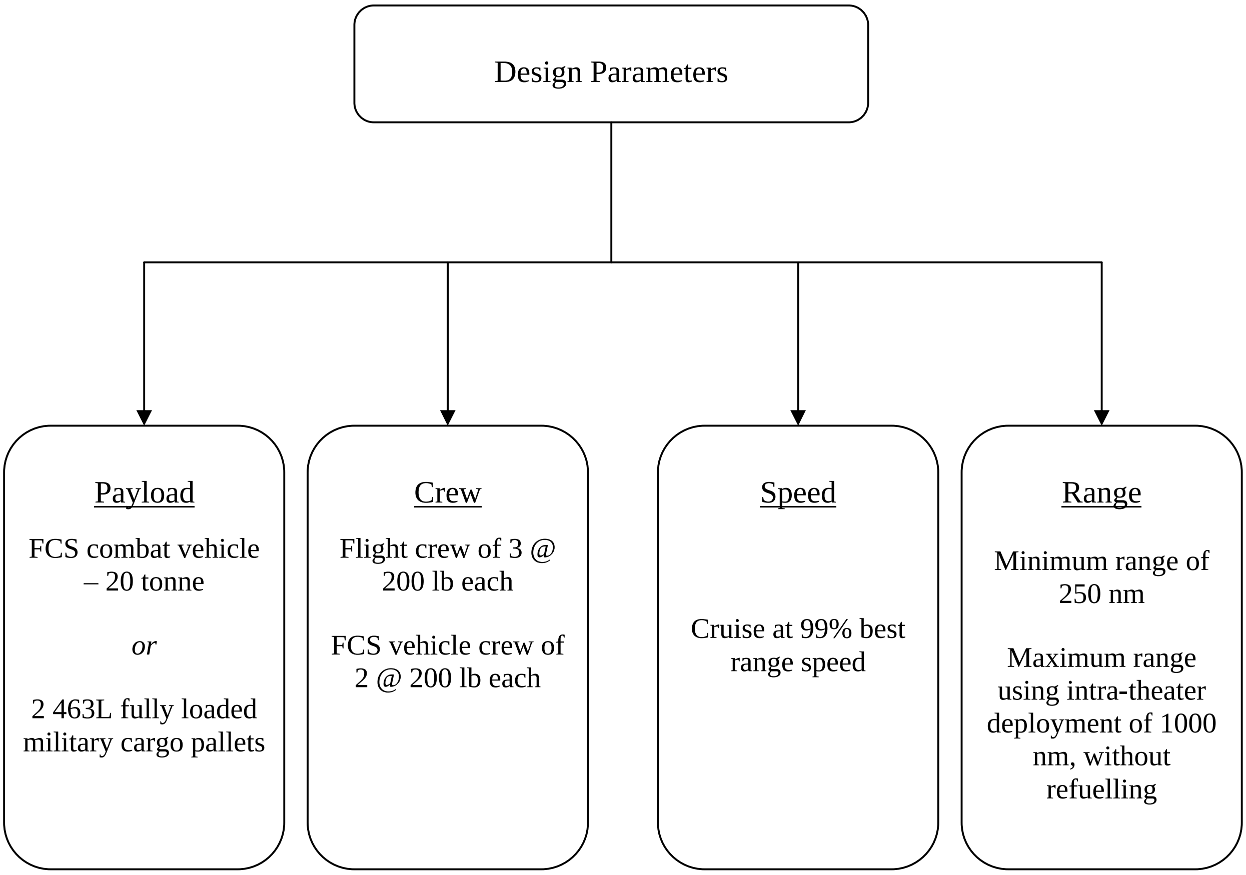


*Figure 10: Design parameters for the Cronus*

## 5.2 Design Constraints

We are confronted with several design constraints to meet the mission operational requirements. They are illustrated below in Figure 11.

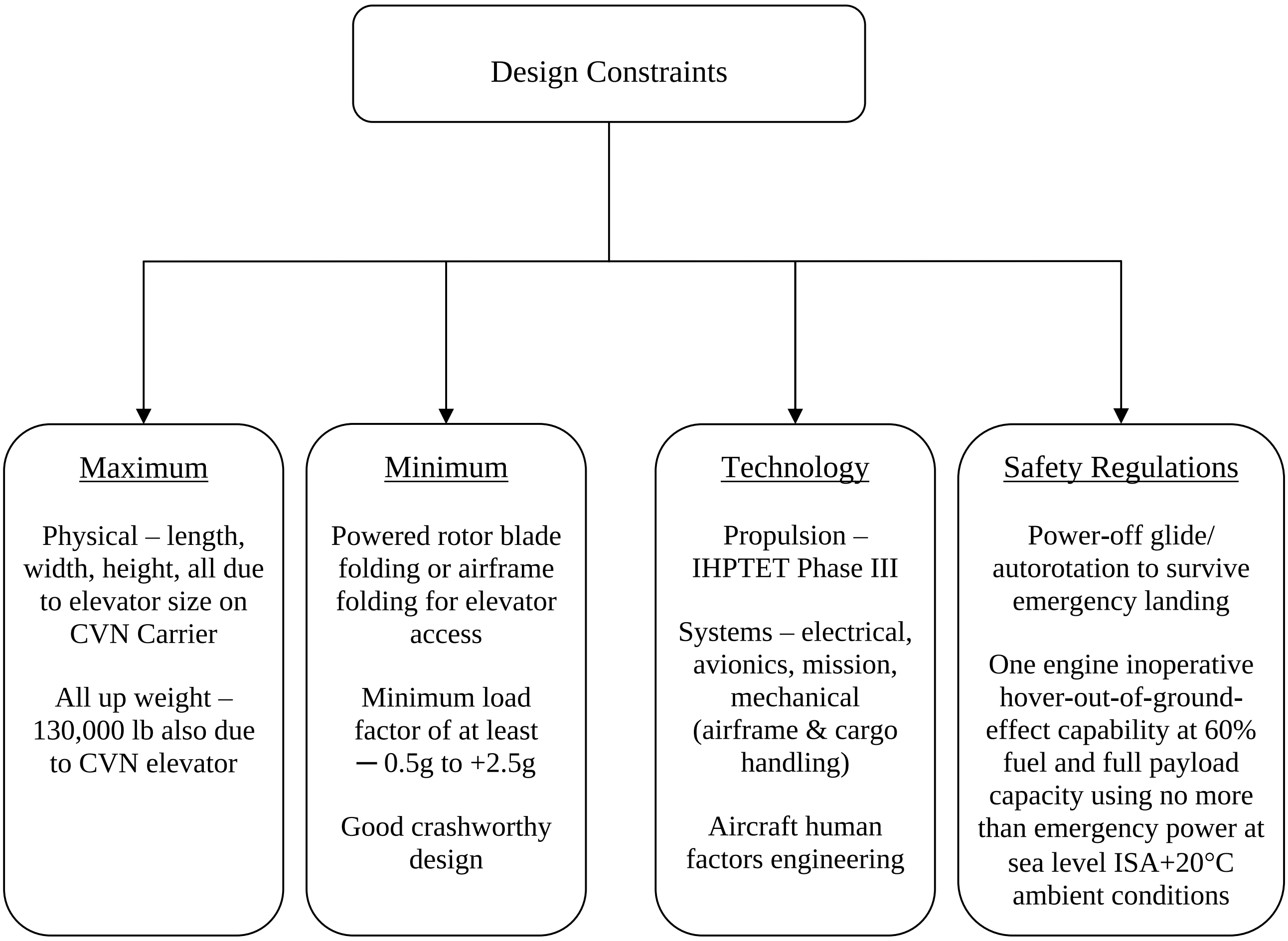


*Figure 11: Design constraints for the Cronus*

The physical limiting dimensions for overall folded length and height are given in the RFP and are summarised in Table 1.

*Table 1: Limiting Dimensions*

| **Dimension** | **Value** |
|---|---|
| Height | 7.62 metres (25 ft) |
| Width | 15.85 metres (52 ft) |
| Length | 25.91 metres (85 ft) |
| Weight | 59 tonnes (130000 lb) |

# 6 PRELIMINARY DESIGN PROCESS

The preliminary design process (Sinha , 2004) consists of several critical stages that are aimed at making best use of existing trends and limitations in helicopter design. This process is vital if a realistic preliminary design is to be developed in a time efficient manner.

This is a highly iterative method. The results of each stage are entered into a spreadsheet (not included with this report for the sake of brevity) such that iterations may be easily made and re-computation time minimised.

## 6.1 Stage 1 – Weight and Power

The design parameters given by the client (AHS) are studied and compared to similar rotorcraft. Gross weight and its basic breakdown provide the first estimation of weight. Power requirements are indicated by the engine rating of the helicopter. The primary power design case is for fully loaded hover at 3000 ft on an ISA +20°C day, but this is not considered for initial iterations. These values are benchmarks for later iterations.

## 6.2 Stage 2 – Fuel Estimation

A detailed mission profile is developed such that a conservative estimation of mission time may be determined. The weight of fuel required is determined from the time required to accomplish the mission and the specific fuel consumption rate (SFC) of the given engine selected in the previous stage. Note that the engine performance (including SFC) is a projected development of the IHPTET JTAGG program, Phase III (see Section 9.3).

## 6.3 Stage 3 – Weight

Although an initial estimation of weight was used previously, this was only to set benchmark values. These values are now refined by looking at the useful weight and factorising it to a gross weight based on chronological statistics (Prouty, 2002). Useful weight consists of crew, mission payload and fuel. Mission payload consists of the systems required onboard for mission accomplishment, such as armament, fire control, observation, navigation, and survivability. This has been determined by the mission requirements. Some of the systems that are to be included are detailed in Sections 4.1 (survivability systems), 4.2 (observation systems), and 9.2 (cargo handling systems).

## 6.4 Stage 4 – Vertical Drag

An estimation of vertical drag can be obtained based on disk loading and projected area of all components in the remote wake. However, it is beneficial to assume in preliminary design a vertical drag of 5% gross weight (4% is usual, but 5% was used to be conservative). This may be reconsidered at a later stage, as projected area is rarely constant during initial design stages.

Once a reasonable estimation of projected area has been developed, the high extreme of the allowable disk loading is used as the initial benchmark. Vertical drag is then calculated from these values.

## 6.5 Stage 5 – Main Rotor Design

The vertical drag evaluated in the last stage provides the basis to compute the size of the main rotor system as a first estimate for further iteration. The thrust of the rotor disk is estimated as the sum of the gross weight and vertical drag, therefore the disk area can be determined, giving a rotor diameter. At this stage, iterations may be necessary to adjust the gross weight, as rotor diameter may not be within allowable limits. In this case, the rotor diameter is sought to be minimalised, because of space constraints imposed by the dimensions of the CVN.

Constraints are also in place limiting allowable tip speed. In the case of the Cronus, allowable tip speeds were increased due to allowances in tip sweep for reducing critical Mach number at rotor tips (see Section 12.3). This maximum tip speed determines the allowable maximum forward velocity of the helicopter.

Noise levels and compressibility limits define the upper tip speed boundary, and stall and kinetic energy limits define the lower boundary. Higher kinetic energy values are required for autorotation.

Solidity and number of blades per rotor disk are determined from its survivability requirements. The final design choice was to have two rotor hubs, similar to the Chinook configuration, both with a diameter of 22 metres (72.18 ft). The rate of rotation was made to be 195 RPM, making the vehicle forward speed limit 130 kts.

## 6.6 Stage 6 – Drag in Forward Flight

This is an evaluation of parasite and profile drag. Parasite drag is produced from all non-lifting surfaces, with a large contribution coming from the rotor hubs. The rotor profile contributes to profile drag. In this process, these values are determined to establish the power requirement during forward flight.

### 6.6.1 Parasite Drag

Parasite drag is calculated for both the "clean helicopter" and "external loads". The parasite drag of the clean helicopter is derived from the gross weight. The coefficient of drag for external loads is expressed in terms of "form factor" and "coefficient of friction" and consists of 15% additional interference drag.

### 6.6.2 Profile Drag

Profile drag coefficient is evaluated from the "average angle of attack of the rotor blade" using three methods. Each of these results are considered and are averaged for the preliminary design stage. A value of 0.01 was chosen as a first iteration.

## 6.7 Stage 7 – Power Required for Forward Flight

The evaluated power required for maximum speed is compared with the installed power estimated at stage 3. If the installed power is low, engines can be resized or reselected to meet the power requirements. This also means a re-evaluation of the gross weight.

Power required for vertical rate of climb is also considered in this stage.

## 6.8 Stage 8 – Refinement of Gross Weight and Balancing

Iterations of the weights of the various onboard sub-systems must now be made. From this, appropriate locations of the sub-systems may be found such that it is appropriately balanced.

Following this, trade-off studies are conducted between the parameters addressed in the preliminary design phase.

### 6.8.1 Weight Definitions

- Basic weight: Includes all filled hydraulic and oil systems, trapped and unusable fuel, and fixed equipment.
- Operating weight: Includes the basic weight plus aircrew, the aircrew's baggage, loadmaster's equipment, emergency and other equipment that may be required.
- Total aircraft weight: Operating weight plus the takeoff fuel.
- Takeoff condition (gross weight): Includes the operating weight plus fuel, cargo, etc.
- Zero fuel weight: Operating weight plus payload.
- Landing condition (gross weight): Takeoff gross weight minus items expended during flight.

Increased weight hinders aircraft performance in the areas of: maneuverability; stability; hover performance; takeoff performance; rate of climb; operational ceiling; cruising speed; and range.

Weights are refined based on statistics, experience and judgment. There is a mathematical process through which equations are established. These equations are constantly updated in conjunction with the amount of composite material used and the current technological systems available. These equations are required to determine the weights of each sub-system and section of the rotorcraft.

### 6.8.2 Balancing

After the weights of the components are determined and refined, the balancing process is conducted. For the preliminary design process, calculations of the longitudinal position of the center of gravity (CG) is considered. When balancing, the ideal position for the CG is slightly forward midpoint between the two rotor shafts.

A datum line ahead of the helicopter is appointed and static moments are calculated about this line. The following equation is used for determining all components included in the mission systems and crew and group weights:

$$cg = \frac{\sum M_{gp} + \sum M_{comp}}{\sum W_{gp} + \sum W_{comp}}$$

Where

- cg center of gravity
- $M_{gp}$ moment of group weights
- $M_{comp}$ moment of crew mission systems
- $W_{gp}$ group weight
- $W_{comp}$ missions systems and crew weight

## *6.9 Stage 9 – Trade-Off Analysis*

This is an optimisation stage aimed at reducing the gross weight whilst staying within the design parameters. Component weights and drag estimates are updated and the profile both inboard and external are refined.

## *6.10 Summary of Preliminary Design Stages*

The basic helicopter information was determined through the use of spreadsheets, and is summarised in Table 2.

*Table 2: Preliminary design summary*

| **Attribute** | **Value** |
|---|---|
| All Up Weight (with FCS vehicle) | 41160 kg |
| Installed Total Power (Max. Continuous) | 8650 kW |
| Vertical Drag | 2058 kg |
| Rotor Diameter | 22 m |
| Rotor Chord Length | 0.47 m |
| Number of Blades on Each Hub | 4 |
| Blade Drag Coefficient | 0.01 |
| Fuselage Equivalent Flat Plate Area | 8 $m^2$ |

# 7 DETAILED PERFORMANCE ANALYSIS

The Cronus is designed to minimise the mission time; a single mission comprising the delivery of four FCS vehicles to the drop zone. This does not require a complicated performance analysis; the most unique aspect of the helicopter is its extreme fully loaded weight, in excess of 40 tonnes.

The performance analysis relies on two main factors: the drag values of the fuselage and rotors; and the engine performance at varying altitudes and temperatures. The mission must be deliverable under all reasonable ambient conditions, and taking into consideration appropriate installation losses.

## *7.1 Power Losses*

### 7.1.1 Engine

The Cronus is fitted with two scaleable IHPTET engines, which are discussed separately in Section 9.3. The advanced power-to-weight ratio of 1853.7 (Watts to Newtons, at uninstalled maximum continuous power setting) was held constant, and the engine weight varied until appropriate power levels were obtained.

The final required weight of each engine was 4200 N, or 428 kg (944 lb), the critical design case being fully loaded hover, using intermediate power, at 3000 ft on an ISA +20°C day. The power outputs for an uninstalled engine of such weight are given in Table 3.

*Table 3: Uninstalled Engine Settings*

| Engine Setting | Power Output | Time Limit |
|---|---|---|
| Maximum Continuous | 7914 kW | --- |
| Intermediate | 8658 kW | 30 min |
| Maximum | 9366 kW | 5 min |
| Contingency | 9786 kW | 2 min |

It was found that using 50% of the maximum continuous setting is sufficient for 130 kt cruise at 3000 ft on a +20°C day. Intermediate power is used for all climb regimes, and for hover at 3000ft. Maximum power is to be used for takeoff. Contingency power is only to be used when an emergency landing with a fully loaded aircraft is necessary, ie., when one engine fails.

### 7.1.2 Transmission

A possible configuration of the transmission was chosen, such that an accurate value for transmission losses could be determined. A formula given by Prouty (2002) was used, that assigns different levels of power loss for different gear types (planetary, bevel, and spur) and where they are in the drive train.

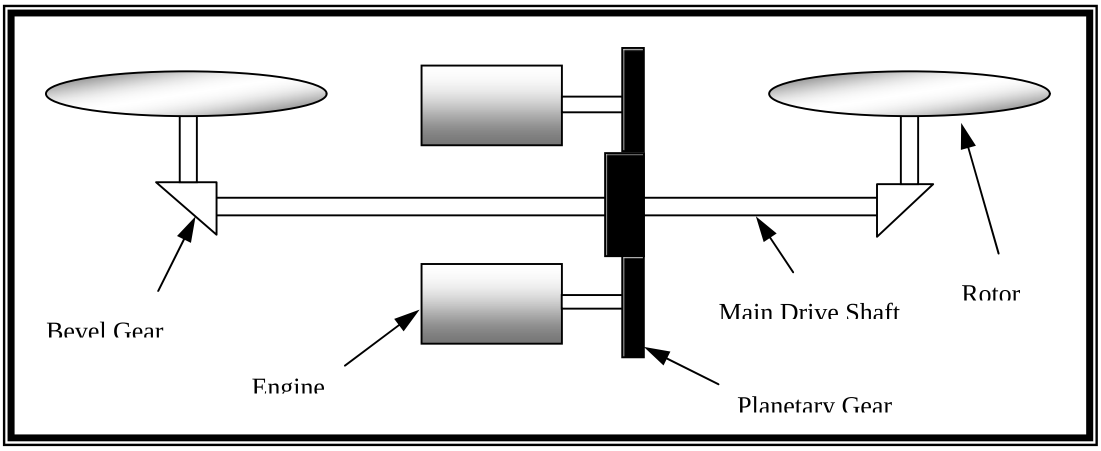


*Figure 12: Possible transmission configuration*

Without being concerned with details of gear ratios and other details, the configuration shown in Figure 12 was chosen. It included two planetary gear systems that operate at half the combined power, one planetary system that takes the power of both engines, and two bevels to each rotor that each operate at half the combined power.

Using such a configuration gives a maximum loss overall of 775kW for the engine weight selected. Although this maximum loss only occurs for contingency power setting, it was deducted from every power setting for a conservative design.

### 7.1.3 Additional Installation Losses

The installed engines will also experience a reduction in power due to accessories, intakes, and exhausts. Although such losses are to be minimised, they are unavoidable. Prouty (2002) was used to approximate the percentage losses from various sources. These are shown below in Table 4.

*Table 4: Installation Losses*

| Source | Percentage Loss |
|---|---|
| Inlet duct friction | 2% |
| Particle separator | 6% |
| Exhaust friction causing back pressure | 1% |
| Infrared suppressor back pressure | 8% |
| Compressor bleed and accessories | 8% |
| Total | 25% |

## *7.2 Mission Environment Limits*

Analysis has been made for a standard ISA day, as well as a +20°C day. The cruise altitude has been stipulated in the RFP as being 3000 ft (914 m), which means the ambient density never drops below 1 kg/m$^3$, even for the +20°C day. Ambient air density is a primary factor in engine performance, power to hover, drag, and other characteristics.

Maximum speed for this military aircraft is more important than best cruise speed, which might be selected on a civilian aircraft for lowest operating cost per air-seat-mile. Efficiency is not so much desirable as shortest time to mission completion. The maximum speed that the Cronus is capable of has a definite limit, resulting from the maximum allowable tip speed. Because provision has been made for a lowered Mach number at the rotor tips (see Section 12.3.1), the maximum speed of the aircraft is 130 kts (66.9 m/s), with the advancing blade tip having a total airspeed of 291.9 m/s. On an ISA day at cruise altitude, this pertains to an acceptable Mach number of 0.819, when the effect of the swept blade tips is accounted for.

The mission limits pertinent to performance are summarised in Table 5.

*Table 5: Relevant Mission Limits*

| **Mission Environment Factor** | **Limit** |
|---|---|
| Forward Speed | 130 kts (66.9 m/s) |
| Altitude | 3000 ft (914 m) |
| Temperature | 35°C (95°F) |
| Ambient Air Density | 1.047 kg/m$^3$ (0.00203 slugs/ft$^3$) |

### 7.2.1 Losses Due to Environment

Environment losses come in the form of reduced engine capability. Greater altitude and greater ambient temperature result in thinner air, which directly affects the thrust output. Both altitude and temperature are made to modify the ambient density, and this density drop is in turn made to linearly reduce the engine power.

Empirical data given in graphical form on page 274 of Prouty (2002) gives the following approximations:

- At sea level, a +20°C day reduces all engine outputs by approximately 15%.
- On an ISA day, operating at 3000 ft reduces all engine outputs by approximately 6%.
- The combined density reduction of +20°C at 3000 ft gives a total output reduction of 20%.

## 7.3 Estimation of Fuselage Drag

The graph on page 130 of Done & Balmford (2001) and reproduced in Hoffmann (2004) was used as an empirical means of estimating the equivalent flat plate area. In terms of configuration, the Cronus is similar to the CH-47 Chinook. However, the CH-47 weighs approximately 145 kN, compared to an enormous all up weight of 404 kN for the Cronus. Assuming, as per the mentioned graph, that weight is related to the equivalent flat plate area, a flat plate area of $8m^2$ would not be unreasonable.

In any case, a conservative design is desirable, particularly when noting the open-body design of the Cronus. Although great care will be taken to design aerodynamic fairings, hubs, and pylons (the main causes of drag), some interference and boat-tail drag will be experienced at the cargo section.

For many helicopters, the profile drag of the body is sought to be minimised because it is the chief limit on the maximum attainable forward speed. Profile drag increases as a cubic with increasing forward speed, and so becomes the dominant power requirement. However, the maximum speed of this particular aircraft is limited by the compressibility effects at the rotor tips before parasite drag becomes a serious problem. For this reason, only minimal attention was paid to equivalent flat plate area estimation.

## *7.4 Power Required to Hover*

An empirical formula given by Hoffmann (2004) gives the conservative result for power required to hover. At first, the basic equation relying entirely on actuator disk theory was used, but the empirical formula gives a result that is at least 30% higher. A vertical drag of 5% of the gross weight was used. The torque coefficient for hover is:

$$C_Q = \sigma C_d/8 + 1.13 C_T \sqrt{(C_T/2)}$$

Total power required for hover is then:

$$P_{req} = C_Q \rho A \Omega^3 R^3$$

All symbols have their usual American definitions. The variation of power required to hover as altitude varies is shown in Figure 13. Results are given for a standard and +20°C day. Clearly the design case for the engine (resulting in the engine weight chosen in Section 7.1.1) is hover at 3000 ft on a +20°C day. This occurs during the flight phase of hover OGE at 3000 ft while waiting for mission cue, as stated in the RFP. It is assumed that intermediate power may be used during this brief hover segment.

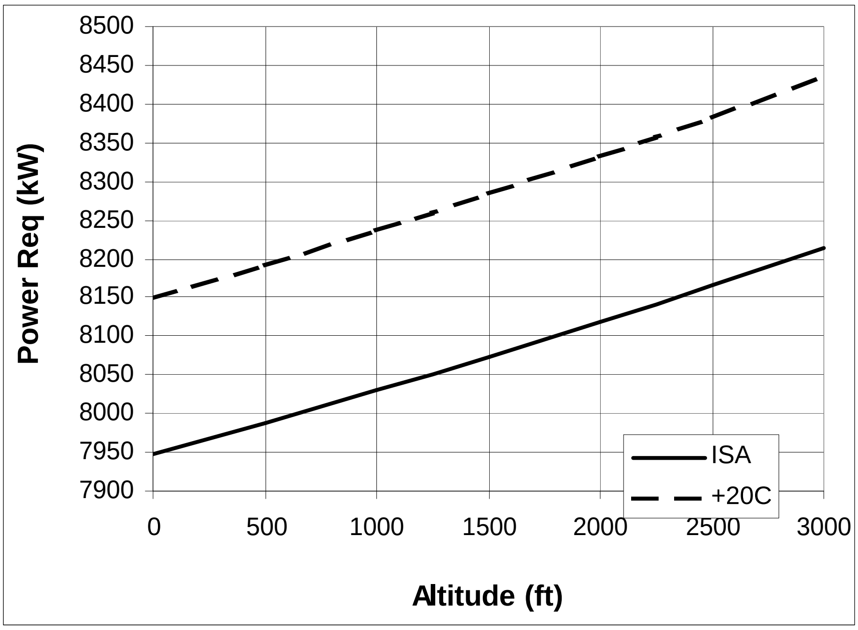


*Figure 13: Power Required to Hover*

## 7.5 Power Required for Forward Velocity

The power required for forward velocity is given by several sources, all breaking down the power requirement into components. The presence of a tail rotor would normally add approximately 6% to the total power requirement, but this is excluded due to the tandem rotor configuration of the Cronus. The other contributions are blade profile drag power (including H-force); fuselage parasite drag power; and induced drag power. Combining these three, having already subtracted the installation and environment losses, gives the total power curve. Figure 14 shows the power contributions for forward flight at sea level, ISA conditions. The empirical formula used is only applicable for forward speeds greater than the induced velocity (approximately 16 m/s (31 kts)). Below this speed, the total power curve is simply made to tend toward the power required for hover (zero forward speed), which was calculated in the previous section.

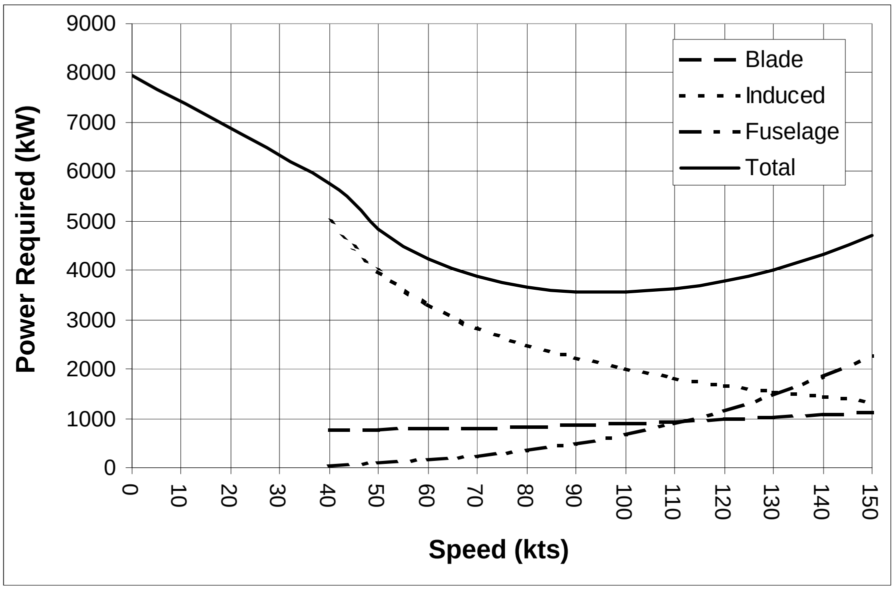


*Figure 14: Components of Power Varied with Forward Speed (Sea Level, ISA)*

The benefit of the total power curve is the ability to pinpoint the forward flight speed requiring the least power, at various altitudes and temperatures. This speed is then the best speed for climb, because there is a maximum excess of power.

Charts are included here for 3000 ft and 0 ft for standard and +20°C days. Total power is shown, without the breakdown of components. Figure 15 pertains to sea level, and Figure 16 to the 3000 ft cruise altitude.

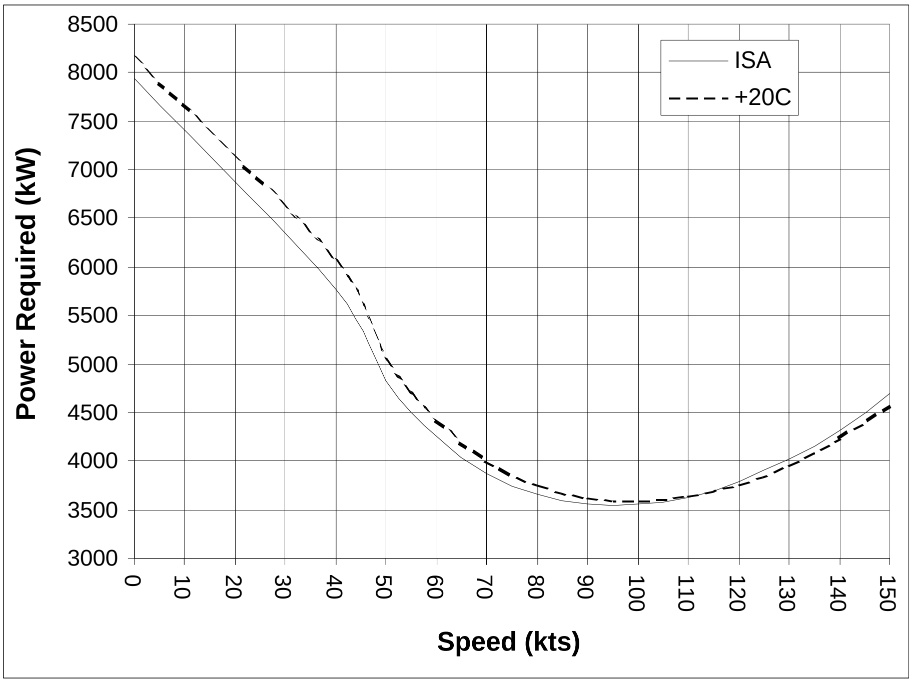


*Figure 15: Variation of Power with Forward Speed (Sea Level)*

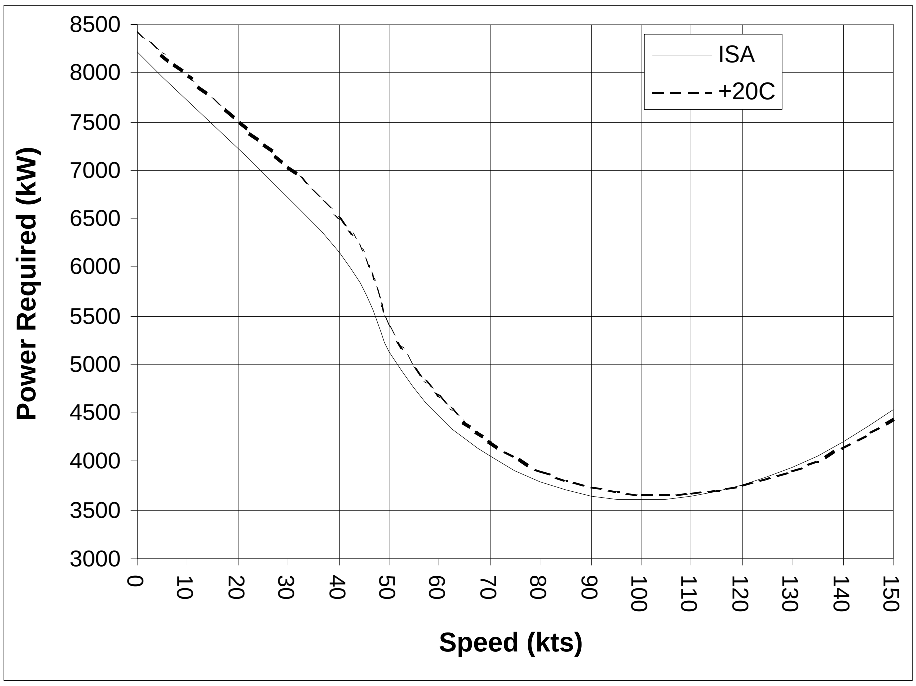


*Figure 16: Variation of Power with Forward Speed (3000 ft)*

### 7.6 Power Required for Forward Climb

A simple formula was used to determine the best angle of climb for each flight regime, at various altitudes and for standard and +20°C days. It was found that the maximum rate of climb is beyond the 130 kts forward speed limit, and so is not covered in any detail. The American version of a formula given by Done & Balmford (2001) gives the power necessary to climb as a contribution of the dimensionless torque coefficient:

$$C_{Q(climb)} = \lambda_c C_T$$

The coefficient $\lambda_c$, the climb inflow ratio, may be approximated to the sine of the climb angle.

Climb is assumed to use the intermediate engine power setting, which has a time limit of 30 minutes, much less than the time taken to reach the maximum altitude of 3000 ft.

The climb performance is not important at 3000 ft, as this is the maximum altitude designated in the RFP. However, climb performance was analysed at sea level, 1500 ft, and 3000 ft, and the results were numerically integrated to give the best climb angle through the increase in altitude. Ground distance covered and time to reach 3000 ft was also determined. The results are summarised in Table 6.

*Table 6: Climb Angle Performance at ISA and +20°C*

| **ISA Conditions** | | | | |
|---|---|---|---|---|
| Altitude | Best Speed | Best Angle | Ground Distance | Time Taken |
| 0 ft | 96 kts | 5.29° | 5.70 nm | 3 min, 30 sec |
| 750 ft | 96.75 kts | 5.12° | | |
| 1500 ft | 97.5 kts | 4.95° | | |
| 2250 ft | 99 kts | 4.78° | | |
| 3000 ft | 100.5 kts | 4.61° | | |

| **+20°C Conditions** | | | | |
|---|---|---|---|---|
| Altitude | Best Speed | Best Angle | Ground Distance | Time Taken |
| 0ft | 99 kts | 3.70° | 8.38 nm | 4 min, 58 sec |
| 750ft | 100 kts | 3.42° | | |
| 1500ft | 101 kts | 3.54° | | |
| 2250ft | 102.5 kts | 3.22° | | |
| 3000 ft | 104 kts | 3.05° | | |

Figure 17 is also included to show the variation of maximum climb angle, and the corresponding rate of climb, with forward speed. This chart shows that the maximum rate of climb is close to 150 kts, which is beyond the maximum possible speed of the Cronus (limited by the rotor tip speeds). The analysis was done only for the design case of +20°C conditions.

Figure 17 also illustrates the fact that a forward speed of approximately 100 kts might be useful if an high obstacle needed to be cleared. The maximum climb angle occurs at this speed. However, if there are no obstacles, the maximum aircraft speed of 130 kts is more desirable, because of the higher rate of climb. Ground distance will of course increase substantially.

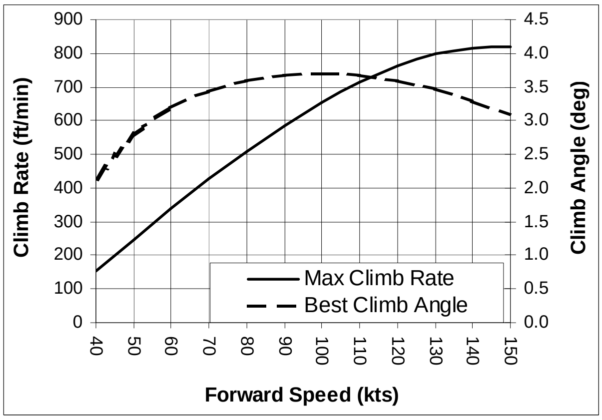


*Figure 17: Variation of Climb Angle and Rate with Forward Speed (Sea Level, +20°C)*

# 8 USE OF COMPOSITE MATERIALS

Because the Cronus is designed for military operation, the need for operational flexibility and adaptability must be emphasized, as well as the need for long operational life. Vulnerability and survivability of the rotorcraft in a combat situation are also very important issues, as helicopters are one of the easiest aircraft to shoot down in a military scenario.

When looking at the structural design, the strongest, lightest and most cost effective structure is required. For an efficient structure, the helicopter fuselage should be fully aerodynamically integrated with the main rotors and any empennage.

The primary structural airframe of the Cronus will be entirely made of composite materials. The use of composites results in reduced weight, with an increase in strength and reliability. The application of these composite materials requires the development of new structural concepts that make optimum use of the material properties and minimise the costs of manufacturing. Composites will also be used for non-structural members, as they provide light weight at a low cost.

While weight reduction is the main reason why composites are used, they also offer a tougher design, increased corrosion resistance and a reduction in the number of parts, thus reducing manufacturing costs. They also offer greater part-for-part consistency; repeatability is a considerable advantage compared to most sheet metal components that are subject to contour variations (Mason, 2005).

Composites also have higher specific strength and stiffness than metals, as well as other attributes such as tailorability to load directional, acoustic damping, and high fatigue endurance. The weight savings, combined with improved structural efficiency, are directly translated into increased payload, reduced acquisition and operating costs, and increased performance.

The primary structure of this rotorcraft is mainly made up of carbon fibre reinforced composite, dimensioned to fulfill crashworthiness requirements compared to MIL-STD-1290A.

## 8.1 Rotor and Hub Design

The rotor blades are manufactured from composites as they have a much longer operational life, they cost 25% less than current metallic blades, and they have almost double the operational life. This will result in an estimated 50 percent reduction in O&S costs (Boeing Co., 2003). In addition, they allow more aerodynamically efficient airfoil shapes and a higher overall twist rate that will result in improved hover and forward flight performance, as well as providing a high fatigue life.

The blades are made from carbon fibre reinforced nose part with a foam core, an aramid fibre reinforced tail part with a foam core, a tungsten bar in the leading edge and carbon reinforcements at the blade root (NLR, Date Unknown). The honeycomb core provides low weight and superior stiffness.

The rotor head, which consists of flexbeams and hub, is also made from composite materials. The flexbeams must be stiff in tension and flexible in torsion. The flexbeam is to be made from a unidirectional hybrid composite (graphite/epoxy and s-glass/epoxy). There are also angled plies at the ends to give reinforcement for the complex stresses. There will be a reduction in shear strains in the flexbeam thus an increase in the high-cycle fatigue life due to the use of hybrid composite (Edwards, 2002). The rotor hub is made from high strength titanium alloy, because of its has a high resistance to stress corrosion crack propagation.

Quasi-isotropic [0/+45/90]s HexPly® M18-1 and M42 (ref 6) carbon fibre laminates are used for the primary structure of the helicopter, because it has a higher specific stiffness and higher specific strength than most other materials. The exact fibre type may be chosen for each individual part, depending on whether the structure is stiffness or strength limited.

## 8.2 Ballistic Protection Considerations

In the areas where armour is required, the ideal fibre will have a high transverse tensile strength to cater for the shear stresses involved in the initial plugging, as well as a high longitudinal tensile strength to resist the tensile stresses created by fibre bending.

Carbon fibres have a relatively low tensile fracture strain, a very low transverse strength compared with their longitudinal strength, and a relatively low tensile ductility. This leads to poor energy absorption compared with that of fibres that have similar strengths but higher tensile ductility. This makes it much less useful for armour applications than for usual aerospace applications (Edwards, 2002).

Standard carbon fibre provides strength and stiffness in both the x and y directions, but there is little or no structural support in the plane perpendicular to the fabric (the z plane). The addition of z-pin fibres, small composite rods 0.28 millimetres in diameter, increases the through thickness resistance of the composite structure (strength in the transverse direction). This method is shown in Figure 18. Z-fibres are also highly effective in improving Mode I interlaminar fracture toughness. The use of Z-pins reduces crack propagation and increases energy absorption, especially where the failure mode includes delamination. All of this increases the ballistic impact damage resistance of the composite structure (Shonaike, 2003).

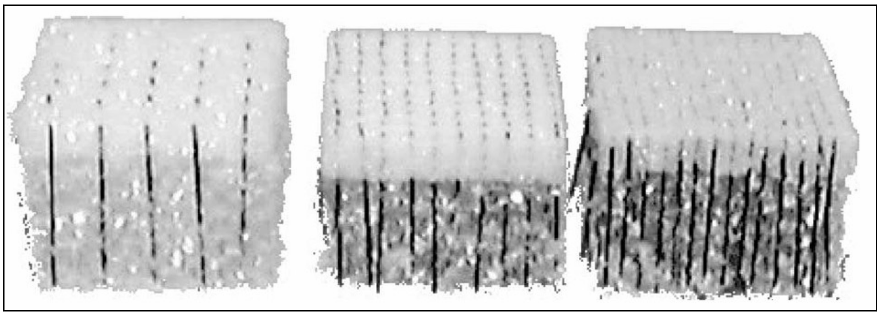

*Figure 18: Z-fibre preforms containing 0.28 mm diameter pins at densities of 0.5, 2 and 4% (image courtesy of www.sciencedirect.com)*

Figure 19 shows an example of armoured seats. They are typically constructed from a one-piece molded aramid composite shell with monolithic boron carbide tile appliqué. The seats are then finished with a covering of durable nylon fabric that suppresses the projectile during ballistic impact. Boron-carbide tiles may also be used for added protection.

*Figure 19: Armoured seats (image courtesy of www.ceradyne.com)*

## 8.3 Fuselage Materials Selection

The fuselage is made by laying up the carbon/epoxy part with automated fibre placement machines. This leads to a 65% fabrication cost reduction, a 25% part count reduction, 34% reduction in fastener count and overall cost savings of 53% compared with established methods. Other large parts, such as sponsons, side-skin panels and longerons, may also benefit from fibre placement. The fibre placement method has reduced defects significantly and almost eliminated scrap on the most difficult parts (National Research Council, 1997).

The side-body panels are made from carbon/epoxy prepreg. Each composite side-body panel is hand layed up as a single piece from either composite or metal tooling. The panels also incorporate phenolic resin-impregnated core material to ensure compatibility with the resin system. Panels are then vacuum-bagged and cured in an autoclave. Composite stiffeners are bonded to the inside of the side-body panels (Mason, 2005).

The cabin is constructed from aramid/epoxy sandwich panels with Nomex honeycomb core, as well as 180°C curing carbon prepreg. The aramid/epoxy is used especially for the cabin back wall and floor as it enhances fire protection and provides some measure of ballistic protection (Mason, 2005).

The panels are cut to the desired shapes, sealed with adhesive, and both adhesively bonded and mechanically fastened into the fuselage assembly. The adhesive provides bonding redundancy while acting as a sealant. Aramid is also used for the structure around the fuel tanks.

The stabilisers are to be made from carbon/epoxy prepregs, as are the doors, hatches, fairings, shrouds, skin panels, spars, glazing bars, and bulkheads. The sub-floor is to be made from Fibrelam composite sandwich panels.

The areas around the engine and the exhaust, such as cowling, empennage, and exhaust doors, which see high temperatures from hot exhaust gases, must also be considered. These sections of the Cronus are to be made from quartz/bismaleimide (BMI), carbon/BMI, as well as some glass/BMI. Phenolic-impregnated core materials are incorporated in the vast majority of these components (Mason, 2005).

## *8.4 Landing Gear Materials Selection*

The landing gear structure is a highly loaded, non-redundant, and has severe restrictions on weight and space. Ultra High Strength Low Alloy (UHSLA) steels are to be used for the undercarriage as they have sufficient static strength to withstand extremely high bending stresses, high fatigue strength, adequate corrosion resistance, adequate toughness to withstand shock loads, and sufficient wear resistance and impact resistance.

Unfortunately, the inclusion of steels incurs a weight penalty. For this reason, the torque links and trailing arm assembly will be made from carbon fibre composites, using resin transfer moulding (RTM). RTM is necessary because the shape of landing gear components are often complex and therefore very difficult to make using traditional prepreg/autoclave technologies (HPCM, Date Unknown).

# 9 SYSTEMS DETAILS

Several of the sub-systems on board are detailed in the following sections, including the landing gear, cargo winching system, and the advanced IHPTET engine.

## *9.1 Landing Gear Design*

The Cronus has a tricycle style landing gear, with a nose gear at the front and a pair of main gears mounted on either side of the rear fuselage. The nose wheel configuration is to be favoured for ease of ground maneuvering as well as for ground looping. Most aircraft, and some rotorcraft, are equipped with a nose wheel (tricycle) type landing gear configuration.

A trade-off study between a *Skycrane* and a more conventional landing gear configuration for the main (rear) gears was conducted. It was decided that a conventional type main gear would be more suitable since it is more effective in carrying the load in direct compression whereas the S*kycrane* configuration carries the load in shear. The conventional type landing gear is also easier to design. The main gears transfer ground loads to the reinforced ceiling of the Cronus.

The longitudinal tip-over criterion states that the main landing gear must be located behind the most aft centre of gravity location by an angle of approximately fifteen degrees (Roskam, 1997, page 218). The most aft centre of gravity was estimated to be 12.192 metres (480 inches) from the nose of the Cronus. This corresponds to the main gears being placed at a distance of 6.985 metres (27.5 inches) behind the aft centre of gravity (Conway, 1958). This Cronus also satisfies the lateral tip over criteria (Roskam, 1997, page 219) with a calculated overturn angle of 47.9°. The main landing gears are not retractable, but will only impose drag penalties when travelling at speeds of over 150 knots, which is more than the maximum speed of 130 kts.

The nose and the main gear are located such that almost 70% of the static ground load will be carried by the main landing gear (Currey, 1988, page 29). Due to the nature of the mission requirements, the nose and main gear tires will have pressures that are lower then the maximum prescribed. This will improve the expected life, and will also prove to be useful when the Cronus operates in uneven landing zones.

The front tire has a diameter of 610 mm (24 inches) and a tire base of 196 mm (7.7 inches), tires will have a pressure of 140 psi. The main landing gear, however, will have tires of 1016 mm (40 inch) diameter and a tire base of 356 mm (14 inches), with a pressure of 160 psi.

Oleo pneumatic shock absorbers were chosen for the main and nose landing gears, due to the superior performance over that of the simple hinged struts or rubber springs. The stroke of the oleo was designed according to FAA requirements (FAR 29.725, 2005), which require that a transport type aircraft must be able to withstand a drop test of 13 inches from the lowest point of the landing gear to the ground. Dropping the Cronus from such a height when fully loaded will result in drop velocity of 2.55 m/s. Assuming a tire stroke equal to a third of the tire radius, and taking into account the oleo and tire shock absorber efficiencies of 0.47 and 0.9, resulted in estimated strokes of 457.2 mm (18 inches) and 466.1 mm (18.35 inches) for the main and nose landing gear, respectively. Note that the stroke was increased by an inch in order to maintain an adequate safety margin (Currey, 1988, page 37). Calculating the oleo dimensions involved assuming an internal pressure of 1,800 psi, an external diameter 1.3 times the internal diameter, and an oleo length that is 2.5 times the required stroke (Currey, 1988). It is assumed that the overall minimum length of the undercarriage leg is equal to three times the stroke of the shock and the total distance from the ground to the top of the strut will be equal to the total gear length plus the radius of the tire being used. This results in a total distance of 1880 mm (74.02 inches) from the ground to the top of the strut for the main landing gears, and a distance of 1703 mm (67.06 inches) for the nose gear.

The issue of crashworthiness was also taken into consideration when selecting the locations of the landing gears. The main gears pose no threat to any crew or FCS operators, as they are located a considerable distance to the rear of the seating area. In the event of a crash, the nose landing gear poses no threat, since it is designed to penetrate the cockpit in between the pilots.

The various landing gear dimensions are shown in Figures 20, 21 and 22.

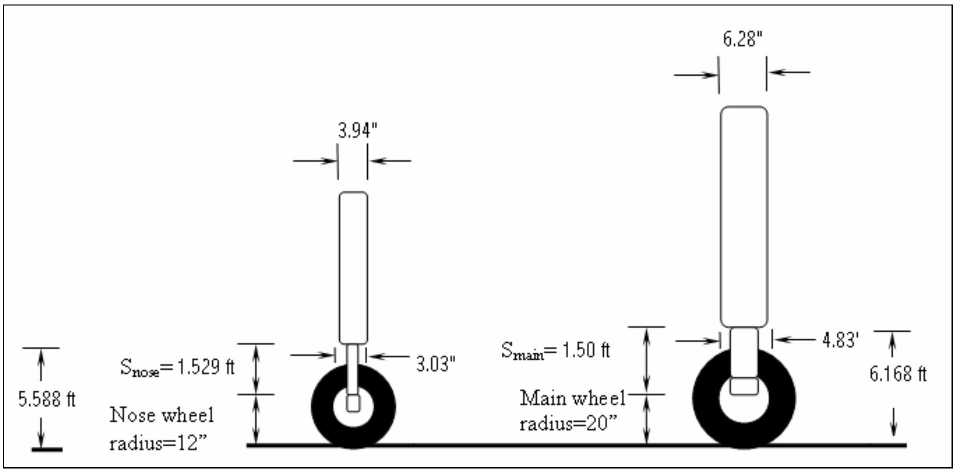


*Figure 20: Landing gear strut lengths*

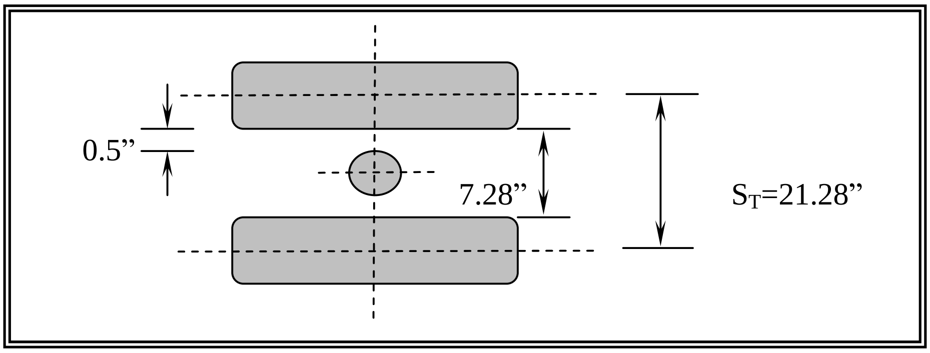


*Figure 21: Main landing gear tire dimensions*

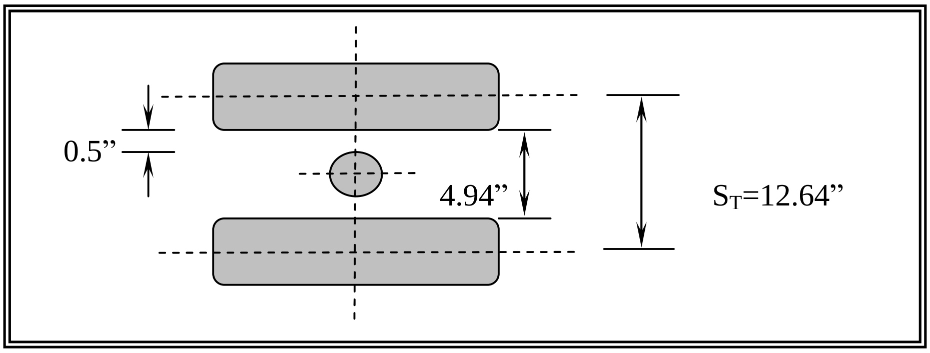


*Figure 22: Nose landing gear dimensions*

### 9.2 Cargo Handling Systems

The key mission role of the Cronus is to move and haul cargo in a short time-frame and/or into awkward locations. This means that cargo handling must be as efficient as possible. The Cronus' fuselage is of an open-body design, where the interior of the fuselage will contain the FCS vehicle. The Cronus will have the flexibility of allowing the FCS vehicle to either be driven up the ramp into the cargo bay or be winched up through an opening in the floor of the helicopter. Similarly the FCS vehicle can be unloaded either by winching (when in flight) or under its own power.

Research was conducted and it was found that the most efficient way to carry the payload is via a capable winch system. A winch system that can carry such a load (20 tonnes) has never been implemented on a helicopter, but the technology is currently available. It is therefore assumed that such technology could be used on helicopters in the near future. The selected design will include four hydraulic winches, located at the top four corners of the fuselage interior. The specifications of each winch (weight approximately 24 kg) is given in Table 7.

*Table 7: Winch specifications (BCG, 2003):*

| **Line Pull** | **Line Speed** | **Rope Capacity** |
|---|---|---|
| (kg) | (mpm) | (m) |
| 9090 | 8 | 11 |

It should be noted that if the FCS vehicle is to begin its mission by being winched up into the Cronus and then unloaded by being driven out, then a platform must be installed to cover the majority of the opening in the floor of the Cronus. This platform will allow the winched FCS to be released onto it and secured to it by the loadmaster until it is ready to be driven out of the Cronus. As it can be seen in Figure 23 below, the platform is hinged to the floor of the Cronus and easily covers the opening. Note that the floor around the opening is reinforced to withstand the loads transmitted to it.

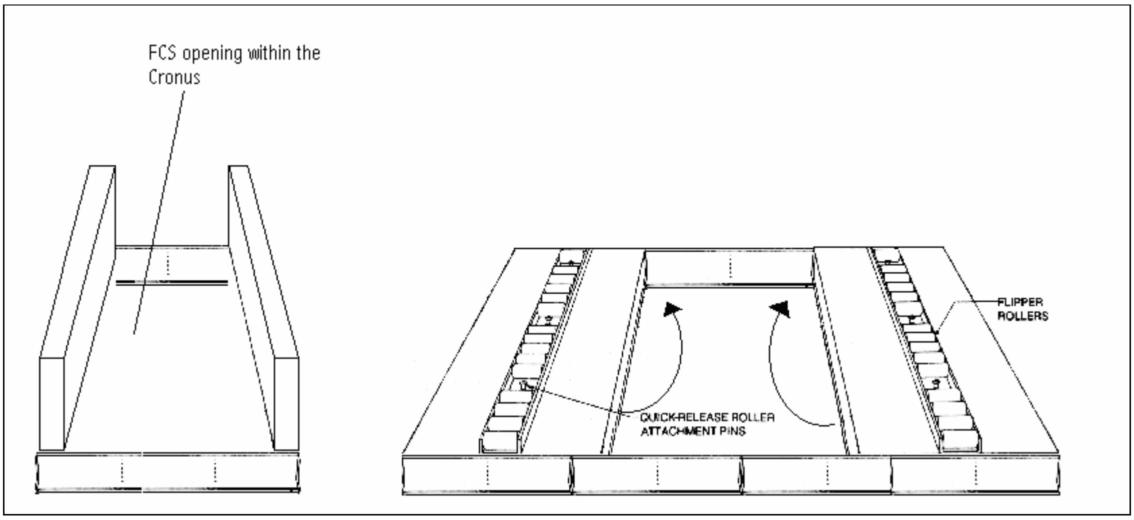


*Figure 23: Cronus cavity and platform*

### 9.2.1 Cargo Hooks

A hook will be attached to each winch. Sensors will be located at each hook in order to relay information back to the cargo management system (CMS). The CMS monitors the status of all cargo systems during the transportation of the load. This monitoring system will be integrated into the cockpit displays and will provide the aircrew with crucial information about the load, power available, power required for the current conditions, and other relevant data.

The FCS vehicle must be tightly fastened and secure throughout the mission. It will be fastened to the helicopter using certified heavy lift cargo hooks, which are attached to each winch. Each cargo hook is encased within two aluminium side plates and a spacer, which are bolted together. The internal parts are made of high quality steels and have been heat-treated and surface protected. Each hook is designed to withstand a static limit load of 13000 kg (28660 lb), ultimate load of 20500 kg (45195 lb).

The payload may be released manually with a release knob, or electrically using a solenoid mechanism. However, it is still the responsibility of the loadmaster to ensure that the payload has been released successfully.

The Cronus is also capable of transporting up to four 463L pallets in one flight mission. It should be noted that the pallets can be transported using the Chinook external cable method, the FCS vehicle (internal) method, or simply secured within the cargo bay. The exact method depends on the number of pallets to be transported; delivery mode; and location of delivery. These pallets will be joined together prior to transportation using standard pallet nets.

Similar to the CH-47 Chinook, which is equipped with one primary and two secondary cargo hooks, the Cronus will also have one primary cargo hook (shown in Figure 24), in addition to the winches. This hook is located at the centre ceiling of the cargo bay, at a distance of 10.97 metres (432 inches) from the nose, and is capable of carrying 11.8 tonnes (26015 lb).

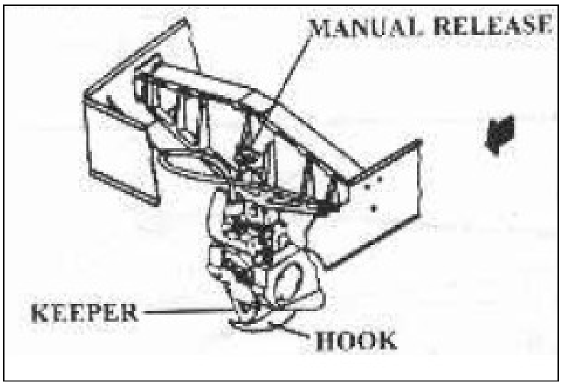


*Figure 24: Main payload hook (DAHHC, 2004)*

There are also two other secondary hooks (shown in Figure 25), located underneath the Cronus aft and forward of the cargo bay at distances of 15.04 metres (592 inches) and 6.88 metres (271 inches) from the nose of the Cronus. These secondary hooks have a combined lifting capacity of 11.3 tonnes (24910 lb), or 7.7 tonnes each (16975 lb) (DAHHC, 2004).

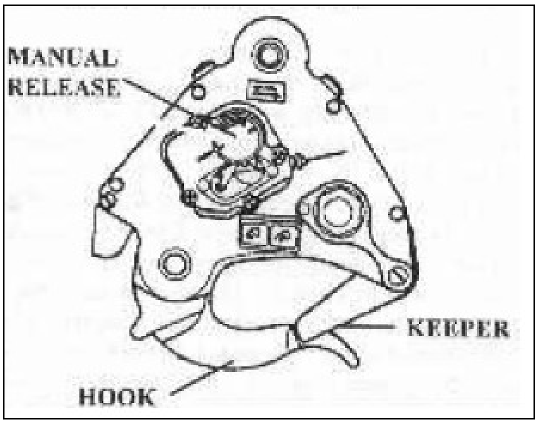


*Figure 25: Forward and aft secondary hooks (DAHHC, 2004)*

These cargo hooks are designed to lift, transport and release external loads from a cable suspended from the Cronus. All of the hooks contain only four or five moving parts, which equates to less downtime and lower maintenance costs. Figures 26 and 27 show the summarised layout of the winches and cargo hooks.

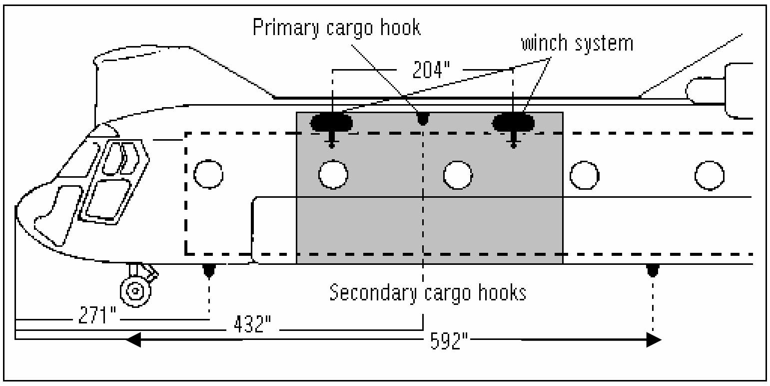


*Figure 26: Winch and hook locations (side view)*

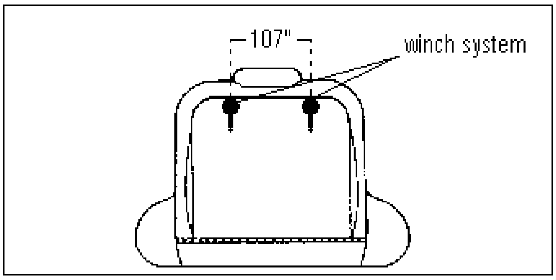


*Figure 27: Winch locations (rear view)*

### 9.2.2 Advanced Handling Cargo System

The floor of the Cronus will be fitted with an advanced cargo handling system similar to that found on the Chinook. This system is designed to provide rapid loading/unloading in a hostile environment and thus minimizing turn-around time and exposure to threat weapon systems. This system significantly increases the mission effectiveness and combat survivability. It has a high strength-to-weight ratio, is made up of interchangeable parts, and is salt water resistant (DRST, 2005).

The cargo system will provide a vertical down restraint that is by using two rows of roller trays that are bolted to the aircraft track (DRST, 2005). Additional restraint is provided through tie down rings and tie down straps. There are lateral rows of side rails which contain positioned cargo locks and associated controls that provide forward and aft restraints of any 463L pallets. The Cronus also has the option of being equipped with commercial LD3 container restraints. This is similar to the AM-104 cargo handling system, found on the CASA CN-235 aircraft. It comes equipped with a pendulum release and tow plate for accomplishing tactical airdrop operations (DRST, 2005).

The cargo bay will have a built in Night Vision Goggle (NVG) and white lighting system, shown in Figure 28. This system is environmentally sealed to provide maximal efficiency of operation in salt water environments. The NVG lighting system is made up of dual dome lights on each side of the cargo system that features a "dimming" function to both the NVG and white-light; these will be located along the cargo bay.

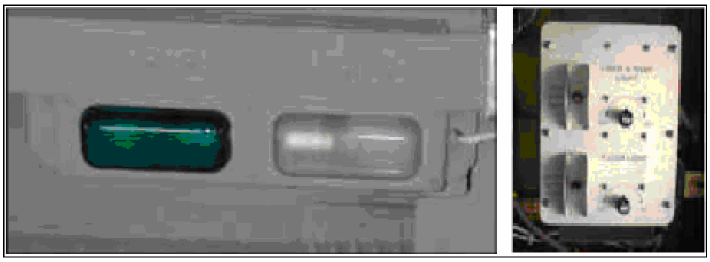

*Figure 28: NVG and white lighting system (DRST, 2005)*

Combat vehicle guides will be installed on the rotorcraft to ensure that driver steering input is kept to a minimum. These guides have a contoured edge in order to minimize the vehicle's tendency to climb. These guides are depicted below in Figure 29.

*Figure 29: FCS vehicle guides (DRST, 2005)*

A flip-over roller system (the Chinook version is shown in Figures 30 and 31) is also installed, allowing for rapid reconfiguration of the cargo floor. This system is comprised of lateral side rails and a logistic locking system that are affixed to the outboard row of tie down rings. Rows of embedded flip-over roller systems are integrated into the floor in the cargo compartment and ramp. Two different roller assemblies are provided for the hatch cover/flippers and to bridge the gap between the cargo compartment and ramp. Each lateral row of side rails contains logistic cargo locks and associated controls to allow for incremental release of the locks from the aft portion of the aircraft forward. The system also includes a new heater duct providing mounting points for the NVG lighting system and the vehicle guide system (DRST, 2005).

*Figure 30: Chinook cargo handling system (DRST, 2005)*

*Figure 31: Flip-over roller system (DRST, 2005)*

## 9.3 IHPTET Engines

The Cronus rotorcraft will require a very large amount of power to be able to fulfil the required mission. To carry a payload of 20 tonnes, at least two turboshaft engines are required. There are no engines currently on the market that have enough power and a low enough specific fuel capacity (SFC) such that they could be used on this vehicle. However, as the aircraft will not be operational until 2018, future technology may be assumed.

This advanced engine technology is known as Integrated High Performance Turbine Engine Technology (IHPTET) Phase III. This program was introduced in 1987, in an effort to double aircraft propulsion capability. The IHPTET program is a national collaborative effort among the United States Air Force, Navy, Army, NASA, DARPA, and aircraft industry (National Research Council, 1997). The Joint Turbine Advanced Gas Generator (JTAGG) demonstrator represents the small-to-medium size turboshaft/prop/fan class.

The Phase III of the JTAGG program estimates that its engines will (AHS, Date Unknown):

- decrease SFC by 40%;
- increase power-to-weight ratio (shp/wt) by 120%;
- increase range by 100% or increase payload by 30%;
- reduce maintenance costs by 35%; and
- reduce production costs by 35%.

This increases the turboshaft's engine capabilities to 6,000 hours of creep/stress rupture, 15,000 cycles for low cycle fatigue in compression system, and 7,500 cycles for low cycle fatigue in the hot section (turbines, combustor, and exhaust) (AHS, Date Unknown).

To estimate the possible engine capabilities, a pre-1987 state of the art turboshaft engine may be used as a baseline. The changes to the engine listed above may then be applied to the engine properties. The resultant power-to-weight will then be held constant, and the engine weight scaled until the desired power output is achieved (the design case for power requirement is given in Section 7.4)

Considering the IHPTET Phase II and III technologies, this engine is expected to be in production by the year 2014. This engine will be in the 7,500 kW (10,000 SHP) class, giving a four-fold increase in range and doubling the payload capability over today's heavy lift helicopters.

The general idea to doubling propulsion system capability is well known. Higher temperatures at combustion are required to decrease fuel consumption or increase maximum flight speed. Higher maximum temperatures are required to increase the specific thrust or power/weight ratio. All of these advances must be accomplished while maintaining or increasing component efficiencies, durability, and life and by reducing cost (DOD, 2000).

Specific technology development areas include:

- advanced materials that exhibit higher temperature capability and higher strength per unit weight;
- improved aero-thermodynamic design for improved component efficiencies and heat transfer;
- innovative structural concepts for a reduction in moving parts, resulting in improved durability; and
- compatibility of developments with lower cost manufacturing processes and maintenance requirements (DOD, 2000).

These are some of the changes made to the engine components, some examples of which are shown in Figure 32:

- **Splittered rotor with improved aerodynamics:** produces a high-pressure ratio in a single stage.

- **High temperature rise combustor:** features a ceramic matrix composite (CMC) liner with an advanced dome. The combustor demonstrated a low pattern factor at the goal fuel-to-air ratio resulting in higher performance. For combustors, the major advances required are increases in both inlet and outlet temperature capability, reductions in weight, and reduced cost. For augmentors, the major advances required are increases in temperature capability, reductions in weight, and increased efficiency (AFRL, 2005b).

- **High pressure compressor (HPC) impeller with split inducer/exducer:** Has demonstrated its suitability for use in JTAGG III. Incorporates forward sweep and independent inducer and exducer for high efficiency, increases in specific output of a compression stage, reductions in weight and leakage flows, and operation at high pressure ratio and higher compressor exit temperatures (AFRL, 2005a).

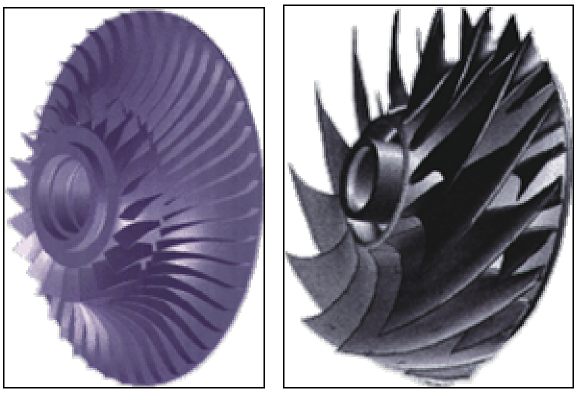

*Figure 32: (left) JTAGG III advanced concept centrifugal impeller; (right) integrating forward sweep and splitter technology (images courtesy of AFRL, 2005a)*

- **Low-pressure compressor with integrating forward sweep and splitter technology:** Key features to achieving high efficiency and high pressure ratio. This stage will be utilized as the JTAGG III.

- **Monolithic ceramic low pressure turbine (LPT) blades:** successfully tested for burst and tip rub to demonstrate durability and reduced cost.

- **Very small high pressure turbine blades:** made from advanced MX4 materials, have been water flow tested to verify cooling flow and represent the latest 3-D shape for high efficiency. The major advances are increases in temperature capability, reductions in cooling flow requirements, increases in work done per stage, and increases in thermodynamic efficiency (AFRL, 2005b).

- **Exhaust system:** The major advances required for exhaust systems are decreases in weight for both treated and untreated nozzles, reductions in leakage, and improved functionality in thrust vectoring.

- **CARL 2 stage rig:** Improve stage-loading levels of IHPTET Phase III.

The new technologies going into these engines are what makes the previously stated goal factors possible. Specifications for the Cronus engines may now be obtained.

Considering the existing engine from the Boeing CH-47F, and the Honeywell T55-L-714 (Figure 33), we can use existing data and multiply it by the goal factors.

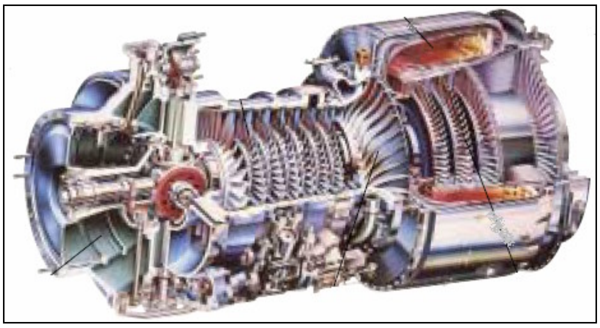

*Figure 33: Honeywell T55-L-714 (image courtesy of HA, 2002)*

Multiplying the power-to-weight ratio by 120%, gives a power of approximately 6,800 kW (9,170 SHP) and an engine weight of 377 kg (830 lb). The installed power requirement from both engines was given in Section 7.4 as 8,430 kW at intermediate power setting, 3000 ft altitude on an ISA +20°C day. This means the engine must be scaled up to a weight of 420 kg (926 lb). Reducing the SFC by 40% gives a new figure of 0.29 (HA, 2002). The improved engine performance is detailed in Table 8.

*Table 8: Existing and new engine specifications (sea level, ISA conditions)*

| Power Rating | Existing Power (kW) | New Power (kW) | Existing SFC | New SFC |
|---|---|---|---|---|
| **Contingency** | 3,780 | 9,465 | 0.496 | 0.2976 |
| **Maximum** | 3,630 | 9,090 | 0.493 | 0.2958 |
| **Intermediate** | 3,375 | 8,455 | 0.491 | 0.2946 |
| **Max. Continuous** | 3,110 | 7,785 | 0.494 | 0.2964 |

The following are some of the factors that might also significantly improve core engine performance. These should be comprehensively studied as they may improve the specific core power and specific weight of the IHPTET Phase III engines by up to 50%.

- High-temperature metals and non-metals;
- improvements in aerodynamic component efficiencies;
- active closed loop flow path control;
- improved cooling;
- variable cycle engines (Figure 34); and
- environmental implications (National Research Council, 1997).

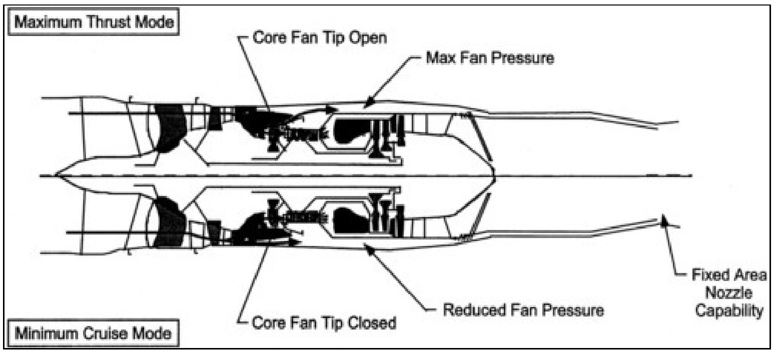


*Figure 34: Schematic of a variable cycle engine (image courtesy of General Electric Aircraft Engines)*

# 10RELIABILITY AND MAINTAINABILITY

Based on the experience of the Navy and the general helicopter community, there are several reliability and maintenance issues that will be expected and even unavoidable. The unique nature of the mission requirements for this vehicle means that it will be exposed to over 20 flying hours for every separate mission, in conditions ranging from desert to marine. Any subsequent issues of maintainability are also compounded by the fact that the vehicle is based on an relatively isolated Navy ship, meaning access to specialised equipment and endless spare parts is unavailable.

A strong part of the design philosophy for this aircraft is the concept of modularity; that is, various sub-systems are quite separate from each other, and are easily removed, repaired, replaced, or upgraded. Minimal but adequate means will be used to attach sub-systems to the vehicle structure, such that in the event of mid-mission malfunction, the sub-system in question is quickly replaced.

The reliability and maintenance issues must be kept in mind at all stages of the design process. Parties responsible for the design of sub-systems and their integration should first look to avoid foreseeable problems, and if avoidance is not possible, to minimalise and control the problems. For critical sub-systems, a fully redundant design will be used. For example, the inclusion of two engines means that, should one fail, the other is entirely capable of either finishing the mission or returning safely, even whilst carrying the maximum payload.

The current and subsequent design phases must also anticipate and make use of technology that will become available in ten years' time. This is another reason for modular design; Navy operations and resources are expected to advance beyond their current state. The aircraft must be reliable and maintainable within the future Navy framework, and not the current.

### 10.1 Problems and Solutions

The expected reliability and maintenance issues are listed here with solutions.

---

Corrosion, due to ship-bound nature of operations.

> Utilise composites wherever possible; ensure regular inspection of components that are critical or known to be a problem; design for minimum of exposed moving parts.

---

Frequent breakdown of avionics equipment, due to vibration and heat.

> Design helicopter for minimal vibration; place avionics racks in easily accessible locations; ensure avionics compartments are not in direct sunlight and are well sealed against the environment.

---

Gearbox wear, due to huge engine power and two or three gearboxes.

> Utilise high-strength and high-wear materials; keep moving parts to a minimum; employ redundant oil cooling system; design gearboxes to be easily accessible and attached with as few fasteners as possible.

---

High technology parts, especially avionics and engine, needing specialised tools.

> If components do require specialised tools, use simple adapters to integrate them into the airframe; try to select systems common to future Navy operations; familiarise maintenance staff with new technologies and tools.

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Folding rotor blades are a point of weakness for an extremely critical component.

> Employ redundant locking devices; include cockpit warning and status systems; use non-corrosive materials; employ highly rigid maintenance and pre-flight preparation procedures.

---

Avionics becoming obsolete and needing replacement.

Design avionics components to be highly modular and easily removed; position avionics racks in easily accessible locations.

---

Relatively isolated from spares and supplies, both ship-bound and in-field.

Include redundant systems wherever possible, especially for critical systems; attempt to select systems common to future Navy operations.

---

Winching system inspection and easy replacement/repair.

Protect winching system from environment; design system to be easily accessible; attach entire system to airframe as simply as possible, with standard fasteners; include as few moving parts as possible.

## 10.2 Additional Design Philosophies

Although not seen as problems to be solved, the following issues are pertinent to reliability and maintenance. These design actions have been implemented on the aircraft.

- Fuel tanks must be designed for ease of inspection and cleaning.
- The engines must be designed for easy removal and maintenance.
- Blade fatigue must be closely monitored, with blades made to be individually removed and replaced.
- All other main sub-systems must also be easily accessible for inspection, removal and/or repair.
- Parts lists, with particular attention to specialised fasteners and tools, must be made up and delivered with the aircraft.
- Use standard parts and tools as much as possible.
- An exhaustive and rigid maintenance schedule must be designed along with the aircraft itself, drawing on helicopter and Navy experience.

## 10.3 Advanced Technology Inclusions

Because the project timeline sees initial deployment over ten years from now, several technologies that are currently untested and risky are to be included. These advanced technology inclusions should serve to improve safety, by allowing the aircraft to communicate more effectively with flight and maintenance crew concerning the "health" of certain critical sub-systems.

### 10.3.1 Gearbox Health

Naturally, the gearbox is a highly critical sub-system, and should be replaced or repaired well before failure occurs. Instead of periodically removing and inspecting the gearbox, two separate sensors will be installed to monitor its operation. One sensor in the sump detects the amount of particle matter, which gives an indication of how much wear has occurred. Excessive wear may occur because of incorrect or insufficient lubricant, poorly manufactured components, fatigue limits reached, or foreign matter entering in from the environment. When particle concentration reaches a certain level, the gearbox will be inspected, and then repaired or replaced.

The second sensor detects the vibration signature of the gearbox. Standard signatures for the full normal range of engine operation will be compared to, and any deviation will send a warning to the flight deck. If the deviation is due to an overloaded flight regime, the pilot should correct immediately to avoid doing damage to the gearbox.

### 10.3.2 Rotor Health

The rotors are highly critical to safety and mission success and every attempt should be made to ensure their health. In particular, the rotors will be subject to large bending moments and extreme cyclic fatigue. Composites are used extensively, and flight and maintenance crew must be made aware of any immediate or gradual damage.

A new technology will be used that sees sensors embedded in the composite matrix of each blade, especially near the blade root where bending moments are greatest. These sensors detect excessive or abnormal strain due to bending, and send a warning to the flight crew.

Depending on the severity of the malfunction, the pilot may choose to adopt a less strenuous flight regime, or attempt an emergency landing.

Because each rotor is also hinged and foldable, extra sensors will also be required, so that the flight crew are aware of the exact status of the folding mechanism, ie., fully folded or fully extended and locked.

# 11VIBRATION CONSIDERATIONS

The rotor forces and moments causing fuselage vibration are transmitted from the blades to the rotor hub, and then by means of the main rotor drive shaft into the main rotor gearbox bearings, and hence into the gearbox casing. The fuselage itself is then affected through the gearbox attachment points. The control of these vibrations is important for four main reasons, improving:

1. crew efficiency and hence safety of operation;
2. the comfort of passengers;
3. the reliability of avionic and mechanical equipment; and
4. the fatigue lives of airframe structural components.

It is therefore very important to consider vibration control at all stages of design, development and service.

## *11.1 Dynamic Design of Rotor Blades*

The successful design of a helicopter rotor system is concerned with meeting the exacting dynamic requirements. These must be satisfied if acceptable behaviour is to be achieved in the areas of handling qualities and fuselage vibration levels. A trend towards increasing mechanical simplification combined with increased aerodynamic and structural efficiency has intensified the difficulties experienced in achieving the optimum dynamic characteristics. However, the particular properties of composite materials, to be employed on the Cronus, have more readily enabled the design aims to be met.

The correct dynamic design of rotor blades is essential for 2 main reasons:

- The minimisation of the amplification of the rotor blade aerodynamic vibratory loading, which is transmitted to the fuselage.
- The minimisation of the total vibratory loading of the blade to provide an acceptable fatigue life.

## 11.2 Rotor Gearbox Mounting Systems

In order to minimise the vibratory forces fed into the fuselage, various methods of mounting the rotor gearbox have been considered. The following descriptions form a representative selection of the main passive systems, which have been used with varying degrees of success.

- Simple soft mounting of the rotor/gearbox/engine system. The lower the ratio of natural frequency to excitation frequency, the more reduced the force transmitted through the spring to the support. Since the quasi-static deflections of the suspended system would be unacceptably large if conventional soft mountings are placed at the gearbox feet, it becomes necessary to isolate as large a mass as possible. Hence it is attractive to mount the gearbox and engines on a raft, and then attach the raft to the fuselage using the chosen mountings. By this means, an adequately low natural frequency of the isolated system can be achieved without unacceptably large displacements of the rotor, gearbox, engines and controls under trim and manoeuvre loads. However, attenuation of the vibratory forces by this means is only modest and certain loadings may be transmitted without any reductions.

- A rotor/gearbox/engine mounting system may be designed to respond to the $b\Omega$ (where b is the number of blades and $\Omega$ is the rotation rate) forcing frequency in such a manner that the spring and inertia forces generated by the response cancel out, as far as possible, the effects of the rotor generated force.

- Systems may employ the Dynamic Anti-Resonant Isolator (DAVI). Originally developed for the isolation of crew seats by the Kaman company, this principle has been applied very successfully to gearbox mountings.

## 11.3 Dynamic Response of the Fuselage

Due to the fundamental design requirements of the helicopter, primarily in terms of the requirements of crew and equipment, and aerodynamic drag, the basic shape of the fuselage must generally be determined by considerations other than vibrational characteristics. In the early stages of design, the fuselage can be represented by an assemblage of beams of defined bending and torsional stiffness properties together with the appropriate mass distribution. Calculations can then be performed which will:

- indicate the rough proximity of any major beam bending mode frequency to the bΩ rotor forcing frequency; and

- indicate the sensitivity of the forced response to changes in the stiffness of structural components which may be amenable to significant alteration, eg, main rotor head stiffness relative to the fuselage, and tail boom stiffness. Thus, if ground vibration or flight tests indicate evidence of a major forced response problem, it may be possible to significantly reduce the response by making an unacceptable change to the structural stiffness in a particular area.

## *11.4 Vibration Absorbers*

Having made all the beneficial stiffness and mass distribution changes of which are possible without major redesign, it may still be the case that the levels of vibration experienced in flight at crew stations or passenger locations are unacceptable.

By attaching an extra small mass by means of a spring or elastic mount to a body undergoing forced vibration, the amplitude of vibration can be reduced to zero if the natural frequency of the sprung mass by itself is made to coincide with that of the forcing frequency.

## *11.5 Active Control of Vibration*

### 11.5.1 Application of Higher Harmonic Pitch Control

Since airframe vibration originates with the rotor blade oscillatory air loads, a potentially attractive method of vibration control is the application of blade pitch at frequencies greater than the rotor rotational frequency, in order to produce a forcing system generating an airframe response which would oppose that arising from the basic rotor forcing.

### 11.5.2 Active Control of Structural Response

This consists of connecting a number of actuators between convenient points on the airframe to apply oscillatory forces to the structure. The magnitude and phase of the loads generated by the actuators are determined by a control algorithm which minimises the vibrational response of the fuselage at a number of key positions.

If a force F is applied to the structure at a point P and an equal and opposite force (the reaction) is applied at a point Q, then the effect will be to excite all the modes of vibration of the structure which possess relative motion between points P and Q. This requirement for relative motion in the modal response between the points where the actuator forces are applied is an essential feature of this procedure.

### 11.5.3 Vibration at Frequencies Other Than bΩ

The minimisation of forcing frequency components from the rotor system which are normally self-cancelling within the rotor is very much related to the ability to manufacture identical blades and lag dampers. The major residual is usually at 1Ω frequency, and this is minimised by blade balancing and tracking procedures on a whirl tower and on the helicopter during ground running, and if necessary also in hovering and high-speed forward flight.

Reasonably identical lag plane damper performance characteristics are required so that undesirable vibration under conditions of 1Ω blade flapping can be avoided.

A quite distinct type of vibration problem of aerodynamic origin is due to the effects of vortices shed from the region near the main rotor head which can, under certain flight conditions, strike the tail rotor and the fixed vertical and horizontal tail surfaces. This phenomenon has been observed mainly as a response at the fuselage fundamental lateral bending frequency and is particularly dependent on the sideslip angle in flight. This problem may be intensified by the addition of excrescences such as a radome to the upper surface of the fuselage aft of the rotor. This is not a real problem for the Cronus, because of the absence of anti-torque devices.

There can also be vibration problems due to mechanical excitation arising from the transmission system. The effects of these sources of vibration can be held to acceptable limits by the correct positioning of shaft whirling speeds, an adequate standard of static and dynamic balancing of the rotor hubs and the transmission shafts, and the accurate machining of gear teeth profiles.

# 12DETAILED ROTOR DESIGN

The Cronus employs a tandem, contra-rotating rotor configuration, which eliminates the need for an anti-torque system. This also facilitates the overall design process, because only one rotor system need be designed, which will then be used front and rear. The main rotor is a critical design component, requiring careful consideration of often conflicting boundary requirements, in order to realise the desired performance characteristics of the Cronus.

This section will present and justify the prominent design features of both the rotor and hub, with particular emphasis on the baseline rotor design, rotor sizing, blade and hub configurations and prospective and final solutions to noise and vibration reduction concepts, by consideration of various blade tip planforms.

## 12.1 Baseline Rotor Design

*Table 9: Main rotor design parameters*

| Parameter | Value |
|---|---|
| Blade Diameter | 22 m (72.13 ft) |
| Number of Blades | 4 |
| Aerofoil Chord Length | 0.47 m (1.54 ft) |
| Solidity | 0.0544 |
| Twist | 4° |
| Sweep (Outboard 0.1R) | 12° |
| Tip Speed | 831.5 – 662.7 ft/s |

The main rotor design derives from a series of iterative calculations based on initial estimates, which were progressively refined by means of selecting optimum design parameters that would ensure adequate hover and forward flight performance characteristics. Table 9 summarises the main rotor design parameters that were opted for as a result of conducting the iterative rotor design process.

## 12.2 Blade Diameter

In generating a suitable blade diameter, numerous determinants had to be evaluated, such that the desirable performance characteristics were achieved, whilst ensuring that the adverse effects of the selection were minimised. Some of the factors, which limited the final sizing of the rotor diameter are as follows.

Paired with the rotor's angular velocity (RPM), the blade tip speeds were a governing factor in selecting an adequate blade diameter. Compressibility effects at the leading edge of the advancing blade was used as the critical case criteria in ensuring that the blade diameter is adequate (given a discrete rotational speed) to avoid the effects of adverse noise and vibration due to the presence of local supersonic flow regimes. Other considerations which were taken into account when finally deciding upon a relatively large diameter of 22 metres, were the inherent advantages of rewarding auto-rotation characteristics and lower induced drag requirements, which came at the cost of accepting unfavourable vertical drag characteristics. The final blade diameter of 22 metres with four blades resulted in a disk loading of 56.8 kg/m$^2$ (12.36 lb/ft$^2$), for the engine design case hover at 3000 ft altitude on an ISA +20°C.

## 12.3 Blade Tips

The tip speed is limited by the upper and lower speeds imposed. An acceptable diameter for the rotor can then be found if a maximum tip speed is assumed. This maximum tip speed will occur at the maximum allowable Mach number for a blade tip, which occurs at around 0.82 for a straight through blade, and increases towards Mach 1, for an advanced aerodynamically designed tip, incorporating such features as sweepback and anhedral. For the iterative design process, it will be assumed that the design of the tip will result in a reasonable decrease in the effective Mach number of approximately 20%. With an upper limit for the tip speed, an acceptable limit for diameter is then calculated. The design of the disc and blade then follows this.

The selection of the blade tip speeds required careful consideration of various and often conflicting limiting parameters including: compressible flow; induced noise and vibration; blade stall; and autorotation characteristics. Initially, for a given cruise altitude of 3000 ft and a cruise speed of 100 knots, it was determined that the overall tip speed should be such that the tip speeds of the advancing blade should be no more than 0.8M for the local atmospheric conditions. This corresponds to a maximum allowable tip speed of 268.8 m/s (882 ft/s) for the specified altitude. Given that the cruise velocity had been established to be 100 knots (168.8 ft/s), these flight conditions imposed an upper limit tip speed on the advancing blade of 217.3 m/s (713.2 ft/s) in cruise. Hence the blade tip difference is accounted by the difference between forward flight and hover. This corresponds to a lower limit tip speed of 165.9 m/s (544.4 ft/s), which is employed throughout the hover phase of the mission.

The final tip speeds however, did not reflect the ideal solutions discussed above. Through the iterative design process it was determined that an rotational speed of 195 RPM was required, and through the numerous changes that were made to the blade diameter (an overall increase), the final tip speeds were 291.9 m/s (957.7 ft/s) for cruise and 225.0 m/s (738.2 ft/s) for hover (the difference is again accounted for by a modified cruise speed of 130 knots). The implication of this compromise resulted in tip speeds that are operating at 0.838M, which is unlikely to result in excessive shock wave formation. However, the provision of innovative blades, such as the BERP tips, is still a necessity, to ensure that the compressibility effects are eliminated.

### 12.3.1 BERP Tips

These special rotor blade tips were designed under the British Experimental Rotor Program, and have been successfully used on several high-speed helicopters. The tips to be used on the Cronus are shown in Figure 35. The high progressive sweep delays the effects of compressibility to a higher advance ratio. The larger area ensures that the mean centre of pressure is located close to the elastic axis of the blade, which is important for blade stability and vibration effects. The distinctive ‘notches’ on the inboard and outboard regions of the tip allow for a retarding of flow separation by means of a vortex.

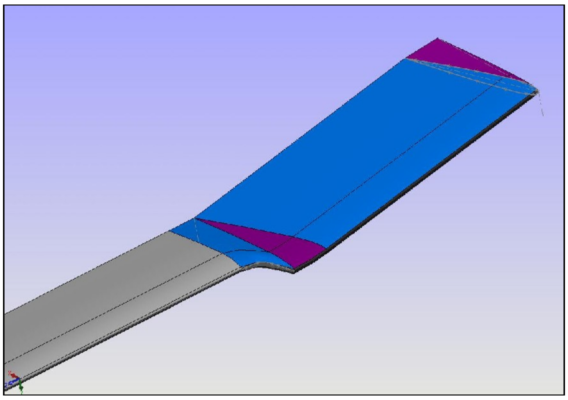

*Figure 35: Actual BERP tip*

It has been assumed that because of the sweep angle of 12°, the component of local velocity over the chord of the blade tip is simply the unmodified local velocity multiplied by cos12°. Therefore the local Mach number is 0.819, which is below the speed at which compressibility effects become a problem (Hoffmann, 2004, page 96). This is a conservative way of analysing the problem, because it does not take into account the other anti-shockwave effects of the BERP tips.

### 12.4 Solidity and Number of Blades

One of the governing factors that confine the solidity and number of blades, in addition to flight performance is the implication of weight and cost. By having a low blade area both the weight and cost are reduced, whilst also enhancing hover performance, which is achieved by providing an optimum angle of attack for each blade element and reduces the tip vortex interaction by the following blade. Countering these benefits however, a high blade area is suited for high-g maneuverability in forward flight with a low blade stall.

The number of blades affects the solidity of the rotor disc, that is, the ratio of area of the blade to the area covered by the rotor disc. Solidity of the rotor affects the disc loading for the rotor, and the rotor's performance. Increasing the number of the blades also has the effect of decreasing the effect of various vibrations. The more blades, however, the larger and more complex the design of the hub will be. The decision on the number of blades must be made considering the rotor solidity, the advantages and disadvantages of the effects of vibration and mass, and the design of the hub. The Cronus design incorporates two four bladed rotors.

Once an optimum blade area has been established by means of considering the weight and cost drivers, an adequate number of blades needed to be selected. It was decided to use four blades on each hub, given that a lower number of blades not only reduces the cost and weight of the hub but is also less prone to ballistic damage. In addition to this, the provision of a four blade rotor design also enhances the ease-of-storage requirement that is imperative in meeting the spatial elevator restrictions. A fold back hub mechanism with only four blades is very feasible. Furthermore, through the minimal blade number approach structural benefits are also obtained, in that the torsional blade stiffness is increased.

## 12.5 Sweep

Incorporating sweep into the blade's design can reduce the effective Mach numbers along the outer sections of the blade, and this, coupled with the effect of a large enough taper, reduces the transonic effects that would be felt by the blade at the upper limits of the flight speed. A tapered blade prevents shock from forming along the entire leading edge at once, and so sudden shock effects may be reduced or effectively eliminated.

## 12.6 Twist

Incorporating twist into a blade gives a significant improvement on the span-wise pressure and lift distributions on a blade, when compared to an untwisted blade. Negative twist in a blade (decreasing angle of attack towards the blade tip) gives a more even span-wise lift distribution, since the faster moving sections of the blade towards the tip present a lesser angle of attack, while the slower moving sections towards the blade root present a coarser angle of attack. This also reduces some of the undesired effects of high lift at the blade tips, and improves the loading conditions on the blade. Twisting of a blade, similar to washout of a fixed airplane wing, has the effect of bringing the onset of tip stall before the stall of the major section of the wing, and thereby giving indication that stall is occurring, before the entire blade itself becomes too close to stall conditions. Twist also has the effect of improving vibration characteristics, and of presenting a lesser angle of attack, when higher lift may be experienced closer to the root of the blade.

## 12.7 Taper

Tapering the blade improves the aerodynamic performance of the blade. It has the effect, similar to the effects of twist, of giving better pressure and span-wise lift distributions, since the higher speed of flow experienced toward the tip of the blade meets a smaller chord, whilst the lower speeds nearer the root meet a larger chord section. There is a cyclic effect in improving the span-wise lift distribution, since the loading on and hence the stresses in the blade are improved, reducing the strength required of the blade. Also, the effect of taper is to directly increase the structural strength of the blade, as the section properties become smaller toward the tip. An exponentially varying taper will yield ideal taper characteristics, but the correct linear taper, carefully chosen, will provide near-ideal characteristics.

## 12.8 Blade Sections

Suitable airfoil sections were chosen for the specific mission. Different sections were used for the inboard, outboard, and tip regions, and were blended into each other to provide an even lift distribution. The different sections, and their reasons for inclusion, are listed in Table 10. The actual annotated drawings are included in the next section.

*Table 10: Airfoil sections*

| **Location** | **Airfoil** | **t/c Ratio** | **Comments** |
|---|---|---|---|
| 0.05R–0.65R | RAE 9648 | 12% | Counters the pitching moment from the main lifting airfoil |
| 0.65R–0.85R | RAE 9645 | 12% | Main lifting airfoil; high lift coefficient (1.55); high pitching moment |
| 0.85R–1.00R | RAE 9634 | 8.5% | Low t/c means a higher drag divergence speed; cambered for a weak shock wave and low pitching moment |

## 12.9 Rotor Schematics

The complete design of the rotor and hub was made using CATIA v.5. Figure 36 shows a single blade, with the various airfoil sections described. Figure 37 shows the detail of the hub area; the aerodynamic fairing (shown in Figure 38) is removed for visual purposes. Figure 39 shows the folded hub. Folding, as already discussed, is a necessary means of minimising the helicopter's size, such that it may fit below the deck of a CVN-class carrier.

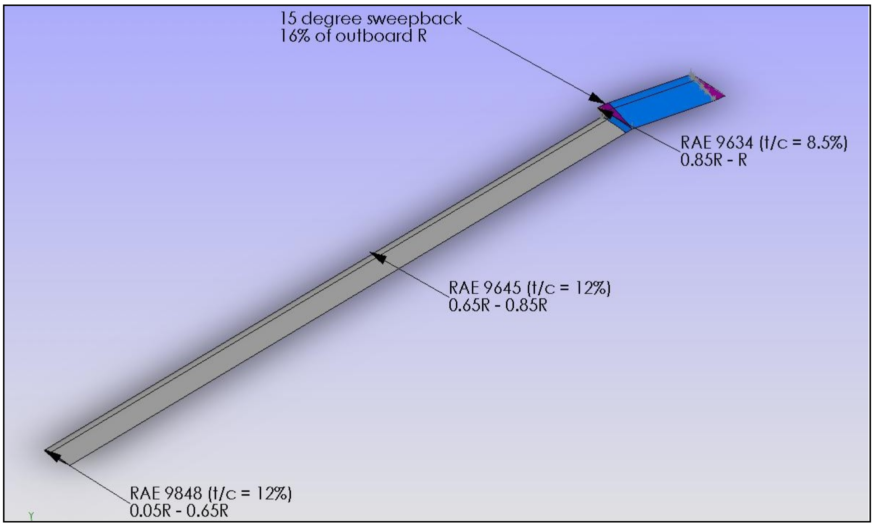


*Figure 36: Annotated blade planform*

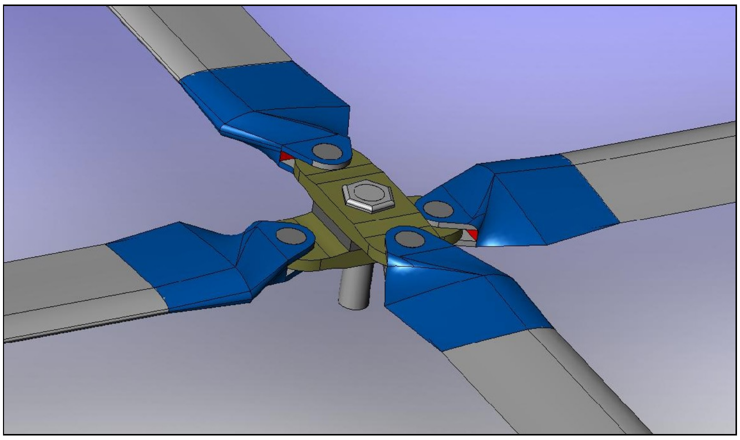

*Figure 37: Rotor hub, locked position (aerodynamic fairing not shown)*

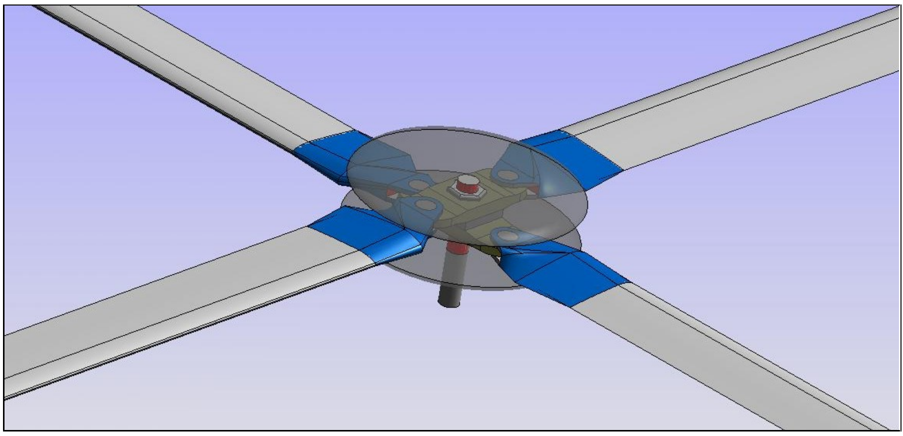

*Figure 38: Rotor hub, locked position, complete with aerodynamic fairing*

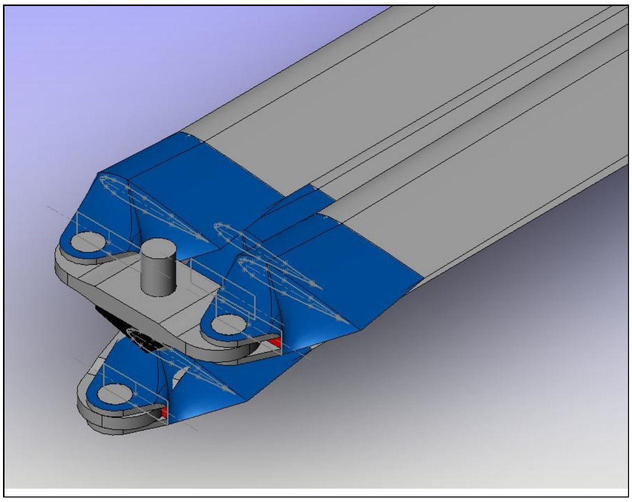

*Figure 39: Folded rotor hub (aerodynamic fairing not shown)*

# 13ROTORCRAFT DRAWINGS

The summary of the overall structure of the Cronus heavy-lift rotorcraft is best made using the following CATIA drawings. The various design options and configurations have all been combined, and an aerodynamically smooth fuselage has been made to integrate them into. The captions under each drawing adequately describe each of the drawings.

The engine intakes are visible under the rear rotor, one for each engine. The smoothed pods along the side of the fuselage are for fuel storage, and are armour-plated as described in Section 8.2, being a mission-critical sub-system. The high-visibility cockpit has as large as possible a ratio of windows to skin. Some of these windows include integrated mission displays.

Fuselage frames are also shown in detail. These frames are necessary to stiffen the fuselage as the fully-loaded rotorcraft performs manoeuvres. The main and nose undercarriage are clearly visible, being unretractable. Supports for the main undercarriage run right through the walls of the fuselage to the main structural ceiling. The drive train also runs through the ceiling, from the engines at the rear to the forward rotor hub.

In relation to the available space, the Cronus uses 91% of the available height, and 79% of the available length, when the rotors are folded.

A particular design aspect of the Cronus, mentioned in Section 9.2, is the open fuselage floor. This design options means that the load/unload time is reduced. The FCS vehicle has the option of being driven onboard, or, if terrain does not permit, may be winched up through the floor. Such adaptability will be increasingly necessary in the net centric combat world.

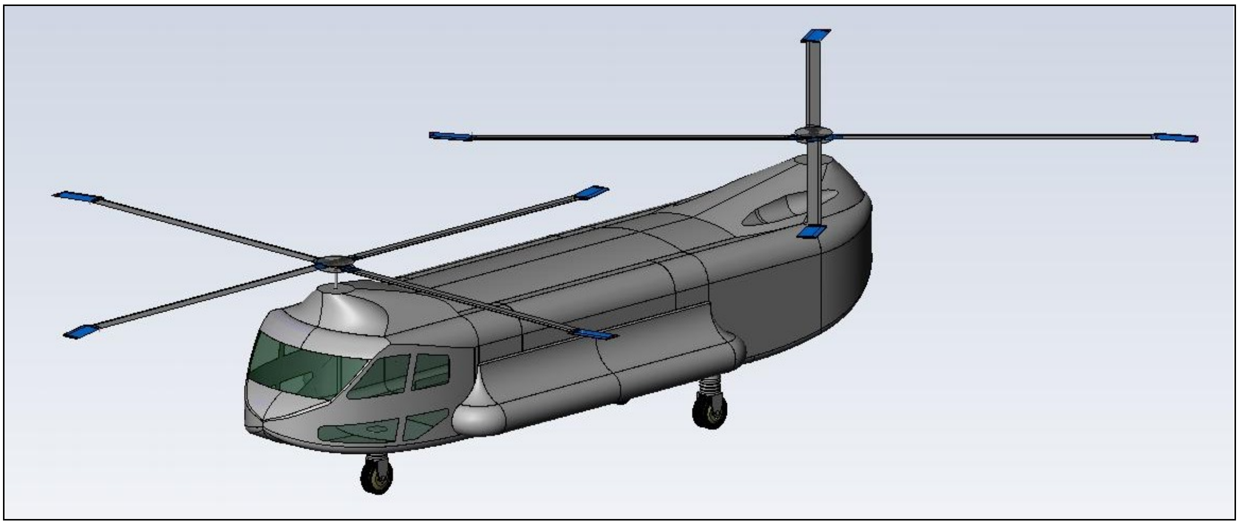

*Figure 40: Trimetric view of the entire rotorcraft*

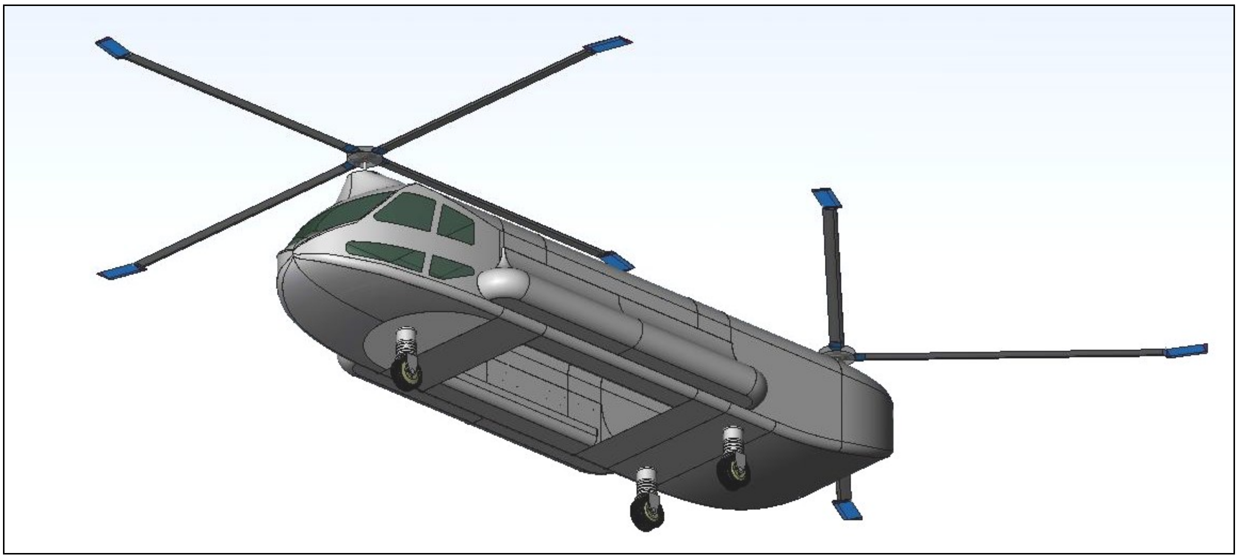

*Figure 41: Cargo bay, showing the open floor design, allowing for quick payload release*

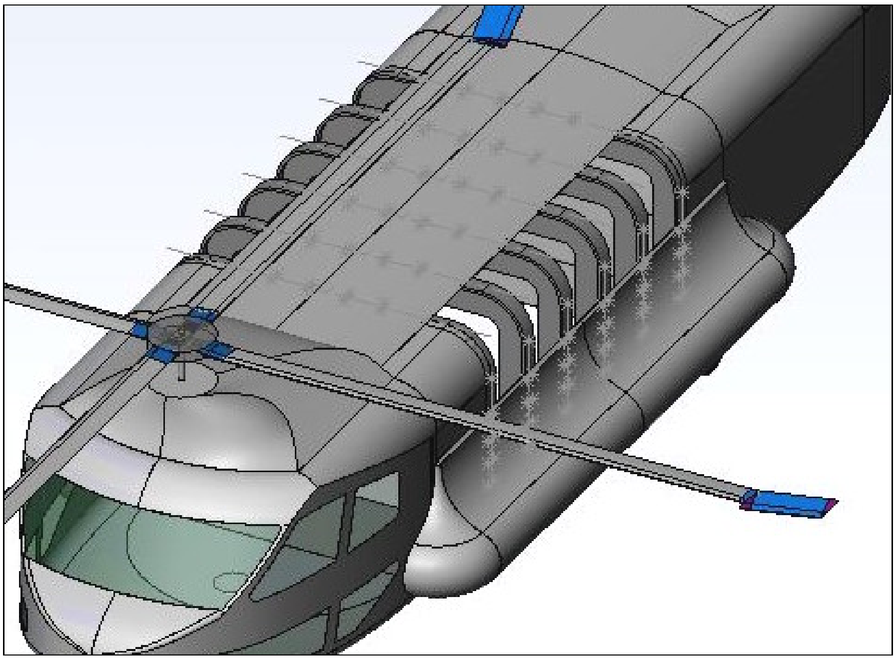

*Figure 42: Cutaway showing fuselage frames, required to stiffen the payload bay*

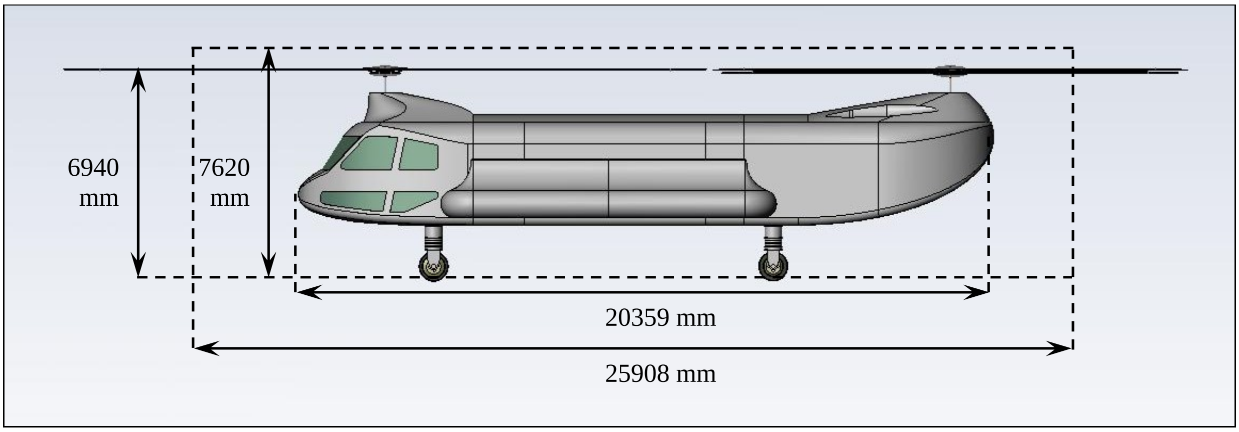


*Figure 43: Height and length, compared to the CVN elevator dimensions*

# 14CONCLUDING REMARKS

The use of the nine-stage preliminary design process has led to the conceptual design of a feasible and likely heavy-lift rotorcraft. It is envisioned that this vehicle will function superbly in a design niche that is yet unfulfilled. The primary mission of FCS vehicle delivery may be satisfactorily performed under any reasonable conditions, and alternate missions, such as intra-theater troop delivery, or humanitarian operations, are also achievable with minimal changes.

The design and report have been completed during the first semester of 2005, in the space of three months. Due to the short time-frame available, certain considerations have not been covered. In the second semester, issues such as auto-rotation performance, costing, detailed fuselage measurements, fuselage structural information, and cockpit design may be looked at. The design will be further developed, and improved iterations will be made of the weight and balance, and performance under various conditions.

The Cronus would make an entirely feasible and valuable addition to the future plans of the United States military, and other bodies worldwide. It is expected that the further design work mentioned above will only confirm and improve what is already an competent design.

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# 16APPENDIX A – PRELIMINARY DESIGN FORMULAE

The following formulae were used in Section 6 to perform iterations.

Stage 2 – Fuel Estimation

$$W_f = sfc \times P_i \times T_m$$

Stage 3 – Weight

Useful weight, $W_u = \sum W_c + \sum W_{ms(\mathrm{int})} + \sum W_{ms(ext)} + \sum W_p + W_f$

Gross weight , $W_g = \dfrac{W_u}{W_u / W_g}$

The $W_u / W_g$ factor is read from a graph (chronological ratio of useful weight to gross weight (Prouty, 2002))

Stage 4 – Vertical Drag

$$D_V = 0.3 \times D_L \times A_p$$

$$D_V = 0.04 \times MTOW$$

Stage 5 – Main Rotor Design

Disk Loading, $D_L = \dfrac{L}{A_D} = \dfrac{W + D_V}{A_D}$

Tip speed / advance ratio, $\mu = \dfrac{V_f}{\Omega R}$, where $V_T = \Omega R$

Solidity, $\sigma = \dfrac{A_b}{A_D} = \dfrac{bc}{\pi R}$

Thrust coefficient, $\frac{C_T}{\sigma} = \frac{T}{\rho\sigma A_D (\Omega R)^2} = \frac{W_g + D_V}{\rho\sigma A_D (\Omega R)^2}$

Stage 6 – Drag in Forward Flight

---

Parasite drag, $C_{dp(clean)} = k\left(\frac{(W_g - W_{ms(ext)})}{W_g}\right)^{\frac{2}{3}}$, $C_{dp(ext)} = 1.15 \times \lambda_f C_f$

$$\lambda_f = 1 + 1.5\left(\frac{d}{1}\right)^{1.5} + 7\left(\frac{d}{1}\right)^3$$

Form Factor, $C_f = 1.3\,\mathrm{Re}^{-0.5}$ $for\_\mathrm{Re} < 10^5$

$C_f = 0.45(\log \mathrm{Re})^{-2.5}$ $for\_\mathrm{Re} > 10^5$

$$C_{dp} = C_{dp(clean)} + C_{dp(ext)}$$

Profile drag, Method 1

Average angle of attack, $\alpha_{av} = \frac{6C_T/\sigma}{a}$

$M_{(0.75)} = \frac{2\pi R \times 0.75 \times \Omega}{60 \times a}$, where $\Omega = \frac{V_T}{R}$

Profile Drag, Method 2

$$C_{do} = \delta_0 + \delta_1 \alpha_{av} + \delta_2 \alpha_{av}^2$$

Profile Drag, Method 3

$$C_{do} = \delta + \sum \alpha_{av}^2$$

Stage 7 – Power Required for Forward Flight

---

Power required sustaining forward flight at maximum speed

This is equal to power available

$$P_f = P_{if} + P_{of} + P_p + P_m$$
$$P_f = \frac{W_g^2}{2\rho A_d V_f} + \frac{\sigma C_{do} \rho A_d (\Omega r)^3 \left(1 + 4.65\mu^2\right)}{8} + \frac{\rho C_{dp} A_d V_f^3}{2} + P_m$$

Stage 8 – Refinement of Gross Weight

$$cg = \frac{\sum M_{gp} + \sum M_{comp}}{\sum W_{gp} + \sum W_{comp}}$$

Miscellaneous

Disk Loading and Vertical Drag, $D_L = \frac{L}{A_D} = \frac{W + D_V}{A_D}$

Disk loading varies from 19.53 – 58.59 $kg/m^2$

Rate of Climb, $ROC_{\max} = \frac{P_a - P_c}{W_g}$

Minimum Power required for hover, $P_h = \frac{T^{\frac{3}{2}}}{FM\left(2\rho A_d\right)^{\frac{1}{2}}}$, where FM ranges from 0.75 – 0.80