3. Aircraft Conceptual Design

3.1. "Conceptual" Design

Ulrich and Eppinger define the initial stage of design as "concept development," during which "... the needs of the target market are identified, alternative product concepts are generated and evaluated, and a single concept is selected for further development." Aircraft "conceptual design" encompasses these activities in addition to some initial aspects of system-level design. However, the normal terminology of "conceptual," "preliminary" and "detail" design phases of aircraft development correspond roughly to Ulrich and Eppinger's "concept development," "system-level" design and "detail" design, respectively.

3.1.1. Identifying Customer Needs/Establishing Product Specifications—RFP's

An aircraft is a huge investment from the point of view of both the purchaser and the manufacturer. It is thus imperative that the product meet customer needs before it emerges from the design process. It is indeed customary for commercial airframe manufacturers to have purchase commitments from several airlines before making a decision to continue into production of an aircraft type. Large airlines often participate extensively in determining specifications for a new aircraft design by virtue of their up-front commitment. Thus the processes of identifying the customer needs and establishing product specifications are often combined into one. Note that smaller airlines will generally have less input into the design of an aircraft and are usually left to select from among those models designed for their larger competitors.

The military, on the other hand, will often create its own specifications in the form of a "Request for Proposal," or RFP. The RFP invites manufacturers to *propose* a design meeting the specified requirements. Each proposal is evaluated and, usually, one is selected, and that manufacturer becomes the sole provider for the aircraft. Note that there is not an alternative F-18 (at least, not in this country). It would simply be too expensive for a company to design and build a competing aircraft in the hope that the Navy (the single domestic customer) might change its mind and prefer a different vehicle.

One way or another, a set of specifications is usually presented to the aircraft design team as a *fait accompli*. Nevertheless, one should always scrutinize the requirements carefully. It may be possible to relax certain criteria if large gains can be made in another area by doing so. For example, if a small reduction in maneuvering capability can result in significantly lower cost, it is likely that the customer might seriously entertain a suggestion for altering the original requirement.

It will be instructive to use a representative RFP throughout the discussion of the aircraft design process. These notes will refer to the *1998 RFP for Civil Transport Rotorcraft* developed by Boeing and the American Helicopter Society for the Annual Student Design Competition. Since it is intended for students, this RFP does not request as much detail in the design as an actual military RFP would do. It also provides some information that the military version would not. Nonetheless, the document provides a set of specifications for the proposal and for the aircraft, both of which are representative of a military RFP.

Note that section 1 of the RFP describes rules of the competition, a schedule of due dates and requirements for the proposal itself. Though engineers tend to be concerned primarily with technical aspects of the design, these considerations are equally important to the success of the proposal which must be legal, on time, and in the format specified.

3.1.2. RFP Overview

The AHS request for proposal describes the requirements for a vertical take-off and landing commuter aircraft in both 12- and 19-seat configurations. The two variations of the aircraft must have "substantial" commonality, meaning that most of the parts must be the same for both. The RFP specifies that the aircraft must be designed in accordance with FAR Part 29 and FAR Part 25. These are airworthiness standards of the Federal Aviation Regulations pertaining to transport category rotorcraft (Part 29) and transport category aircraft (Part 25). There is a part pertaining to commuter aircraft (Part 23), but it is restricted to aircraft with gross weight less than or equal to 12,500 lbs. Though it is not yet obvious, a vehicle capable of meeting the mission requirements of the RFP will have gross weight considerably higher than the Part 23 limit. One difference between these parts is that under Part 25 operating conditions, a flight attendant is required for 10 or more passengers, while under Part 23 none is necessary if the aircraft holds 19 passengers or fewer. Other differences between the two include design limit load factors, takeoff and landing distance specifications and climb-gradient minimums. Some of these regulations are not relevant to this aircraft as it takes off and lands vertically. Federal aviation regulations are available at:

http://www.faa.gov

The longest segment distance required of the Civil Transport Rotorcraft is 200 Nmi; no required top speed is given, though the design mission is to be flown at speed for minimum direct operating cost (DOC). Airline costs are divided into *direct* and *indirect* operating costs. As might be guessed, direct costs are those associated with operation of the aircraft and include fuel and oil, crew salary, maintenance, depreciation, and insurance. DOC is usually given in cost per seat-mile flown so that it gives some measure of the productivity of the airplane. Fuel costs can be directly related to distance flown; flight crews are paid per block hour which includes not only the flight time but the time spent in the airplane on the ground. Pilots are also paid according to the weight of the aircraft they are flying. Maintenance is performed after a certain number of flight hours and so can be related to seat miles as well. Depreciation and insurance are difficult to estimate without knowing the initial cost of the aircraft, but can be considered to be higher for larger aircraft and to increase linearly with the number of aircraft in use.

Airline Operating Costs	
Direct Operating Cost	Indirect Operating Cost
Fuel	Aircraft ground handling
Depreciation	Landing fees
Maintenance	Aircraft service
Cockpit crew	Cabin attendants
Insurance	Food and beverage
	Passenger handling
	Reservations and sales
	Baggage and cargo handling
	Advertising
	Administrative and overhead
	Ground facilities and equipment

Table 3.1: Airline direct and indirect operating cost.

Each airline and each aircraft manufacturer has its own model for predicting DOC for a given aircraft. These models are usually proprietary and not available for outside use. Other cost models exist; there is one in Raymer's book, but it is outdated. It is impossible to simply add an inflation factor to it since costs for the various items have increased at different rates. It is therefore difficult for students to estimate cost, though it is still a valuable exercise to consider it since the success or failure of a design, particularly a commercial one, will depend largely on its operating cost relative to its competition. For purposes of this study, we will use a standard measure of productivity given by

$$P = \frac{1}{N_{ac}} \frac{W_{payload} * V_{block}}{(W_{empty} + W_{fuel})}$$

In this equation, N_{ac} is the fleet size, $W_{payload}$, W_{fuel} and W_{empty} are payload weight, fuel weight and empty weight, respectively, and V_{block} is the block speed or number of miles flown for a typical mission divided by the total block time for that mission. Note that our RFP specifies an additional 15 minutes on the ground for refueling and passenger loading and unloading. A high value for P indicates high productivity and, therefore, lower direct operating cost.

Note that the specifications indicate that the fleet must be capable of flying 700,000 passenger miles per sixteen-hour day based on a 70% load factor. This *load factor* refers to the capacity of the aircraft, so that the 12-passenger aircraft carries an average of 8.4 passengers on each flight. The number of aircraft required to fly this number of miles depends on the speed at which they fly. There is therefore a tradeoff between aircraft speed and number of aircraft—higher speed will cost more in terms of fuel used, but a larger number of aircraft will cost more in terms of crew salaries, maintenance, and depreciation.

The proposal must include estimates of acquisition cost as well as direct and indirect operating cost. Raymer's book includes a methodology for estimating development cost (directly related to acquisition cost). Again, the reliability of this method is questionable, especially for this unusual type of aircraft. Indirect operating costs refer to the costs of buildings, advertising, sales and customer service, administration and overhead. IOC's vary widely from airline to airline and are nearly impossible to determine statistically. Raymer suggests setting IOC equal to the DOC as a means for obtaining a rough estimate.

The RFP specifies some capabilities beyond simply meeting the mission requirements. Some clarification of these follows:

1. Autorotation Capability—Rotorcraft are usually required to autorotate to a safe landing in the event of a power failure. This requirement has implications for the rotor design. The requirement is not specific, and should be clarified to specify under which conditions (speed, altitude) this requirement must be met.

2. Folding Rotor Blades—A conceptual design should indicate how the blades will fold and how they will lock in place. Folding blades are generally heavier and more costly than non-folding.

3. Adequate Seat Pitch—The FAR's do not specify seat pitch (except at an emergency exit row) or seat width. The RFP requires that the passenger comfort be equivalent to competing

modes of transportation, which probably means commuter aircraft. Typically, the seat pitch for commuter aircraft is from 30–32 inches, and the seat width is 16–18 inches. FAR Part 25 specifies aisle width to be at least 12" at floor level and at least 20" at a height 25 inches above the floor. FAR's also specify that, for this size aircraft, there should be at least two type III emergency exits (20" wide by 36" high and less than 20" above the floor), one on each side of the fuselage.

4. Flight Envelope—The flight envelope, or V-n diagram, is determined according to regulations in the FAR's. Gust and maneuvering loads are considered, as well as loads on flaps and control surfaces. The regulations provide a method to determine the maximum limit load factor. The structure is designed to the ultimate load factor which has a factor of safety of 1.5 above the limit load. Transient turn capability is generally not considered to put a power or thrust constraint on the aircraft, but is intended to impose a maneuvering load constraint at a given airspeed. In this RFP, the transient turn capability requirement seems to be imposing an aerodynamic constraint—*i.e.,* are the aerodynamic surfaces capable of generating the load factor ($n = 1.15$) required to enter the 30° banked turn without stalling when the aircraft is in cruise? This is normally not a problem for conventional aircraft, but could be a problem for a helicopter, which generates both lift and thrust with the rotor.

5. One-Engine-Inoperative Hover—This requirement will probably size the engines. FAR Part 29 does specify engine-out climb requirements, but these can be satisfied with some forward velocity which will require less power than vertical flight.

6. Two-person flight crew—This does not include flight attendants. As mentioned above, FAR Part 25 requires one flight attendant for aircraft over 7500 lbs and with 10 to 51 passengers.

7. Pressurization—It should probably be assumed that this aircraft will be pressurized. This has implications for the fuselage design, making the cross-section circular, "double circular," or elliptical.

8. Production Quantity—This is required in order to determine development cost. The RFP is not clear, however, and the proposer may request clarification from the agency.

9. Maintainability—This is just one of the "ilities" which have become fashionable in the industry. Fashion or not, it is good common sense to design an aircraft to be easily maintainable

in order to reduce maintenance costs. Another major factor contributing to increased maintainability is a reduced parts count. In other words, use the same rivet in more than one place if you can.

10. Crashworthiness—This is a complex aircraft and should be designed with survivability in mind. Regulations are given in FAR Part 29.

11. Low Noise—Helicopters tend to be noisy both from the community-noise standpoint and inside the cabin. Any helicopter-like vehicle will exhibit characteristic noise patterns of a rotorcraft. The noise level can be reduced, not only by reducing the tip Mach number, but by observing other design rules for the rotor. These include keeping the thickness low at the rotor tips and designing the rotor so that the outboard portion does not carry as much of the aerodynamic load. Unfortunately, design for low noise tends to drive the rotor in the opposite direction to design for high efficiency. This illustrates another important design tradeoff.

RFP's are usually filled with jargon and acronyms. Some of those used here include:

KEAS: knots equivalent airspeed

HOGE: hover out-of-ground-effect

MCP: maximum continuous power

IRP: Intermediate rated power

OEI: One engine inoperative

IFR: Instrument flight rules

SHP: Shaft horsepower

IHPTET: Integrated, High-Performance, Turbine-Engine Technology

MIL-STD: Military standards

3.2. Aircraft Sizing

3.2.1. Configuration Selection

The initial phase of design for any aircraft involves 1) the configuration selection and 2) sizing of major components such as fuselage, wings, tails, powerplants and, in the case of rotorcraft, rotor systems. The class of aircraft known as "High-Speed Rotorcraft" is relatively new, and configuration selection can require several sizing exercises in order to adequately evaluate each possible configuration. The selection matrix represents one alternative for methodically evaluating the relative merits of each candidate. Each possibility should be assessed in several categories including, but not necessarily limited to:

- Ability to meet RFP requirements
  - Vertical takeoff and landing (must)
  - OEI hover
  - Growth capability
- Ability to be operated safely
- Low operating cost
  - Speed capability
  - Cruise efficiency
  - Hover efficiency
- Low technical risk
  - Minimal new technology research and development
  - Mechanical simplicity
  - Conversion simplicity
- Acceptance
  - Smoothness of conversion
  - Appearance

In coming up with a list of evaluation criteria, care should be taken to not cover the same criterion more than once. For each consideration, such as cruise efficiency, for example, it is necessary to decide what is the "real" issue—in this case, the fuel used and, ultimately, the operating cost is the governing concern. Criteria that *must* be met should be listed first. If a configuration type cannot meet the RFP requirements or if it cannot be operated in a safe manner, it should be eliminated from consideration, and no further evaluation is necessary. There exist any number of "correct" categorizations for the selection criteria—what is important is that each is clearly defined and that each contributes only once to the total score.

Each evaluation category should be assigned a relative weight. For example, safety is likely of primary concern and may receive a weighting factor of 10. Since operating cost will determine whether or not the aircraft wins the competition, it should rate close to safety in importance. Acceptance is surely important, but relative to other categories it may receive a low ranking. Absolute values for the weighting factors are not of particular importance as long as their relative values are consistent. A scale of 10 for the weights is purely arbitrary.

Each candidate configuration should then be evaluated in each category. This may not be possible *a priori* for all configurations, particularly when novel designs are under consideration. In an industrial setting, some analysis is required in order to properly assign a score to each configuration. In the classroom, and to some extent even in industry, some extrapolation will take place at this point in order to narrow the field of "good" or even feasible designs. However, scoring should be based on quantitative models whenever possible and should *always* be objective. A truly complete design process will allow for more than one possible configuration through at least the conceptual phase. It will often be the case, however, that the configuration is, in fact, determined before any neutral selection process is attempted. (It would be surprising if Bell or Boeing (PA) were to select anything but a tilt rotor for the Civil Transport Rotorcraft mission. This is not necessarily the incorrect answer, particularly for a company with a large database on and much experience in tilt-rotor construction and design. But it should always be kept in mind that there may be more than one solution and that some other way could be better!)

3.2.2. Sizing Methodology

The aim of the sizing exercise is to determine to within some limits the size and weight of the aircraft that will meet the mission objectives and to determine overall design parameters such as wing loading, disk loading, power required, rotor tip speed, wing aspect ratio, *etc.* At this stage, details of the construction are not known so that heavy reliance is placed on statistical formulas for component weights. Relatively simple analysis techniques are used to determine aerodynamic characteristics such as aircraft drag, maximum lift coefficient and lift-curve slope of wings and/or rotor blades, and stability and trim parameters. Often actual engine data (termed an engine "deck") is not available since the engine selection has not yet taken place. In such cases, data for similar engine types is scaled to provide estimates of power or thrust available and fuel consumption as functions of altitude, velocity and throttle setting.

Fuselage Size (or Where do you Start?)

The overall size and shape of the fuselage is determined by the payload. In the case of the Civil Transport Rotorcraft, the fuselage must hold 12 (or 19) passengers and their luggage along with two flight crew. A flight attendant is also required since the aircraft must be certified under FAR Part 25. Standards for aisle width, seat width and pitch and pilot seat size are available and can be used to estimate the interior dimensions of the fuselage. Decisions such as whether to place passengers two-abreast, three-abreast or in some other arrangement need also be made. Because of the nature of the current mission and payload, the fuselage will likely more or less resemble that of a typical commuter aircraft rather than that of the conventional helicopter or large transport.

As an example, consider a 30" seat pitch and 18" seat width in a three-abreast configuration. The total aisle width is 20". The minimum fuselage inner diameter to accomodate this interior comes out to about $6\frac{1}{2}$ feet. According to Raymer, at least 2–8 inches are needed for insulation and other space between inner wall and outer skin so that an outer diameter of 7' is realistic. For the 12-passenger configuration, four rows of seats are necessary. These four rows plus additional space for an entry door, a small galley and, perhaps, a lavatory result in 13-foot-long cabin section. The luggage compartment will be located behind the passengers. That, enough space for the cockpit, and tapered sections at the tail and nose will result in a fuselage length of 41 ft.

The *fineness ratio* (length/average diameter) of this configuration comes out to approximately 5.86, within the range for reasonable tradeoff between pressure drag and skin-friction drag. Some guidelines for the fuselage design include:

- The overall fineness ratio should be at least 4.
- The fineness ratio of the aft fuselage (converging section) should be approximately 3 with maximum upsweep angle less than 14°.
- The optimal fineness ratio of the nose section depends on the aircraft design Mach number. The fineness ratio should be at least 1 regardless of Mach number. For design Mach numbers

between 0.76 and 0.9, the ideal nose fineness ratio can be obtained from:

$$\frac{\ell}{d} = 74.7714M^2 - 113.0765M + 43.7671 \tag{3.1}$$

As long as the above guidelines are met, the fineness ratio should be minimized if low drag is the only goal. Of additional primary concern is the moment arm from aircraft center of gravity to the horizontal and vertical tail surfaces. Short moment arms lead to large tail areas in order to meet stability and control-power requirements. Normally the minimum fineness ratio should be about 6 to ensure reasonably sized tails and control surfaces.

For the 19-passenger version of the aircraft, a fuselage "plug" of about 5 ft. can be added which extends the fineness ratio to a value of 6.57. The "basic" fuselage would then have an equivalent flat-plate drag area of about 2.4 ft^2 while the extended fuselage would have $f \approx 2.8$ ft^2. Note that these estimates are based on a standard drag build-up method outlined in Chapter 1.

Very little else about the aircraft is known at this point. Even the fuselage weight cannot yet be determined since the loads experienced by the fuselage structure (and therefore the weight of that structure) depend on the maximum weight of the aircraft. Consequently, the usual next step consists of guessing—experience will often be invoked at this time in order to guess values within reason. The objective of the succeeding sizing exercise is to refine the design to optimal values of the parameters, originally guessed, but now selected with conviction. The first guess to be made is of the aircraft design gross weight (GW).

Reducing the Number of Parameters

There are, of course, an extremely large number of variables involved in the design of an aircraft. Only a relative few of these are needed to arrive at an initial "size" for the vehicle. For a fixed-wing airplane, these important primary variables normally include the wing loading, power or thrust loading, wing aspect ratio and wing sweep and thickness. For a helicopter, these are disk loading, power loading (or power-to-weight ratio), solidity, number of blades and rotor hover tip speed. Sizing of our hybrid aircraft will likely require consideration of all of these variables and, perhaps, a forward-flight tip speed as well. (It may be decided that the rotor should spin at one Ω in hover and at another in forward flight as does the V-22 tilt rotor.) Note that with this number of variables specified, one can obtain a very good idea of the aircraft size, shape, engine power or thrust rating and weight.

Sometimes the variable values, or relationships between two or more of them, can be determined simply by meeting one or more of the prescribed design constraints. Consider two examples for aircraft types that may meet the Civil High-Speed Rotorcraft specifications.

A compound helicopter has a rotor (like that of a helicopter) and a wing. In hover and low-speed flight, the rotor provides lifting and propulsive forces and it acts as a helicopter. As the speed increases, the wing takes on more and more of the required lifting force, and the rotor force is shared between lift and propulsion. In some cases, additional thrust is provided by turbofan engines in high-speed flight. The designer would prefer to specify the smallest possible wing in order that its adverse effects during low-speed flight and hover (effective increase in weight from the rotor wake impinging on the wing) be minimized. At some speed, however, it will be necessary that the wing take on a significant proportion of the aircraft lift and, if the cruise speed is high enough, it must support the entire weight of the vehicle. There is some maximum lift coefficient that a wing is able to generate. ($C_{L_{\max}}$ will depend on the high-lift system selected, yet another design decision.) Given $C_{L_{\max}}$ and the speed at which the wing must generate all the lift and a suitable safety factor, a maximum wing loading can be determined from

$$\frac{W}{S} = \frac{1}{2}\rho v_W^2 \, C_{L_{\max}} \tag{3.2}$$

where v_W is the speed at which the wing will take on the aircraft weight. It should include a safety factor; it is usual to always remain at a velocity at least 30% higher than the stall speed in cruise configuration. Eq. 3.2 establishes a relationship between design wing loading and the velocity at which the wing must carry the aircraft weight.

The value of v_w will depend on the rotor tip speed. This comes about because of the reduced lifting capability of a rotor at high advance ratio, $\mu = v_\infty / V_{\text{tip}}$. Generally speaking, an advance ratio of 0.5 is the absolute maximum possible for a conventional helicopter—beyond that there must be some auxilliary lift. As illustrated in figure **??**, at an advance ratio of 1.0, the lift produced on the retreating blade is virtually zero since it is entirely within the reverse flow region. Given the rotor tip speed and the advance ratio at which we would like the wing to support the weight, v_w and, thus, the wing loading can be specified. For a given (or guessed) gross weight, the wing area is also known. For example, if the tip speed is 600 ft/s, the advance ratio will reach 0.5 when v_w = 300 ft/sec. The wing stall speed should then be 300/1.3 = 231 ft/sec. Assuming a cruise maximum lift coefficient of 1.5, the maximum wing loading will be about 95 lb/ft^2. (Sea-level density was used in this calculation).

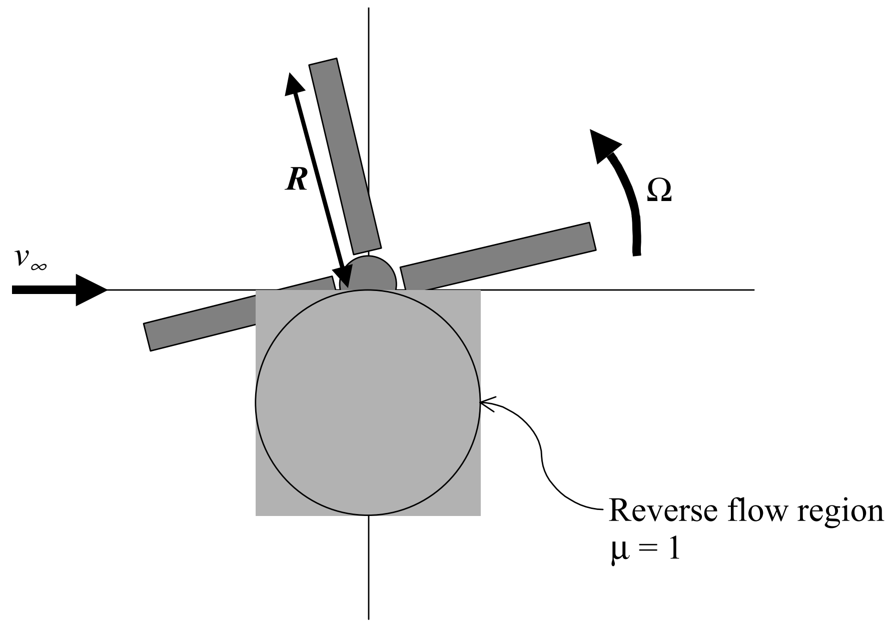

Figure 3.1: Helicopter rotor top view showing reverse flow region for advance ratio of 1.

Other initially non-obvious relationships may exist between the parameters. As an example, consider the design of a tilt-rotor aircraft. In this case, the rotor radius is restricted to be less than the wing half-span by an amount determined by the fuselage width. This places an upper limit on the disk area (or a lower limit on the wing area for a given aspect ratio), so that the disk loading and wing loading are related. Such a relationship reduces by one the number of primary design variables and, thus, reduces the size of the "design space." Finding these relationships is important not only to make the problem faced by the designer smaller, but also to make certain that the design is feasible.

The combination of power-to-weight ratio (P/W) and disk loading (W/A) is usually determined by a maximum speed, a minimum vertical-rate-of-climb (VROC) or a one-engine-inoperative (OEI) hover constraint. Sometimes a maneuvering constraint will prevail. Generally, it is impossible to specify one or the other parameter independently.

Sizing Procedure

A somewhat simplified procedure (but including the main ideas) for sizing the rotorcraft proceeds as follows and is illustrated in figure 3.2:

1. Supply fuselage, payload data and any fixed weights. Also choose some "standard" values for

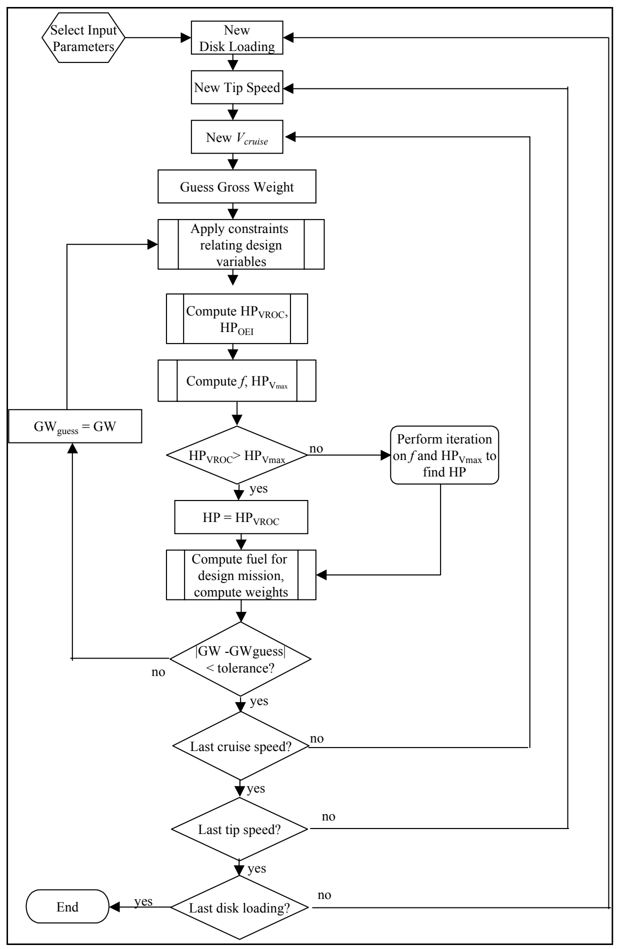

Figure 3.2: Sizing procedure for Civil Transport Rotorcraft.

certain parameters based on typical values for the type of aircraft. These may include Æ_W, wing thickness, number of rotor blades, rotor-blade maximum lift coefficient, *etc.* You will probably want to go back and investigate these further once the initial sizing is complete.

2. *Specify* design variables: V_{tip} (you may need a hover and a forward-flight value), W/A, and cruise speed. (For this particular RFP, you will need to use speed as a design parameter in order to select the speed for minimum DOC.) In class, you will probably not choose more than three initial design variables since this represents about sixty or so aircraft you will be considering.

3. *Guess* aircraft gross weight (GW).

4. *Apply* constraints, such as V_W, or relationships among design parameters to size the wing. Use tail-volume coefficients to size the tails.

5. *Apply* further constraints (max speed, VROC, HOGE with OEI, *etc.)* to size the engines. Include engine drag and weight in the calculations. *(A little iteration will be required here.)*

6. Find component weights and drag using statistical and other methods.

7. "Fly" the aircraft through the design mission to determine the required fuel.

8. Determine the total weight of the aircraft equal to the sum of the component weights, the payload and crew weights and the fuel weight.

9. If the computed sum does not equal the guessed gross weight, use the newly calculated weight and return to step 4.

10. If the computed sum equals the guessed gross weight, specify a new parameter and return to step 3.

11. The exercise is completed when the desired number of parameters has been evaluated.

Once all the data is available, the "best" combination of parameters can be chosen to serve as the baseline for further design studies. Various levels of automation and sophistication are available, but, regardless of the number of variables included or the detail in the weight, drag or performance estimates, the above illustrates the basic idea of converging on a feasible, and then optimal, solution.

Design Goal

It remains to determine which of the feasible designs is the best. The answer depends on the objective of the design exercise. For aircraft as well as many other things, the goal is often to minimize cost. But cost can include several aspects, such as development cost, flyaway cost (purchase price), operating cost, *etc.* Most airframe manufacturers as well as users (airlines, the military) have aircraft cost models, based on statistics, for estimating the cost of owning and operating a particular aircraft. Unfortunately, up-to-date cost models are generally proprietary and difficult to obtain. Students designing aircraft are often restricted to using old and often irrelevant models or to simply minimizing gross weight as a measure of the cost of an aircraft.

Upon occasion, cost does not provide the main driver for the design of an aircraft. For example, because of the nature of the powerplant, the designer of a human-powered aircraft would likely wish to minimize power required to maintain level flight. (It is my understanding that the total energy output is not the problem (humans have the requisite energy reserves), but the energy expended per unit time is limiting (humans cannot apply a large enough force fast enough).)

3.2.3. Approximate Sizing Techniques

Standard methods for performance evaluation can be used to accomplish such tasks as determining fuel burn during various phases of the design mission or evaluating operational constraints. In order to do so, however, both the drag and the weight of the configuration must be estimated. Both quantities are complicated to calculate, and the weight, in particular, cannot be determined with certainty until the preliminary structural design is completed. However, standard methods, based on statistical evaluation of previous aircraft, are used in the early stages of the design process to estimate both the weight and the drag. (Note that the drag calculation is much more accurate at this stage since the outer dimensions are determined more specifically. Also, the drag methodology is more exact (maybe "refined" is a better word), and will probably be used exclusively to determine parasite drag unless actual tests are performed.) Appendix **??** gives examples of statistical weight equations for high-speed rotorcraft components.

Guidelines for Tail Sizing

The actual sizes of the vertical and horizontal tail surfaces must be determined using a thorough stability and control analysis. However, at this point in the design process, not enough is known about the aircraft to perform such a study, so the following guidelines are provided.

1. Unless information is known to more closely estimate the center of gravity, place the wing on the on the fuselage so that the aerodynamic center of the wing (0.25c of the mean chord) coincides longitudinally with the midpoint of the passenger (or cargo) compartment (wing-mounted engines), and with the 67% point of the payload compartment for rear-fuselage-mounted engines.

2. Tail surfaces must have a higher divergence Mach number than that of the wing. Therefore, for high-speed aircraft with cruise Mach numbers above about 0.6, tail sufaces should have more sweep (by about 5°) and/or lower thickness than the wing.

3. Tail surfaces should be placed as far aft as possible to maximize moment arm. It is favorable to place the horizontal tail so that it lies in the portion of the fuselage tail cone where the velocities are decreasing.

4. Aspect ratio of horizonatl tails is usually about 4. Since tail cruise lift coefficients are small, the tail induced drag is small.

5. Aspect ratio of vertical tails is 1.2 to 1.8 based on the exposed area.

6. Taper ratio of tails is usually 0.4 to 0.6. The taper ratio of a vertical surface of a T-tail will be closer to 1.0 to provide a large tip chord as structural support for the horizontal surface.

Once the basic geometry of the tails is set, it remains to determine the tail size. As an initial estimate, it is customary to use "tail-volume coefficients" to find tail areas. The horizontal-tail-volume coefficient is defined as:

$$V_H = \frac{S_H \ell_H}{S_W \, \bar{c}}$$

where S_H is the horizontal tail area and ℓ_H is the distance between the aircraft c.g. and the horizontal tail aerodynamic center. The vertical-tail-volume coefficient is given by

$$V_V = \frac{S_V \ell_V}{S_W \, b}$$

where S_V is the horizontal tail area and ℓ_V is the distance between the aircraft c.g. and the vertical tail aerodynamic center. The actual values of V_H and V_V depend on the type of aircraft under consideration, the allowed c.g. range and the tail design. Standard values for a commuter-type aircraft (given by *Raymer*) are $V_H = 0.9$ and $V_V = 0.08$. For T-tails, the coefficients can be reduced by 5%.

Engine Performance

As a rule, engine performance (thrust or power and fuel flow vs altitude and Mach number) should be determined using engine decks provided by the engine manufacturer. If these are not available, calculations based on thermodynamic and gasdynamic analysis can be used. The guidelines given in Chapter 5 are limited, but can be used if other information is not available.

One new type of engine which has been tested but which has never been in production is the "convertible" engine. This powerplant can act as both turboshaft and turbofan. The conversion between them is enacted through a series of clutches and shutters. As a rule of thumb, the thrust available under a set of atmospheric and operating conditions is approximately 1.2 times the available horsepower (*i.e.*, T (lbs) $= 1.2$ (lb/hp) $\times$ hp). Thus the thrust-specific fuel consumption becomes the power-specific fuel consumption divided by 1.2. (Thrust-specific fuel consumption has units of lbs fuel/ lb-hr.)