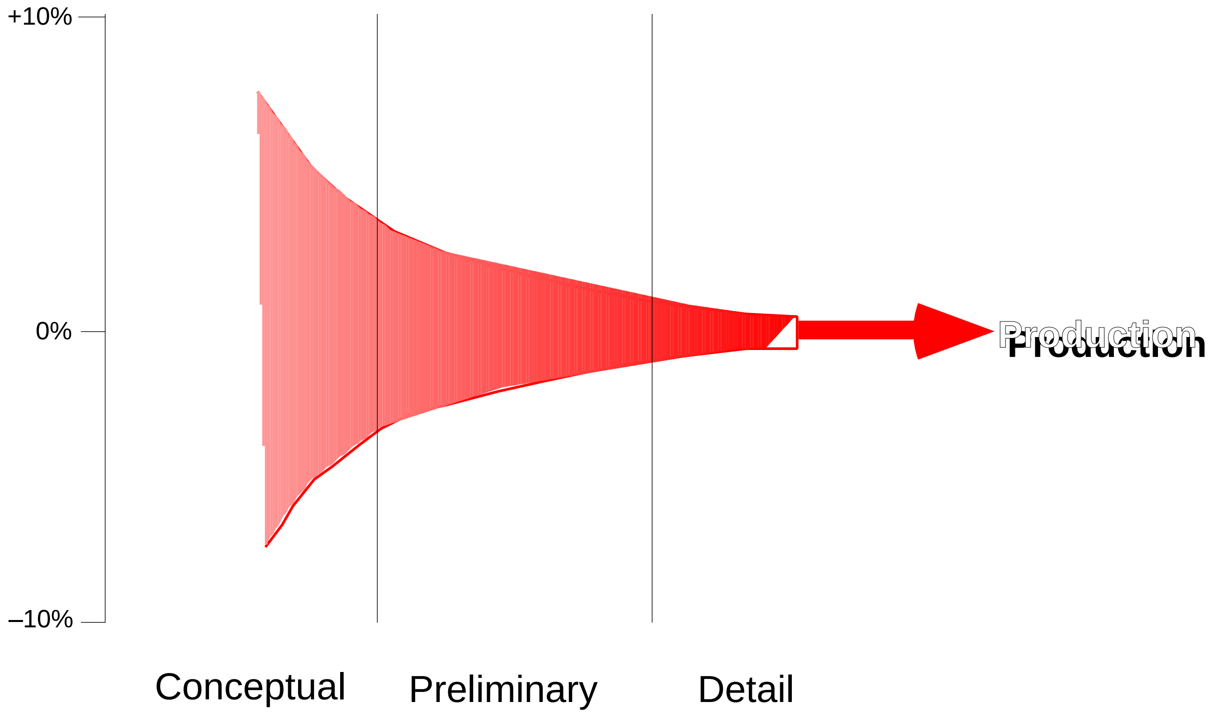
AIRCRAFT CONCEPTUAL AND PRELIMINARY DESIGN



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on

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DRAFT

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ABOUT THIS TEXT

This multi-volume series is a compendium of methods and mindsets applicable to aircraft conceptual and preliminary design projects. A wide variety of aircraft missions exist so several diverse missions are covered in this series. The first volume presents a general, philosophical approach to conceptual design. Examples are limited to one or two mission types. Material in the first volume will also include a section on how to make drawings, and how to investigate parametric alternatives. The first volume specifically includes chapters on:

- Reading an RFPor AO;
- Defining the design domain;
- First-cuts (zero level) at the basic disciplines;
- Doing basic parametrics, as in thinking through a design problem in general terms to arrive at a set of equations that can be explored to see what parameters drive the design;
- A lengthy section on presenting material in briefings and final reports including industry proposals; and
- Adecent and flexible costing method.

The next volumes are presented in the form of case studies which specifically address conceptual design and take a brief look at a variety of projects from the perspective of what was going on in the heads of the design team (if anything) as the configurations progressed.. The case studies are:

- Case #1: An Unlimited Class Reno Air Racer;
- Case #2: A High Altitude Long Endurance Sensor Aircraft;
- Case #3: A Very High Altitude Science Platform;
- Case #4: A V/STOL Transport;
- Case #5: A Science Platform for Mars; and
- Case #6: ASubsonic Regional Airliner.

Material will be added during the coming academic years to include case studies of the following:

- supersonic business aircraft;
- military trainers and multi-mission aircraft; and
- high performance military aircraft.

FOREWORD
Configurators and the Industry

Of the 25,000 or so aerospace engineers in the industry at any one time, roughly one percent are actively involved in advanced concepts work and could be labeled configurators. These individuals can be characterized as free-thinkers and people capable of finding innovative approaches to challenging mission scenarios. These are the people who see a glass of water as neither half empty nor half full, but twice as big as it needs to be. They're frequently not the best specialists in any detail-oriented discipline but they understand enough of specialized disciplines that they're capable of seeing subtle interactions which create opportunities for novel solutions. Configuration design is a mindset—perhaps even a worldview—a way of seeing the world in ways others may not in order to find the optimum solution to a design challenge.

Romantic images of an airplane designer at his or her creative best include Kelly Johnson sketching on the back of an envelope what would become the Lockheed P-38 Lightning, or Dutch Kindleberger offering a new design, just barely sketched on a cocktail napkin, to replace another company's existing fighter his company had just been asked to tool up to produce. Usually, airplanes are designed in a more mundane fashion by teams of individuals, all working at their drawing boards (or CAD stations) or computer terminals. The initial impetus for a conceptual design study may be nothing more than a verbal or written request from the chief engineer to examine a set of mission requirements for a government or civilian sponsor. Dozens, or even hundreds of configurations may be examined before a company's management decides to turn one into a formal proposal.

Recommended Course Content

There's nothing magic about the content of aircraft design courses or about the way in which material is presented. There is something magic that happens when students who are completing their technical curricula are exposed to design courses where they're expected to integrate the knowledge they've acquired in other schoolwork. If design courses are presented well, students will gain an appreciation of what engineering really is. The companies they join after graduation will then be able to mold them into productive staff members more quickly than if they not had design courses.

The task set for the teacher, then, is to guide the beginning configurator through the process, to instill an appreciation of how much acquired aeronautical knowledge the young configurator already possesses, to pass on new knowledge, and, perhaps most importantly, to convey the successful configurator's attitude or mindset in the form of the teacher's "seat-of-the-pants" feel for potential design problem areas and for engineering tradeoffs. This includes teaching an understanding of what constitutes an acceptably accurate answer as well as how much work really needs to be done at each level to get that acceptably accurate answer. The axiom, "measure it with a micrometer, mark it with a grease pencil and cut it with an axe" typifies the engineering approach early in conceptual design work.

A word about the phases of the design process is in order here. The design process for any mechanical device can be divided into stages based upon the degree of detail involved in the stage. In industry, the stage from initial conception through a proposal is called conceptual design. From proposal acceptance to manufacturing approval the device is said to be in the preliminary design stage. The final stage is called detailed design and takes place from manufacturing approval to initial production. The design process continues after initial production to incorporate changes dictated from operations or to incorporate changes in state-of-the-art of components. The first course in a design sequence should address conceptual design and the next courses in the sequence should address preliminary design with the preliminary design phase being defined as lasting from the selection of a single configuration to proposal submission.

It goes without saying (but we'll say it anyway) that design courses should be carefully structured to introduce students to real-world engineering problems where there are time constraints and no obvious right answers. The approach taken here will be to start with simple conceptual design problems such as determining mission performance and weights of one or more aircraft. This will lead to creating individual configurations as a response to mission statements. Students will then analyze these configurations and will be amazed as they compare answers with their peers at how many different approaches there are to "acceptable answers".

For a multi-course design sequence, work during the first course should involve design drills to reinforce their mastery of the conceptual design process. Configuration studies should be increasingly complex and may or may not be limited to individual work. The decision of when to begin working in groups should depend upon how well students learn initial lessons and how they handle their workloads. One very important lesson every engineer must learn early on (but all too few do) is how much time to devote to certain tasks. The design sequence of courses, coupled with other class work, will force time compromises to avoid being overwhelmed, or at least to minimize it.

The second course in the design sequence should involve group work almost entirely in order to stress that engineers in the real world rarely deal with problems in an isolated way. Most decisions made during design studies affect more than one discipline and, therefore, require both engineering tradeoffs and managerial decisions. In that regard, students should be required early on to learn basic parametric data analysis and presentation techniques. Student groups should be required to justify technical data and managerial decisions in periodic briefings to faculty and industry individuals citing how they arrived at decision points. This helps them develop awareness of how their decisions affected their evolving designs. Students should also be required to submit a final report in an accepted industry format such as that presented in Battelle's Report Writer's *Guide to Writing and Style*, available from Poor Richard Press in San Luis Obispo, California as a reprint and listed in the references.

To the maximum extent possible, outside sources of mission requirements should be used during the design sequence. This extra effort takes problem assignments out of the realm of academia and into the real world. Students are presented with interesting and challenging assignments which they might otherwise not get in a strictly classroom setting.

CHAPTER 1
EARLY CONFIGURATION DESIGN

This section will present cookbook methods and procedures which will get students started in the conceptual design process and show how to juggle all the knowledge they need to create viable and integrated designs. The best way to learn design is to do it. And do it. And do it. And do it until you know it. The design process starts with conceptual definition of whatever it is a customer wants to buy or marketing or the chief engineer defines. Usually, someone else dreams up the mission and designers are handed a set of requirements which frequently conflict. A new designer isn't likely to have amassed enough experience to have a firm idea of how to satisfy all the requirements, so here are some first steps to follow:

0. Fly an airplane.
1. Do a background investigation.
2. Find out who the end-users are and, if possible, talk to them.
3. Quantify stated and unstated mission requirements.
4. Use a simple sizing method to get started.
5. Define the design domain.
6. Do a conceptual layout of one or more configurations.

FLY AN AIRPLANE

It isn't necessary to learn to fly, although knowing how to fly airplanes is a big asset to a designer. Get some actual air time in any kind of airplane, the smaller and simpler the better. Share the cost of a two-seater or a four-seater with a fellow aero student or find a flying faculty member or friend and get airborne. Don't sit in the back, sit in a seat that has flight controls in front of it and, if possible, actually try flying the airplane. Experience a phugoid oscillation or Dutch roll instability firsthand. Ask the pilot to do a short field takeoff and landing. A visit to a local sailplane airport on a Sunday afternoon will provide opportunities to help launch and recover sailplanes and maybe even get a ride in one to experience that airplanes don't need engines to fly.

DO ABACKGROUND INVESTIGATION

Read the mission requirements. If you're not familiar with the context in which the airplane will be used, read up on it. Don't just read overviews of the particular requirements, find past examples and read about their development. Find books written by end users of several generations. Example, what do the Fokker Triplane, the Mitsubishi Zero-Sen, the North American Aviation Sabre Jet, the Folland Gnat and the General Dynamics Fighting Falcon have in common? They're all superbly maneuverable air superiority fighters representing novel innovation and courageous applications of state-of-the-art technologies.

Reading background materials will be more suited to a work environment and to leisure time than to seniors in aero engineering, but still find some time to understand the context of the requirement you'll be designing to. As a junior engineer in the conceptual design group at North American Aviation's Columbus Aircraft Division I built plastic models of production aircraft in the performance categories in which I was tasked to design something. That gave me an opportunity not only to learn about real airplanes, but to study their design by building models and detailing them.

It is useful when exploring new aeronautical concepts to examine where in the total domain of powered flight they lie. Figure 1 shows the relationship of most aircraft classes with other classes. Within each of these smaller domains lie parametrically feasible regions for specific sets of mission requirements. The designer's task is to define and explore these workable regions, as shown in Figure 2.

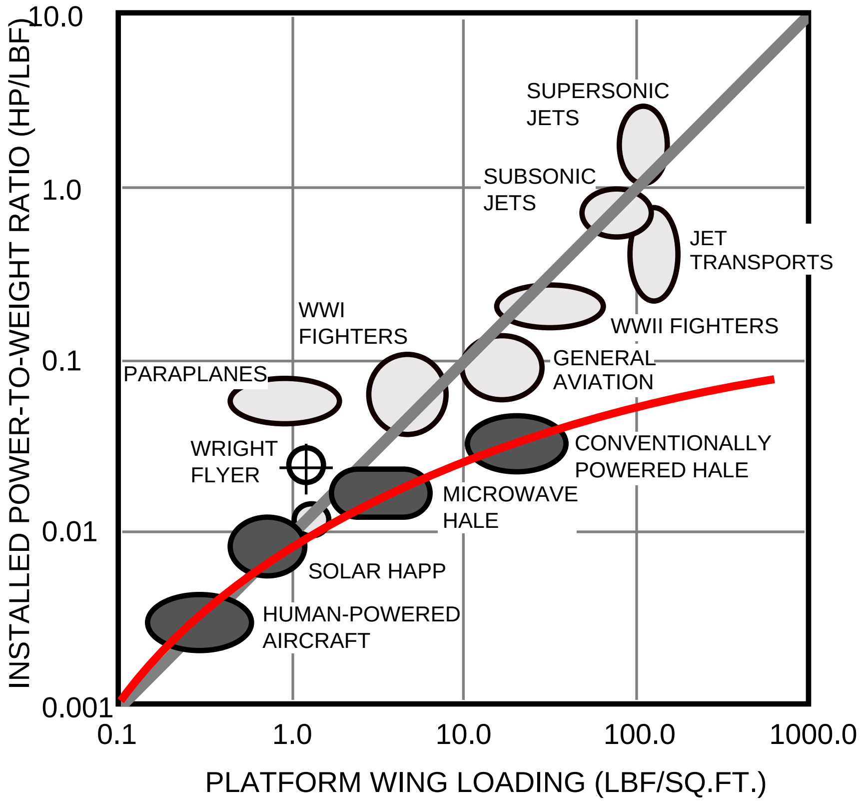


Figure 1. Trends are Apparent with Both Conventional and HALE Aircraft Classes.

The conceptual design process will start by estimating key design parameters and expressing mission requirements, and the pertinent laws of physics, as combinations of cruise thrust-to-weight ratio (T/W) and wing loading (W/S). A feasible region should lie inside these mission-specific constraint lines which may take the form of minimum range requirements, maximum or minimum endurance requirements, maximum attainable lift coefficient, or maximum attainable altitude, maximum permissible cruise speed for science experiments, and Reynolds Number effects on flying surfaces and/or propulsion performance. There will be points within each feasible region which look promising for fulfilling all mission requirements. These points may be plotted, as shown in Figure 3, then cross-plotted to obtain a minimum or maximum of some design variable.

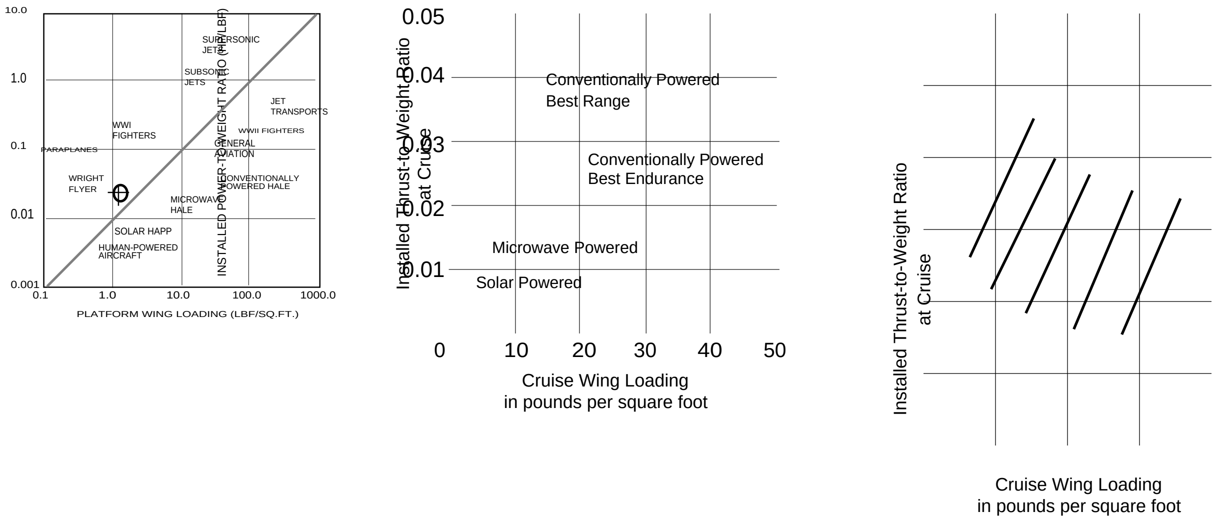


Figure 2. Parametric Carpet Plots Provide Visual Confirmation of the Effects of Mission Requirements on Basic Design Parameters.

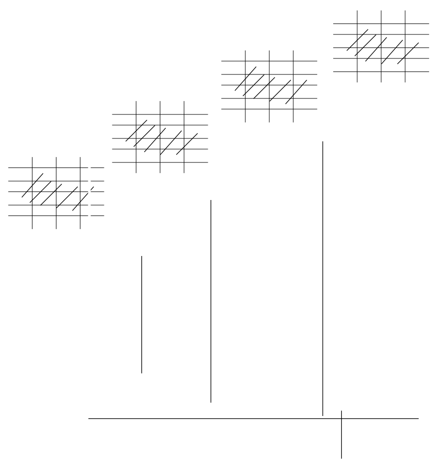


Figure 3. Data from the Plots Can Be Cross-Plotted to optimize Mission Parameters.

FIND OUT WHO THE END-USERS ARE AND, IF POSSIBLE, TALK TO THEM

If possible, talk to users to find out how airplanes in the performance category under consideration fulfill their mission requirements. Find out not only how missions are flown but what users want in a new platform that they don't have now. And, equally importantly, find out what features they have now they don't need or don't want that may adversely affect reliability, cost and maintainability.

QUANTIFY STATED AND UNSTATED MISSION REQUIREMENTS

It's important to figure out how design requirements affect the size, weight, cost and operation of an airplane. Look at the design requirements carefully and try to translate them to ranges of performance numbers which must be satisfied to make your design viable. Also look at legislative requirements in enough detail to determine how they affect your design. Most importantly, determine which requirements “drive” the design, and which are the most important requirements to meet. If any of the requirements conflict, how can they be traded off?

Airplanes are composed of components and subsystems and all exist to perform some type of mission, whether it’s flying for the joy of flight or carrying cargo from point A to point B. All these missions can be quantified in terms of the following design parameters:

Table 1. Mission Design Parameters are Often Interrelated.

Mission performance required	What parameter will this affect?
Mission Class	TakeOff Gross Weight (TOGW)
Payload type, size, weight	Fuselage shape, pod shape & location, maybe even wing size & placement
Range or Radius of Action	Aspect Ratio (AR)
Reserves	TOGW
Loiter or Endurance	AR, Glide Ratio (L/D), Wing Loading (W/S)
Takeoff and landing requirements	W/S, Thrust-to-Weight Ratio (T/W)
Dash requirements	Wing Sweep (Λ), T/W, shape
Speed Range	Sweep variability, vectored thrust
Survivability	Shape, TOGW, cost
Construction cost	TOGW, AR, Λ, Taper Ratio (λ)
Altitude envelope	T/W, W/S, AR, TOGW, pressurization
Maneuverability	T/W, W/S, Specific Excess Power (SEP), maneuvering load factor (n_{man}), stability margins
Carrier suitability	Everything

A good way to determine these factors is to look at other aircraft in the category under consideration, then, if possible, talk to pilots and other users involved with that category of aircraft. If you’re designing a private or commercial passenger or cargo carrier, talk with both the airlines and the local FAA office, if possible. Federal Aviation Regulations (FARs) frequently spell out clearly what minimum performance and safety standards civilian designs must meet and Military Specifications (MilSpecs) do the same for military.

Play with the applicable equations of motion without plugging in numbers to get a feel for the interdependencies of design parameters, then quantify as much of the mission requirements as possible, particularly payload size, shape, weight, loading or unloading requirements, viewing requirements, environmental requirements (such as pressurization, heating or cooling). Also quantify range, endurance and altitude requirements.

CHAPTER 2
SIZING METHODS

Where you start a conceptual design isn't really important; but getting started is. Use an existing method for making first estimates of weight and physical size, or perhaps, several different methods and compare their results. This chapter presents starting methods from industry, from text books in the reference library, and from other designers' notes. These specific methods lend themselves to modification to fit new classes of aircraft and personal working style. The challenge here is to reduce often non-specific requirements to quantifiable forms. The weight fraction method and some associated rules of thumb will help do that.

AZERO-ORDER AIRCRAFT SIZING METHOD

At this point, going through design process steps have provided an idea of what's been done in the past in the mission area of interest. Other early analytical steps have provided an understanding of mission requirements and, perhaps, have allowed creation of a candidate mission profile. Given that and payload requirements (or estimates) it is now possible to do a first estimate of takeoff gross weight (TOGW) using a weight fraction method similar to that found in several current design texts. The example to be used here is a subsonic bizjet with the following payload requirements. The mission profile looks like that shown in Figure 4.

Table 2. ABusiness Aircraft Will Have a Quantifiable Payload.

Item	Number Required	Unit Weight	Total Weight
Crew	2	170.#	340.#
Passengers	6	170	1,020
Baggage	8	30	240
Total			1,600.#

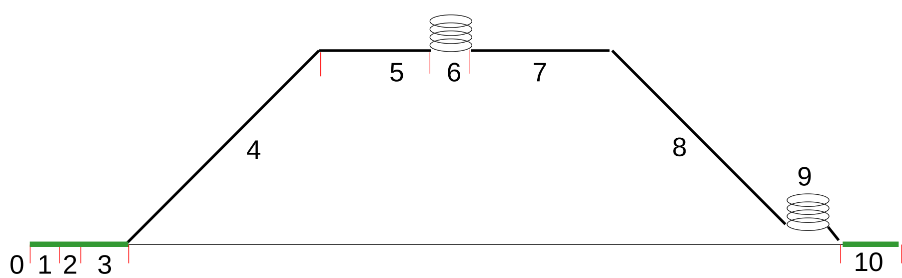


Figure 4. The Mission Profile for a Business Aircraft Can be Very Simple.

Using textbook methods, the various segments above may be quantified in terms of weight change from the beginning to the end of each segment. This zero-order weight estimating approach will begin with considering the TOGW of the aircraft to be the sum of three terms.

$$TOGW = W_{empty} + W_{fixed\ weights} + W_{fuel} \tag{1}$$

where $W_{fixed\ weights}$ is the sum of non-expendable items such as crew and expendable items such as installed weapons, cargo, passengers, baggage, etc. for the bizjet class aircraft under consideration here, Figure 5 presents an empirical relationship of empty weight to TOGW. For other aircraft types, see ref Nicolai chapter 5. Equation 1, then, can be used to find fuel weight as a residual. What this method yields is weight fractions for these three terms, only two of which are known and, therefore, TOGW is still unknown.

$$\frac{W_{\text{fuel}}}{TOGW} = 1 - \frac{W_{\text{empty}}}{TOGW} - \frac{W_{\text{fixed weights}}}{TOGW} \tag{2}$$

Next, the mission profile can be used as a starting point for a more detailed estimate of the fuel fraction needed. Refer to Figure 4 and define the following mission segments:

0 Mission Start at Ramp Weight which is greater than TOGW
1 Engine idle, and runup plus delays prior to takeoff
2 Taxi to end of runway
3 Takeoff and climb to specified height above runway elevation
4 Climb to cruise altitude and accelerate to cruise speed
5 Cruise out (referred to as the Transit To Segment elsewhere in this text)
6 Loiter
7 Return to base (referred to as the Transit From Segment elsewhere in this text)
8 Descend
9 Low altitude reserve or hold prior to landing
10 Land and shut down engines.

Given the mission segments as defined in Figure 4 and the general equations developed in the next section, it is now possible, with the continued assistance of empirically based weight fraction estimating methods, to quantify fuel fractions for each segment.

$$1 - \frac{W_{\text{fuel}}}{TOGW} = 1 - \frac{W_1}{W_0}\frac{W_2}{W_1}\frac{W_3}{W_2}\frac{W_4}{W_3}\frac{W_5}{W_4}\frac{W_6}{W_5}\frac{W_7}{W_6}\frac{W_8}{W_7}\frac{W_9}{W_8}\frac{W_{10}}{W_9} \tag{3}$$

Aircraft usually burn between 2.5% and 3% of total fuel from the time they start up, check engines, taxi to the runway and take off, or in terms of weight fractions as presented in Equation 3,

$$0.97 \le \frac{W_1}{W_0}\frac{W_2}{W_1}\frac{W_3}{W_2} \le 0.975 \tag{4}$$

For the climb segment (segment 4), use Figure 5 to estimate the fuel fraction used for climb and acceleration to cruise speed.

A version of the Breguet range equation can be used to estimate fuel fraction required for the Transit To Segment. This equation was presented as Equation 34 and its form to be used here is

$$R = \frac{V_{\text{cruise}}}{tsfc}\frac{L}{D}\ln\frac{W_4}{W_5} \tag{5}$$

for jet aircraft and

$$R = \frac{\eta_{\text{propeller}}}{bsfc}\frac{L}{D}\ln\frac{W_4}{W_5} \tag{6}$$

for propeller-driven aircraft. Terms can be rearranged to provide the required fuel fractions. Glide ratio can be estimated as approximately 95% of maximum glide ratio, or

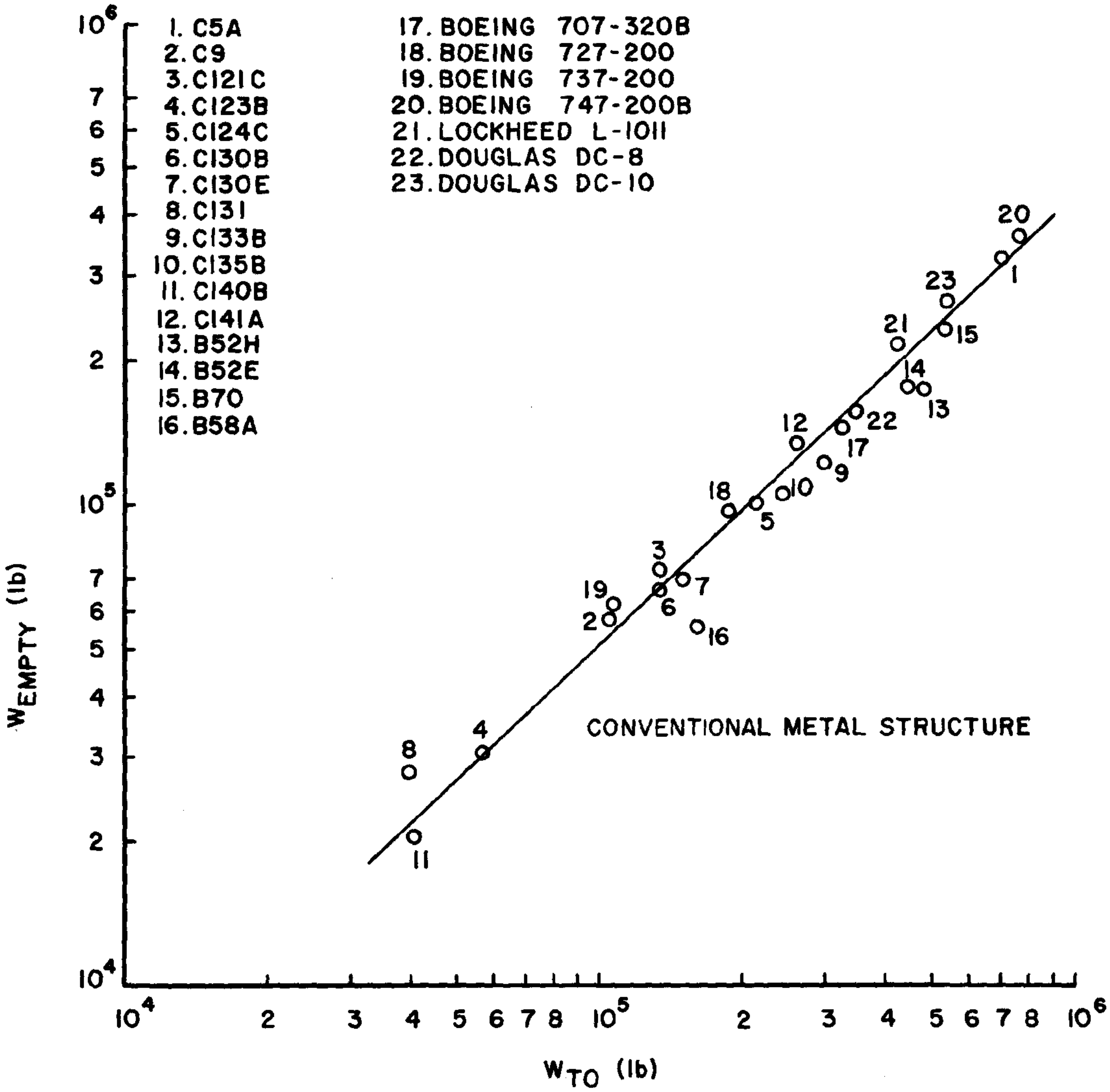


Figure 5. The Empty Weight-to-Gross Weight Fraction is Predictable for Transport Aircraft.

$$\left(\frac{L}{D}\right)_{max} = \frac{1}{2\sqrt{\dfrac{C_{D_0}}{\pi e AR}}} \tag{7}$$

$$\left(\frac{L}{D}\right)_{cruise} = 0.95\left(\frac{L}{D}\right)_{max} = \frac{0.95}{2\sqrt{\dfrac{C_{D_0}}{\pi e AR}}} \tag{8}$$

Cruise speed is frequently given in the mission requirement, propeller efficiency factor may be estimated as 0.85 here and airplane efficiency factor as 0.8. The value for parasite drag may be estimated from Figure 7 using cruise speed.

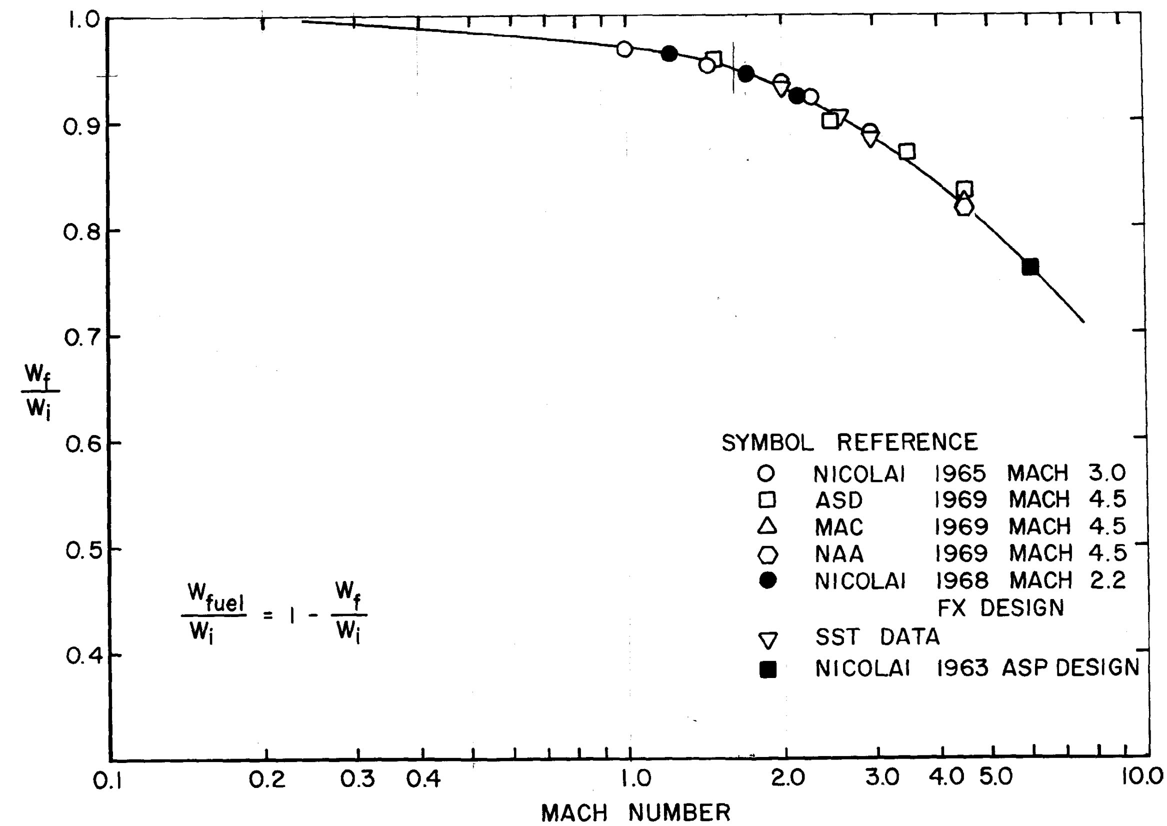


Figure 6. The Fuel Fraction for Climb is Strongly Affected by Cruise Mach Number.

The loiter segment uses a form of the Breguet endurance equation given earlier as Equation 96. For propeller driven aircraft, this is

$$E = \frac{\eta_{\text{propeller}}}{bsfc} \frac{L}{D} \frac{1}{V_{\text{cruise}}} \ln \frac{W_5}{W_6} \tag{9}$$

The Transit From Segment also uses the Breguet range equations in the form:

$$R = \frac{V_{\text{cruise}}}{tsfc} \frac{L}{D} \ln \frac{W_6}{W_7} \tag{10}$$

for jet aircraft and

$$R = \frac{\eta_{\text{propeller}}}{bsfc} \frac{L}{D} \ln \frac{W_6}{W_7} \tag{11}$$

for propeller driven aircraft.

The Descent Segment will probably burn no more than half the fuel used in the Climb Segment, so estimate its fuel fraction as

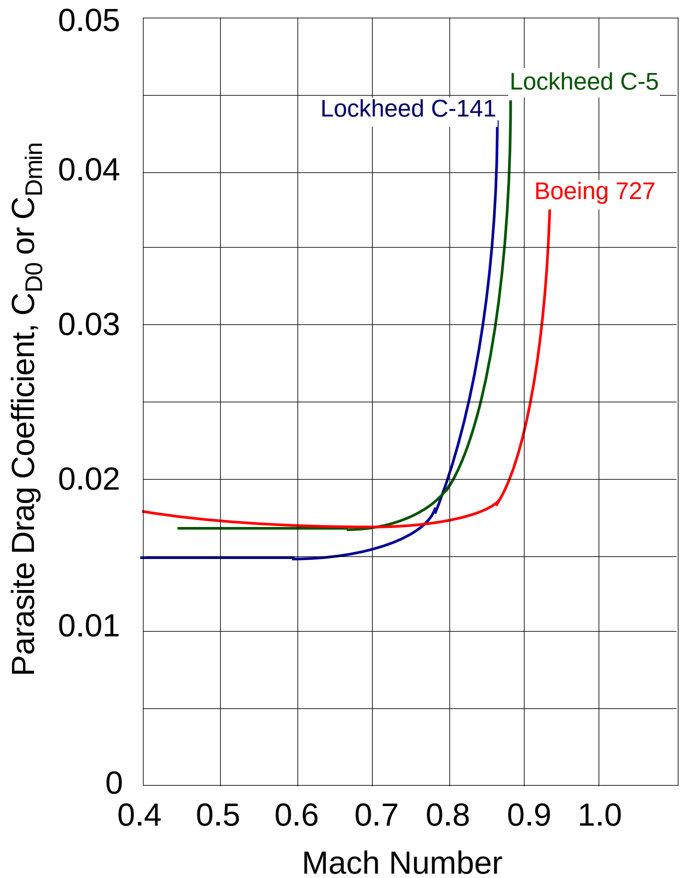


Figure 7. Parasite Drag for a Transport or bomber can be Estimated as a Function of Mach Number.

$$0.985 \le \frac{W_7}{W_8} \le 0.99 \tag{12}$$

and the low altitude Reserve Segment can be estimated using the Breguet endurance equation in the form

$$E = \frac{\eta_{propeller}}{bsfc}\frac{L}{D}\frac{1}{V_{cruise}}\ln\frac{W_8}{W_9} \tag{13}$$

Finally, the Landing Segment can be estimated as about the same fuel fraction as the descent segment, or

$$0.985 \le \frac{W_9}{W_{10}} \le 0.99 \tag{14}$$

The total fuel fraction required for the mission, then, will be the product of all these fractions as shown in Equation 3 which may be rewritten as

$$\frac{W_{fuel}}{TOGW} = \frac{W_1}{W_0}\frac{W_2}{W_1}\frac{W_3}{W_2}\frac{W_4}{W_3}\frac{W_5}{W_4}\frac{W_6}{W_5}\frac{W_7}{W_6}\frac{W_8}{W_7}\frac{W_9}{W_8}\frac{W_{10}}{W_9}$$

Use Figure 8 to find an estimate of TOGW given the revised value of empty weight fraction calculated in the above steps.

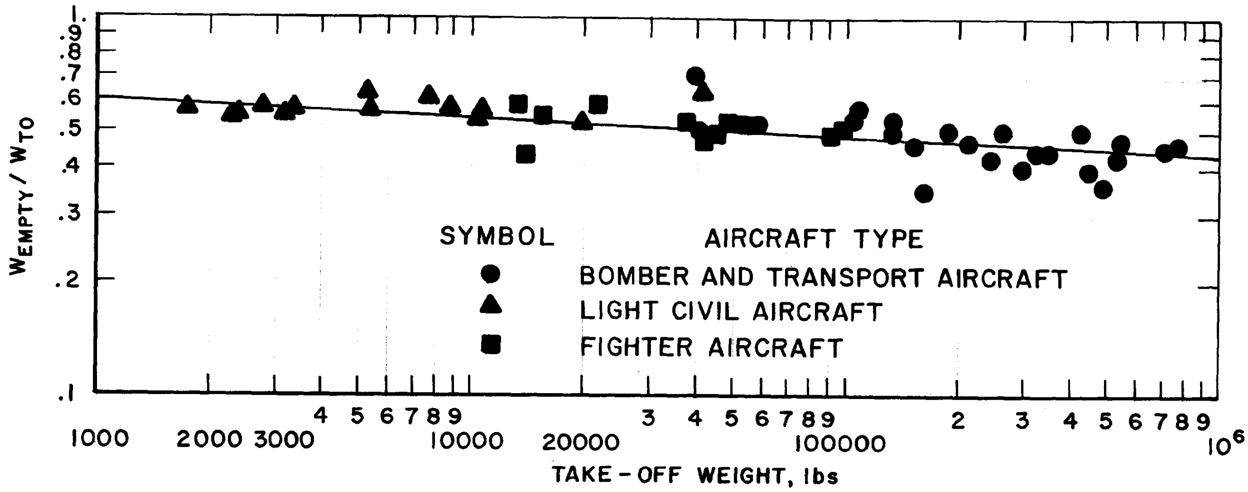


Figure 8. TakeOff Gross Weight can be Estimated Using Previously Found Mission Segment Weight Fractions.

Use this value of TOGW to recalculate Equation 2 weight fractions and iterate through the method again until the differences between the current value of TOGW and the previous one are less than an acceptable percentage error. Fuel fractions for the bizjet example are given in Table 3. Note the absence of a return cruise segment.

Table 3. The Textbook Mission Profile Analysis Method Yields a First Estimate of Fuel Weight.

Segment Number	Segment Name	F_F	$1 - F_F$	Weight Change	Weight at End
0	Preflight & load	1.000	0.000	0.#	14,000.#
1	Start & warmup	0.990	0.010	140	13,860
2	Taxi	0.995	0.005	69	13,791
3	Takeoff	0.995	0.005	69	13,722
4	Climb	0.980	0.020	274	13,447
5	Cruise Out	0.761	0.239	3,214	10,233
6	Loiter	0.921	0.079	808	9,425
7	Cruise Back	1.000	0.000	0	9,425
7	Descent	0.990	0.010	94	9,331
8	Landing, taxi & shutdown	0.992	0.008	75	9,256
9	Trapped Fuel & Oil	0.995	0.005	46	9,210.#
Total Fuel Onboard				4,790.#	

An initial weight breakdown can now be created as shown in Table 4.

Table 4. This First Estimate of Detail Weights Can be Compared to the Author's Figures to Determine Allowable Empty Weight and the Need to Iterate.

Item	Weight
Empty	7,610.#
Crew	340
Passengers	1,020
Baggage	240
Fuel	4,790
TakeOff Gross Weight	14,000.#

The next step will be describing the mission (or missions) to be flown in general algebraic terms such that each mission is expressed in terms of general design parameters which can be calculated as other design parameters are systematically varied.

LOITER PERFORMANCE ESTIMATING METHOD

This chapter will discuss how to describe each segment of the mission profile using equations which incorporate and relate basic design parameters. First, though, let's look at what a basic design parameter is. The mission requirements stated for any airplane will include desired (or mandatory) values of specific performance items to be met which are subsets of the requirements discussed above.

Takeoff and Landing Segment requirements will be stated in terms of field length, maximum and/or minimum allowable approach speed, field elevations and temperatures, runway acoustic and weight footprints, and terminal interface requirements for onloading and/or offloading cargo and/or people. The design parameters affected by these requirements are thrust-to-weight ratio, wing loading, wingspan, maximum lift coefficient and physical relationships of the detailed configuration components.

Climb Segment requirements will be stated in terms of attaining certain altitudes in certain times within certain distances. Design parameters affected by climb segment requirements are thrust-to-weight ratio, best angle and/or rate of climb speed, and times to specific altitudes.

Transit Segment requirements will be stated in terms of attaining range or radius of action within certain times. Design parameters affected will be wing loading, thrust-to-weight ratio, aspect ratio, propulsive efficiency, specific fuel consumption, L/D. These legs can normally be evaluated using the Breguet range and endurance equations since they're flown under almost steady-state conditions. In the case of military aircraft, the "transit-to" leg may have a dash requirement where the design parameters affected will be maximum level speed and level acceleration.

Military aircraft may also have a combat segment where mission requirements would be stated in terms of specific excess power which is defined as

$$SEP = \frac{(T - D)V}{nW} \tag{15}$$

Also important are maximum instantaneous and sustained load factors, turn radius, minimum altitude loss under maneuvering loads, times from one Mach number to another under load, and combat altitude. In the case of bombers or attack aircraft, these may be augmented by required bombing accuracies, maximum or minimum release altitude, target approach speed, turn radius and geometric field-of-view requirements. As you might guess, virtually all design parameters will be affected by these requirements, but good starting points will be wing loading, thrust-to-weight ratio, maximum lift coefficient under load, ultimate and maneuvering load factors.

Loiter and Reserve Segment requirements will be stated in terms of time-on-station or a mandatory low altitude hold. Design parameters affected will be wing loading, aspect ratio and propulsive efficiency. The Breguet endurance equation will adequately calculate required times.

Descent Segment requirements will be stated similarly to climb requirements in terms of mandatory maximum or minimum times from one altitude to another, total descent time and/or distance. Design parameters affected will be drag, load factor, descent speed, wing loading.

The next step is to write general equations for each segment, or each part of each segment, in terms of design parameters so that loiter segment weight change, and therefore time, can be estimated and compared to RFP requirements.

$$\Delta W_{\text{loiter}} = W_{\text{fuel}} - \Delta W_{\text{takeoff}} - \Delta W_{\text{climb}} - \Delta W_{\text{transit to}} - \Delta W_{\text{landing}} - \Delta W_{\text{reserve}} - \Delta W_{\text{descent}} - \Delta W_{\text{transit from}} \tag{16}$$

This method can form the basis for parametric analysis if loiter time is allowed to be negative. This will be addressed in a later section.

Takeoff

Not much is known about the aircraft at this stage so using a detailed takeoff estimation method would introduce unnecessary complexity to the design process. Equation 17 is a published estimating method for takeoff ground roll and terms can be arranged to provide the general configuration descriptors wing loading (W/S) and thrust-to-weight ratio (T/W).

$$\left(s_{ground}\right)_{takeoff} = \frac{13.08\dfrac{W}{S_{ref}}}{\dfrac{\left(C_{L_{\max}}\right)_{\text{power off}}}{\left(\dfrac{V_{\text{takeoff}}}{V_{\text{stall}}}\right)^2}\left(\dfrac{T}{W} - 0.1\right)} \tag{17}$$

This is a zero-order estimate which is accurate enough for initial calculations. It's instructive here to look at more detailed kinematics of the takeoff segment to gain an appreciation for what Equation 17 doesn't include. The following method can be used as a first order method when more is known about aircraft aerodynamics, propulsion and weights. Notice also the thought process involved in analyzing the takeoff segment. Consider the basic forces acting on an aircraft under linear, steady-state takeoff conditions, which are shown in Figure 9.

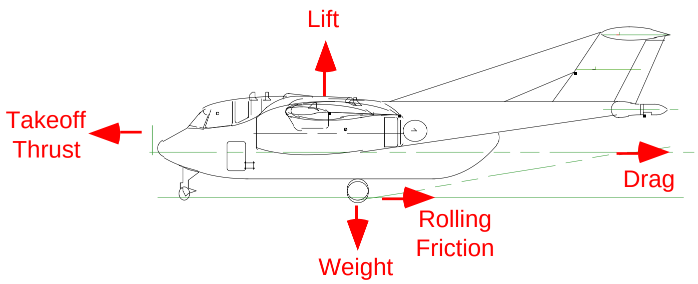


Figure 9. Forces Acting on an Airplane in Linear Motion.

Summing forces in both horizontal and vertical directions yields the equations of motion for takeoff. The horizontal summation should be set equal to a momentum change term.

$$\Sigma F_{\text{horizontal}} = T - D - F = \frac{W}{g}a \qquad \Sigma F_{\text{vertical}} = L - W = N \tag{18}$$

where $F = \mu N = \mu(W - L)$ {since $W > L$ during the takeoff roll}

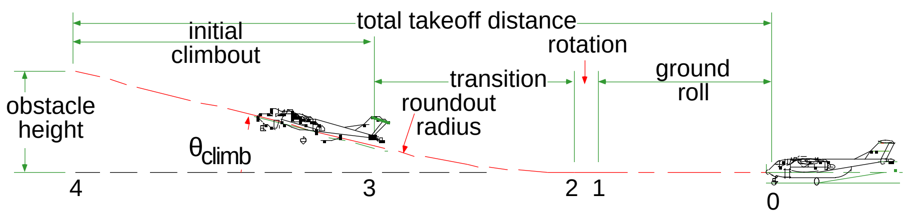
Continuing,

$$T - D - \mu(W - L) = \frac{W}{g}a \tag{19}$$

and

$$a = g\left[\frac{T}{W} - \frac{D}{W} - \mu\left(1 - \frac{L}{W}\right)\right] \tag{20}$$

The takeoff segment will be composed of ground roll, rotation, transition-to-liftoff and initial climbout phases as shown in Figure 10. This figure shows the definition of variables and subscripts which will be used in the following equations.



V =	V_{climb}	$V_{transition}$	$V_{rotation}$	$V_{rotation}$	0
h =	$h_{obstacle}$	0	0	0	0
t =	$\Delta t_{initial\ climb}$	$\Delta t_{transition}$	$\Delta t_{rotation}$	$\Delta t_{rotation}$	0
s =	s_{air}	$s_{transition}$	$s_{rotation}$	s_{ground}	0

Figure 10. Values of Subscripted Variables During the Takeoff Segment.

Values for speeds at each point in the takeoff roll are as follows:

$$V_{\text{stall}} = \sqrt{\frac{2}{\rho_0 C_{L_{\max}}}}\sqrt{\frac{W}{S_{\text{reference}}}} \tag{21}$$

$$V_{\text{rotation}} = 0.84 V_{\text{stall}} \tag{22}$$

$$V_{\text{liftoff}} = 1.05 V_{\text{stall}} \tag{23}$$

$$V_{\text{climb}} = 1.3 V_{\text{stall}} \tag{24}$$

Altitude at start of the sustained climb segment is 50 feet above runway elevation (mean sea level and standard day are the default values). Engines are started on the ramp and takeoff will be straight into the wind after a short taxi. For that reason, no allowance will be made here for fuel burnoff prior to the start of the takeoff roll. For transports and many other aircraft categories, this is not a valid set of assumptions. One simplified takeoff ground roll equation which has wing loading and thrust loading terms comes from ref 10, page 10-8.

$$s_{\text{ground}} = \frac{1.44\left(\dfrac{W}{S_{\text{reference}}}\right)_{\text{takeoff}}}{g\rho_0 C_{L_{\max}}\left[\dfrac{T}{W} - \dfrac{D}{W} - \mu\left(1 - \dfrac{L}{W}\right)\right]} \tag{25}$$

$$s_{\text{ground}} = \frac{0.72V_{\text{stall}}^2}{g\left[\dfrac{T}{W} - \dfrac{D}{W} - \mu\left(1 - \dfrac{L}{W}\right)\right]} \tag{26}$$

where $L = 0.5\rho_0\left(0.84V_{\text{stall}}\right)^2 S_{\text{reference}} C_{L_{\text{rotation}}}$

Speed will be an average value between 0 and $0.84V_{\text{stall}}$. If acceleration is roughly linear during the takeoff roll, then 0.5 V_{stall} should be a conservative estimate of average speed. Time for ground roll will be

$$\Delta t_{\text{ground}} = \frac{s_{\text{ground}}}{0.5V_{\text{stall}}} \tag{27}$$

Fuel burned will be

$$\Delta W_{\text{ground}} = sfc_{\text{takeoff}} T_{\text{takeoff}} \Delta t_{\text{ground}} \tag{28}$$

Rotation typically takes around three seconds for conventional aircraft (ref 17). Distance and fuel burned during rotation will be

$$\Delta s_{\text{rotation}} = \Delta t_{\text{rotation}} V_{\text{rotation}} \quad \text{V in fps or mps} \tag{29}$$

$$\Delta W_{\text{rotation}} = sfc_{\text{takeoff}} T_{\text{takeoff}} \Delta t_{\text{rotation}} \tag{30}$$

The transition phase starts at the end of rotation and ends at liftoff. Speed increases from $0.84V_{\text{stall}}$ to $1.05V_{\text{stall}}$ more or less linearly. An average value would be $0.95V_{\text{stall}}$. Distance, time and fuel burned will be:

$$\Delta s_{\text{transition}} = \frac{V_{\text{transition}}^2}{0.15g}\sin\theta_{\text{climb}} \tag{31}$$

$$\Delta s_{\text{transition}} = \frac{\left(0.95V_{\text{stall}}\right)^2}{0.15g}\sin\theta_{\text{climb}} \tag{32}$$

where $\sin\theta_{\text{climb}}$ may be defined as

$$\sin\theta_{\text{climb}} = \frac{T - D}{W} \quad \text{evaluated at } 0.95V_{\text{stall}} \tag{33}$$

taking a small angle assumption into account for this brief climb phase. Later, $\sin\theta_{\text{climb}}$ will be defined more rigorously.

$$\Delta t_{\text{transition}} = \frac{s_{\text{transition}}}{0.95V_{\text{stall}}} \tag{34}$$

$$\Delta W_{\text{transition}} = sfc_{\text{takeoff}} T_{\text{takeoff}} \Delta t_{\text{transition}} \tag{35}$$

The initial climbout phase of the takeoff segment will be from runway elevation (mean sea level is the default) to 50 feet above runway elevation and with speed increasing from $1.05V_{\text{stall}}$ to $1.3V_{\text{stall}}$. An average value of speed during this phase might be $1.15V_{\text{stall}}$. An equation for the ground distance covered during initial climb comes from ref 18 and is

$$s_{\text{air}} = \frac{50 - h_{\text{transition}}}{\tan\theta_{\text{climb}}} = \frac{50 - 0}{\frac{T - D}{L}} = \frac{50L}{T - D} \tag{36}$$

where drag and lift are evaluated at $1.15V_{\text{stall}}$ and thrust is that generated at takeoff setting. Note, too, that gear and flaps remain at takeoff setting. This is the phase of the takeoff segment where the pilot cleans up the aircraft for climb so we'll assume the takeoff drag coefficient value applies. Continuing, initial climb time and fuel burned are

$$\Delta t_{\text{air}} = \frac{s_{\text{air}}}{1.15V_{\text{stall}}} \tag{37}$$

$$\Delta W_{\text{air}} = sfc_{\text{takeoff}} T_{\text{takeoff}} \Delta t_{\text{air}} \tag{38}$$

Total takeoff distance, time required and fuel burned for the takeoff segment will be the sums of our calculations so far.

$$s_{\text{takeoff}} = s_{\text{ground}} + s_{\text{rotation}} + s_{\text{transition}} + s_{\text{air}} \tag{39}$$

$$t_{\text{takeoff}} = \Delta t_{\text{ground}} + \Delta t_{\text{rotation}} + \Delta t_{\text{transition}} + \Delta t_{\text{air}} \tag{40}$$

$$\Delta W_{\text{takeoff}} = \Delta W_{\text{ground}} + \Delta W_{\text{rotation}} + \Delta W_{\text{transition}} + \Delta W_{\text{air}} \tag{41}$$

$$\Delta W_{\text{takeoff}} = sfc_{\text{takeoff}} T_{\text{takeoff}} t_{\text{takeoff}} \tag{42}$$

Use Appendix A for zero-order estimates of the remaining mission segments. For first order estimates, and to gain an appreciation for what zero-order methods simplify, continue here.

Climb to Altitude

For most aircraft, sustained climb to altitude will probably be at best rate of climb speed (most height gained per unit of time) in order to get to altitude as soon as possible to minimize fuel consumption. For some aircraft types, however, climb may occur at speeds significantly less than maximum rate-of-climb speed because of limiting structural or aerodynamic effects. In either case, initial climb weight will be Takeoff Gross Weight (TOGW) less fuel burned during the takeoff segment.

Later in the design process, enough engine data will be available to construct a matrix of power or thrust available as they vary with airspeed at each altitude. This will be the first order estimate. Lacking this detailed information, a zero-order method can be used to create an approximation of thrust or power lapse rate with altitude and airspeed. It's particularly important to match fuel consumption with thrust levels for later calculations.

The takeoff segment provided initial climb weight, climb angle and climb speed estimates. Assume that, during the first thousand feet of sustained climb, the aircraft will accelerate to its best climb speed and stabilize its climb angle. Climb angle may be defined as (ref 8, page 295)

$$\sin\theta_{\text{climb}} = \frac{C_{L_{\text{climb}}}^2 T}{W\left(C_{L_{\text{climb}}}^2 + C_{D_{\text{T}}}^2\right)} - \sqrt{\left[\frac{C_{L_{\text{climb}}}^2 T}{W\left(C_{L_{\text{climb}}}^2 + C_{D_{\text{T}}}^2\right)}\right]^2 - \frac{C_{L_{\text{climb}}}^2 T^2 - C_{D_{\text{T}}}^2 W^2}{W^2\left(C_{L_{\text{climb}}}^2 + C_{D_{\text{T}}}^2\right)}} \tag{43}$$

and the initial value of rate of climb will be

$$R/C_{\text{start}} = \sin\theta_{\text{climb}}\sqrt{\frac{2W\cos\theta_{\text{climb}}}{\rho_{\text{start}} S_{\text{reference}} C_{L_{\text{climb}}}}} \tag{44}$$

Climb may also occur at constant dynamic pressure, or constant q. For any constant value of indicated airspeed, true airspeed increases with altitude as σ decreases; or if q remains constant with increasing altitude, then

$$q = 0.5\rho_1 V_1^2 = 0.5\rho_2 V_2^2 \tag{45}$$

$$V_2 = V_1\sqrt{\frac{\rho_1}{\rho_2}} \tag{46}$$

If ρ_1 is the start of climb at approximately mean sea level, then ρ_1 is close to ρ_0 and the equation becomes

$$\left(V_{\text{climb}}\right)_{\text{end}} = \left(V_{\text{climb}}\right)_{\text{start}}\sqrt{\sigma} = 1.3V_{\text{stall}}\sqrt{\sigma} \tag{47}$$

Rate-of-climb calculations may then be simplified greatly. With many aircraft types, however, climbing at best rate-of-climb speed will save fuel and lighten total aircraft weight.

Conditions after the first 1,000 feet of climb must now be estimated by searching for the maximum climb rate at that altitude assuming a negligible fuel burn. Set climb speed equal to stall speed at altitude and calculate thrust, drag, lift, climb rate and climb angle. Then increment climb speed a small amount and recalculate all the above. Continue until maximum climb rate occurs and use the differences in climb rates between altitudes to calculate climb time, fuel burn and climb distance as follows:

$$\Delta t_{\text{climb}} = \frac{\Delta h}{\dfrac{R/C_{\text{start}} + R/C_{\text{end}}}{2}} \tag{48}$$

$$\Delta W_{\text{climb}} = sfc_{\text{climb}} T_{\text{climb}} \Delta t_{\text{climb}} \tag{49}$$

$$s_{\text{climb}} = \frac{V_{\text{start}} + V_{\text{end}}}{2} \cos\theta_{\text{climb}} \Delta t_{\text{climb}} \tag{50}$$

Total time-to-climb, fuel burned and horizontal distance covered will then be

$$(t_{\text{climb}})_{\text{total}} = \Sigma \Delta t_{\text{climb}} \tag{51}$$

$$(\Delta W_{\text{climb}})_{\text{total}} = \Sigma \Delta W_{\text{climb}} \tag{52}$$

$$(s_{\text{climb}})_{\text{total}} = \Sigma \Delta s_{\text{climb}} \tag{53}$$

Transit to Station

The transit phase will probably be flown at best range speed unless there is a dash requirement. Therefore, a variant of the Breguet Range Equation may be used. Initial weight going out will be TOGW less fuel used for climb. From ref 19 (page 2:37):

$$R = \frac{V_{cruise}}{sfc_{cruise}} * \left(\frac{L}{D}\right)_{cruise} * \ln\left(\frac{W_{initial}}{W_{initial} - W_{final}}\right)_{cruise} \tag{54}$$

The lift-to-drag ratio to be used above may be estimated as

$$(L/D)_{\text{transit}} = \sqrt{\frac{\pi e_{\text{airplane}} AR}{4C_{D_0}}} \tag{55}$$

or it may be calculated from end-of-climb parameters. The transit distance (R in the equation above) will probably be specified as a mission requirement. If it is, then deduct climb distance from it (if the RFP permits) and solve the equation for final weight:

$$W_{\text{final}} = W_{initial}\left(\frac{1}{e^{\frac{V_{cruise}}{sfc_{cruise}} * \left(\frac{L}{D}\right)_{cruise}}} - 1\right) \tag{56}$$

and best range speed may be found using

$$V_{\text{best range}} = \sqrt{\frac{2}{\rho\sqrt{C_{D_0}\pi e_{\text{airplane}} AR}}} \sqrt{\frac{W}{S_{\text{reference}}}} \tag{57}$$

with wing loading varying over the transit distance. Since both initial and final weights for this segment are now known, average best range speed may be estimated and transit time and distance will be

$$\Delta t_{\text{transit}} = \frac{R}{\left(V_{\text{best range}}\right)_{\text{average}}} \tag{58}$$

$$\Delta W_{\text{transit}} = W_{initial} - W_{final} \tag{59}$$

To summarize, the equations above provide ways of estimating elapsed time, distance and fuel burned for flight segments up to the start of loiter. Now begin working backward from the landing segment to the end of the loiter segment.

Landing

Air Distance Over a Fifty Foot Obstacle

Assume a flight path along straight line at constant velocity such that

$$\sin\theta = \frac{C_{D_T}}{C_L} - \frac{T}{W} \tag{60}$$

$$s_{air} = \frac{50}{\sin\theta} \tag{61}$$

$$\mathrm{K} = \frac{\mathrm{V}}{\left(\mathrm{V}_{\text{stall}}\right)_{\text{power off}}} \tag{62}$$

$$\text{Rate of Descent} = 1.68889 V_{S_{P0}} K \sin\theta \tag{63}$$

Ground Roll Distance without Brakes

The velocity interval from V to V_{brake} with retarding forces drag and unbraked friction where $\left(C_{D_T}\right)_{taxi} > \mu C_L$ will yield the following relationships:

$$P_1 = \frac{\left[\sigma\left(C_{D_T}\right)_{taxi} - \mu\left(C_L\right)_{taxi}\right]}{295\mu \dfrac{W}{S_{ref}}} \tag{64}$$

$$\left(s_{ground}\right)_{\text{no brakes}} = \frac{0.0443}{\mu\sqrt{P_1}} \ln\left(\frac{1 + P_1 V^2}{1 + P_1 V_{braking}^2}\right) \tag{65}$$

$$\left(t_{ground}\right)_{\text{no brakes}} = \frac{0.0523}{\mu\sqrt{P_1}}\left[Tan^{-1}\left(V\sqrt{P_1}\right) - Tan^{-1}\left(V_{braking}\sqrt{P_1}\right)\right] \tag{66}$$

Ground Roll Distance with Brakes

This phase covers the velocity interval from V_{brake} to $V_{\text{reverse thrust}}$ with full brakes applied.

$$P_2 = \frac{-\sigma\left(C_{D_T} - \mu_{braking} C_{L_T}\right)}{295\mu_{braking}\dfrac{W}{S_{ref}}} \tag{67}$$

$$\left(s_{ground}\right)_{braking} = \frac{0.0443}{\mu_{braking}P_2}\ln\left(\frac{1-P_2V^2}{1-P_2V_{braking}^2}\right) \tag{68}$$

$$\left(t_{ground}\right)_{\text{braking}} = \frac{0.0262}{\mu_{braking}\sqrt{P_2}}\ln\left[\frac{1+\left(V_{braking}-V_{\text{reverse thrust}}\right)\sqrt{P_2}-P_2V_{braking}V_{\text{reverse thrust}}}{1-\left(V_{braking}-V_{\text{reverse thrust}}\right)\sqrt{P_2}-P_2V_{braking}V_{\text{reverse thrust}}}\right] \tag{69}$$

Ground Roll Distance with Brakes and Reverse Thrust

This phase covers the velocity interval from $V_{\text{reverse thrust}}$ to 0 with reverse thrust decreasing linearly with velocity.

$$P_3 = \mu_{braking} + \frac{T_{reverse}}{W} \tag{70}$$

$$P_4 = \frac{K_{\text{reverse thrust}}}{W} \tag{71}$$

$$P_5 = \sqrt{P_4^2 + 4\mu_{braking}P_2P_3} \tag{72}$$

$$\left(s_{ground}\right)_{\text{reverse thrust}} = \frac{0.0443}{\mu_{braking}P_2}\left\{\frac{P_4}{P_5}\ln\left[\frac{2P_3+\left(P_4+P_5\right)V_{\text{reverse thrust}}}{2P_3+\left(P_4-P_5\right)V_{\text{reverse thrust}}}\right]-\ln\left[1+\frac{P_4V_{\text{reverse thrust}}-\mu_{braking}P_2V_{\text{reverse thrust}}^2}{P_3}\right]\right\} \tag{73}$$

Other Equations

$$\gamma = 0.0523 = Tan^{-1}\left[\frac{V\sqrt{P_1}-V_{braking}\sqrt{P_1}}{1+P_1VV_{braking}}\right] \tag{74}$$

$$\frac{T\mu\sqrt{P_1}}{\gamma} = Tan^{-1}\left(V\sqrt{P_1}\right) - Tan^{-1}\left(V_{braking}\sqrt{P_1}\right) \tag{75}$$

$$\left(1+P_1VV_{braking}\right)\tan\gamma = V\sqrt{P_1} - V_{braking}\sqrt{P_1} \tag{76}$$

$$\left(P_1VV_{braking}\right)\tan\gamma + V_{braking}\sqrt{P_1} = V\sqrt{P_1} - \tan\gamma \tag{77}$$

$$V_{braking} = \frac{V\sqrt{P_1} - \tan\gamma}{P_1 V \tan\gamma + \sqrt{P_1}} \tag{78}$$

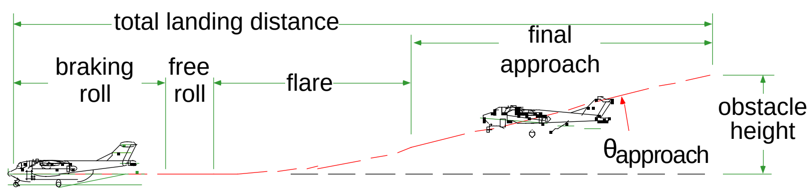
$$Tan^{-1}\left(V\sqrt{P_1}\right) - \gamma = Tan^{-1}\left(V_{braking}\sqrt{P_1}\right) \tag{79}$$

$$V_{braking}\sqrt{P_1} = \tan\left[Tan^{-1}\left(V\sqrt{P_1}\right) - \gamma\right] \tag{80}$$

$$V_{braking}\sqrt{P_1} = \frac{V\sqrt{P_1} - \tan\gamma}{1 + V\sqrt{P_1}\tan\gamma} \tag{81}$$

$$V_{braking} = \frac{V\sqrt{P_1} - \tan\gamma}{\sqrt{P_1} + V\sqrt{P_1}\tan\gamma} \tag{82}$$

Final mission weight corresponds to maximum landing weight. Approaches will be flown down to the runway with a circular arc flare, the initial point of tangency being at the approach flightpath angle and the final point of tangency being at gear contact with the runway. Spoilers are frequently used throughout approaches for glidepath control and to assure that the aircraft is firmly on the runway before the brakes are applied. The sequence used here for calculating landing performance will include final approach from 50 feet above ground elevation, flare, free roll, then a braking roll to a complete stop. Figure 11 shows this sequence of events.



	4	3	2	1	0
V =	0	V_{brake}	$V_{landing}$	V_{flare}	$V_{approach}$
h =	0	0	0	0	$h_{approach}$
t =	Δt_{brake}	$\Delta t_{free\ roll}$	Δt_{flare}	$\Delta t_{approach}$	0
s =	s_{brake}	$s_{free\ roll}$	s_{flare}	$s_{approach}$	0

Figure 11. Values of Variables and Subscripts During the Landing Segment.

Speeds to be used are as follows:

$$V_{approach} = 1.3V_{stall}$$

$$V_{flare} = 1.2V_{stall}$$

$$V_{touchdown} = 1.1V_{stall}$$

$$V_{brake} = 0.8V_{stall}$$

Figure 4 defines the force balance on landing and the landing segment may be expressed as

$$s_{\text{total}} = s_{\text{air}} + s_{\text{flare}} + s_{\text{free roll}} + s_{\text{braking}} \tag{83}$$

Since approaches will be flown down the runway from 50 feet, the air distance covered may be calculated as a constant descent. Practically speaking, the final descent rate, R/D, would probably be in the vicinity of 300 fpm.

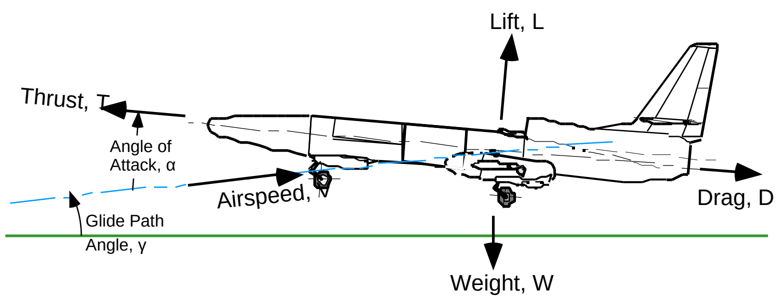


Figure 12. AFree-Body Diagram Shows Forces Acting on an Aircraft on Landing Approach.

$$\tan\gamma_{\text{descent}} = \frac{R/D}{V_{\text{approach}}} \tag{84}$$

$$s_{\text{air}} = \frac{50}{\tan\gamma_{\text{descent}}} \tag{85}$$

Landing flare distance, s_{flare}, will be zero by definition for most HALE applications, STOLaircraft, and carrier-suitable aircraft. The last two terms come from ref 8 (p. 329) and are:

$$s_{\text{free roll}} = \frac{W_{\text{landing}}\left(V^2_{\text{touchdown}} - V^2_{\text{braking}}\right)}{2g\text{R}_{\text{free roll}}} \tag{86}$$

$$s_{\text{free roll}} = \frac{W_{\text{landing}} V^2_{\text{braking}}}{2g\text{R}_{\text{free roll}}} \tag{87}$$

The resistance forces, $R_{\text{free roll}}$ and R_{braking}, may be found by summing horizontal forces:

$$\text{R}_{\text{free roll}} = \text{D}_{\text{free roll}} + \text{R}_{\text{landing}} - T \tag{88}$$

$$\text{R}_{\text{free roll}} = \frac{\text{D}_{\text{touchdown}} + \text{D}_{\text{braking}}}{2} + \mu\left(\text{W}_{\text{landing}} - \text{L}_{\text{touchdown}}\right) - T \tag{89}$$

$$\text{R}_{\text{braking}} = \text{D}_{\text{braking}} + R_{\text{brakes}} - T \tag{90}$$

$$\text{R}_{\text{braking}} = \frac{\text{D}_{\text{braking}}}{2} + \mu\text{W}_{\text{landing}} - T \tag{91}$$

Landing thrust is either zero for a dead stick landing or sizably negative if power is on and propellers are reversed. The drag terms should be evaluated at both touchdown and braking speeds. So should landing roll resistance terms. Rolling resistance coefficient, μ, will be around 0.02 in the free rolling case and around 0.5 with braking applied on a dry surface or 0.3 on a wet carrier deck.

Reserve

Required reserves will consist of enough fuel to fly to an alternate or for a 45 minute loiter at either high or low altitude. Final weight at the end of this segment will be landing weight, so use the Breguet Endurance Equation.

$$E = \frac{1}{sfc_{loiter}}\left(\frac{C_L}{C_{D_T}}\right)\ln\left(\frac{W_{\text{initial}}}{W_{\text{final}}}\right) \tag{92}$$

$$W_{\text{initial}} = W_{\text{final}} e^{\left(\frac{Esfc_{\text{loiter}}}{\frac{L}{D}}\right)} \tag{93}$$

Both C_L and C_{D_T} must be estimated to evaluate the previous equation. For propeller driven aircraft, best endurance speed occurs at the speed corresponding to $(C_L^{3/2}/C_{DT})_{max}$, or for simplicity, at the speed corresponding to $(C_{DT}/C_L^{3/2})_{min}$. Taking the derivative of this with respect to C_L yields

$$\frac{C_{D_T}}{C_L^{\frac{3}{2}}} = C_{D_0} C_L^{-\frac{3}{2}} + \frac{C_L^2 C_L^{-\frac{3}{2}}}{\pi e AR} \tag{94}$$

$$\frac{C_{D_T}}{C_L^{\frac{3}{2}}} = C_{D_0} C_L^{-\frac{3}{2}} + \frac{C_L^{\frac{1}{2}}}{\pi e AR} \tag{95}$$

$$\frac{d}{dC_L} = -\frac{3}{2} C_{D_0} C_L^{-\frac{5}{2}} + \frac{1}{2}\frac{C_L^{-\frac{1}{2}}}{\pi e AR} = 0 \tag{96}$$

$$\frac{d}{(dC_L)^2} = -3C_{D_0} C_L^{-\frac{5}{2}} + \frac{1}{2}\frac{C_L^{-\frac{1}{2}}}{\pi e AR} = 0 \tag{97}$$

$$C_L = \sqrt{3C_{D_0}\pi e AR} \tag{98}$$

Given C_L and landing weight, airspeed may be calculated and distance covered will be

$$\Delta s_{\text{reserve}} = V_{\text{reserve}} E_{\text{reserve}} \tag{99}$$

Reserve distance covered isn't something that will be flown every mission so it shouldn't be subtracted from transit back distance. Initial reserve segment weight will be the final weight for the descent segment.

Descent

The descent segment will consist of a powered descent from final transit altitude to an intermediate altitude which will be close to sea level unless otherwise specified in the mission requirement. Descent will be at constant dynamic pressure, best descent rate, or at some constant rate of descent such as 500 feet per minute. Power will be constant over the altitude band considered and will probably be at flight idle (flight idle at high altitudes may be close to full throttle). Given a high lift-to-drag ratio, descent power may not be necessary to achieve substantial down-range or cross-range requirements to reach an alternate airfield. If power is at flight idle for descent, then

$$\Delta W_{\text{descent}} = sfc_{\text{flight idle}} T_{\text{flight idle}} \Delta t_{\text{descent}} \tag{100}$$

What must be determined next is descent time and one of two methods may be used. The first method will be similar to that used for determination of climb time. Calculations are reversed from those of the climb segment. One difference exists between calculating descent fuel burn and calculating climb fuel burn. The descent segment is being calculated backwards starting with a known weight at the end of the segment. Calculate the final rate of descent at the known weight and then continue working backward. Choose a small value of altitude, say 1000 feet, and calculate the new rate of descent assuming a constant descent angle over that altitude change. Calculate descent time, fuel burn and horizontal distance covered as

$$\Delta t_{\text{descent}} = \frac{\Delta h}{\dfrac{R/D_{\text{start}} + R/D_{\text{end}}}{2}} \tag{101}$$

$$\Delta W_{\text{descent}} = \Sigma\left(sfc_{\text{flight idle}} T_{\text{flight idle}} \Delta t_{\text{descent}}\right) \tag{102}$$

$$\Delta s_{\text{descent}} = \left[\left(V_{\text{descent}}\right)_{\text{start}} - \left(V_{\text{descent}}\right)_{\text{end}}\right] \cos\gamma_{\text{descent}} \Delta t_{\text{descent}} \tag{103}$$

Total time for descent, fuel burned and horizontal distance covered will be

$$\left(t_{\text{descent}}\right)_{\text{total}} = \Sigma\Delta t_{\text{descent}} \tag{104}$$

$$\left(\Delta W_{\text{descent}}\right)_{\text{total}} = \Sigma\Delta W_{\text{descent}} \tag{105}$$

$$\left(\Delta s_{\text{descent}}\right)_{\text{total}} = \Sigma\Delta s_{\text{descent}} \tag{106}$$

The other method of calculating descent fuel burn assumes a constant rate of descent at some figure such as 500 fpm. In this case, descent time will be

$$\Delta t_{\text{descent}} = \frac{\Delta h}{R/D_{\text{constant}}} \tag{107}$$

In order to calculate descent distance, which is a function of the horizontal component of the velocity vector, it will be necessary to calculate descent angle.

$$\Delta s_{\text{descent}} = \Delta t_{\text{descent}} \left(R / D_{\text{constant}}\right) \cot\gamma_{\text{descent}} \tag{108}$$

$$\Delta s_{\text{descent}} = \Delta h \cot\gamma_{\text{descent}} \tag{109}$$

The descent angle for small altitude bands may be found similarly to the way climb angle was found. Total descent distance and fuel burn may then be summed as before.

Transit from Station

The final weight for this segment will be the initial weight for descent. Once again, transit will take place at best range speed and transit distance will be equal to the required distance less the distance covered during descent.

$$R = \frac{V_{cruise}}{sfc_{cruise}} * \left(\frac{L}{D}\right)_{cruise} * \ln\left(\frac{W_{initial}}{W_{initial} - W_{final}}\right)_{cruise} \tag{110}$$

The lift-to-drag ratio to be used above may be estimated as

$$\left(L / D\right)_{\text{transit}} = \sqrt{\frac{\pi e_{\text{airplane}} AR}{4 C_{D_0}}} \tag{111}$$

The transit distance (R in the equation above) will probably be specified as a mission requirement. If it is, then deduct climb distance from it and solve the equation for final weight:

$$W_{initial} = \frac{W_{\text{final}} e^{\frac{V_{cruise}}{sfc_{cruise}} * \left(\frac{L}{D}\right)_{cruise}}}{1 - e^{\frac{V_{cruise}}{sfc_{cruise}} * \left(\frac{L}{D}\right)_{cruise}}} \tag{112}$$

and best range speed may be found using

$$V_{\text{best range}} = \sqrt{\frac{2}{\rho \sqrt{C_{D_0} \pi e_{\text{airplane}} AR}}} \sqrt{\frac{W}{S_{\text{reference}}}} \tag{113}$$

with wing loading varying over the transit distance. Since both initial and final weights for this segment are now known, average best range speed may be estimated and transit time and distance will be

$$\Delta t_{\text{transit}} = \frac{R}{\left(V_{\text{best range}}\right)_{\text{average}}} \tag{114}$$

$$\Delta W_{\text{transit}} = W_{initial} - W_{final} \tag{115}$$

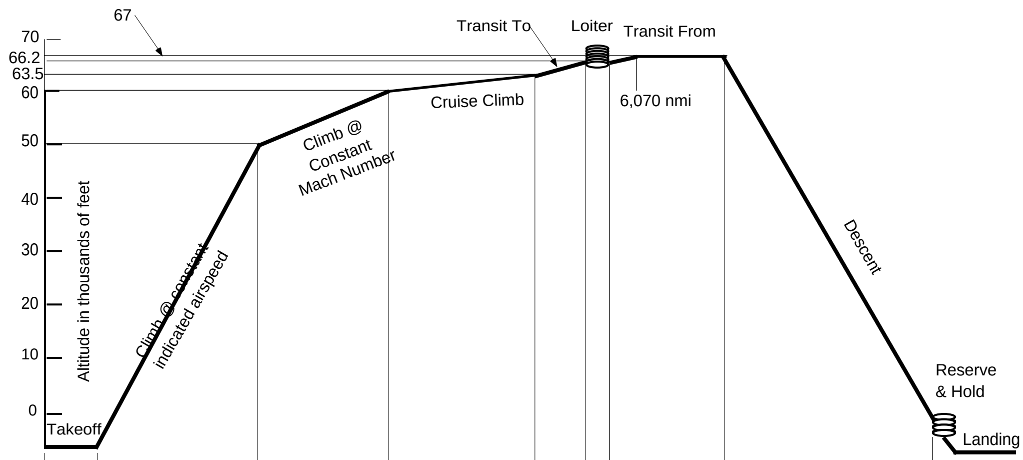
Loiter

Loiter will almost always involve flight at best endurance speed unless there's either a requirement to cover a specific distance or winds aloft are high enough that station-keeping is a problem. The Breguet Endurance Equation can be used for calculations From ref 10 (page 456):

$$E = \frac{1}{sfc_{loiter}} \left(\frac{C_L}{C_{D_T}} \right) \ln \left(\frac{W_{\text{initial}}}{W_{\text{final}}} \right) \tag{116}$$

Both weights are known and E as used here is time-on-station.

Figure 13 is a typical mission profile for a turbine-powered high altitude long endurance (HALE) aircraft. The performance numbers at the bottom were added as the configuration progressed through the above process. This type of presentation of results is typical of what's presented in industry briefings on aircraft sizing and performance and serves as an excellent way to summarize calculations to date. It's equally important to summarize initial sizing assumptions as well.



RPV with 500# payload & fan engine:

Cumulative Distance	n.mi.	0	568 ft	60	158	3,004	3,004	5,393	8,196	8,296	8,397	284 ft
True Airspeed	knots	0	135	345	344	344	344	344	344	135	135	0
Elapsed Time	hours	0	30 sec	0.248	0.533	8.807	8.807	15.753	23.901	24.380	25.130	3 sec
Weight	pounds	4,647	4,631	4,509	4,475	3,793	3,793	3,332	2,843	2,789	2,753	2,747

Figure 13. AHALE Mission Profile is Typical of Transport, Science, and Bombing Category Aircraft.

The most important idea conveyed in this chapter is to regard the early conceptual design phase as an examination of the interactions of basic design parameters and develop a feel for how they interact in theory. Later chapters will provide the opportunity to quantify the general descriptions developed here.

CHAPTER 3
DEFINE THE DESIGN DOMAIN

Given the algebraic expressions of mission segments just formulated, it is possible to begin quantitatively defining the design domain. There are several methods for doing this, one of which is presented in ref Markus on parametrics. The constraint method presented in this section was developed in the 1940s by staff at the Douglas Aircraft Company and their paper is reprinted here in its entirety except for references which are listed with other text references.[1]

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Introduction

The knowledge gained by aircraft operators through experience during the past decade has increased many fold, and with it have come a larger and more demanding series of requirements with which the new airplane must be able to comply. The number of these requirements to be met simultaneously makes necessary some systematic method of handling the problem.

In the preliminary stage the designer is faced with the task of selecting the major design variables so that the combination will result in a configuration that will meet desired specifications. The major design variables are considered as the powerplant, wing area, wing geometry, and weight. There are naturally many other factors involved in the design of aircraft, but the above are fundamental variables to be considered.

The specifications represent the desires of the operator with respect to speed, range, takeoff and landing distances, cost of operation, and safety. To help insure this safety, the Civil Aeronautics Authority[2] together with the operators and designers, have developed a set of regulations designed to insure a safe margin of performance in the event of an emergency. The regulations state that an airplane must be able to maintain certain rates of climb with the critical engine inoperative and with all engines operating. Various conditions are set up to be met during the takeoff, enroute, and during the approach and landing. All the specifications and requirements must be met in order for the airplane to be acceptable.

It is apparent that if the critical design conditions are met, other requirements may be exceeded. Also, an increase in one specific item of speed is increased by decreasing the wing area, the rate of climb, range, takeoff and landing distances may suffer. On the other hand, if the wing area is increased to shorten takeoff and landing distances, the improvement may be made at the expense of speed. It is evident that the problem is too complex for a cursory examination.

A method whereby the effect of changes in the major design variables may be evaluated, as well as the limits within which the airplane may be built and meet specifications and regulations, greatly facilitates the problem. This paper presents one such method leading to a graphical solution which is presented in such a fashion that the limits may be determined and the effect of design changes shown.

This method may be considered a tool with which a special job is done, and as with any tool its utility greatly depends on the skill of the user. The range of major variables chosen for the investigation is no barrier to the final solution; but a judicious choice, based on sound technical knowledge and experience, considerably affects the time and labor involved.

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[1] Cherry, H.H. & Croshere, A.B., Jr, "An Approach to the Analytical Design of Aircraft", *S.A.E. Journal* August, 1947 also S.A.E. Quarterly Transactions 1947.

[2] Redesignated Federal Aviation Administration in 1958.

At the outset let it be understood that it is the purpose of this paper to illustrate one method by which the effect of changes in a set of chosen variables upon an airplane's ability to meet a given set of conditions can be determined. No attempt is made to draw specific conclusions regarding the efficacy of one combination of variables over another.

Method of Analysis

Let us assume for the purpose of illustration that we are to analyze an airplane in the commercial land transport category. Specifications have been set forth for the following items of performance:

1. Stalling speed 85 mph @ sea level;
2. Cruising speed 230 mph at 10,000 feet at 60% normal rated power;
3. CAR takeoff field length 5,000 feet at sea level;
4. CAR landing field length 5,000 feet at sea level;
5. Range 1,600 miles with 20,000 pound payload at the specified cruising speed; and
6. Direct operating cost is not to exceed 20 cents per ton mile as per Air transport Association method in 1948 dollars.

Civil Air Regulations will, of course, apply and are considered as a part of the performance specifications.

The fuselage configuration, selection of flaps, arrangement of landing gear, engine installation, wing placement, etc., are of no immediate concern in aspect ratio: wing thickness, taper ratio, or range of values of wing areas or weights to be analyzed, while of importance to the final result, are not controversial items. In the preparation of the illustrations for this paper these variables have been chosen and are held constant throughout this analysis. The successful arrangement to suit the purposes for which the airplane is to be used is a test of the skill and ingenuity of the designer. The method of analysis herein proposed enables one to determine a satisfactory combination of the fundamental variables chosen.

The clearest way to approach the illustration of this method of analytical airplane design is to consider first the stalling speed. For instance, let us determine the combination of weights and wing areas that will give the specified stalling speed of 85 mph for a chosen type of flap when fully deflected. From basic data, $C_{l_{\max}}$ may be obtained, and the stalling speed for various wing area and weight combinations may be calculated from the standard equations.[3] These may be plotted in the form of V_{stall} versus Weight for selected values of wing area as shown in Figure 15.

Next, combinations of weights and wing areas giving constant values of stalling speed can be determined from Figure 6 and plotted against Weight and Wing area as shown in Figure 16.

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[3] See References.

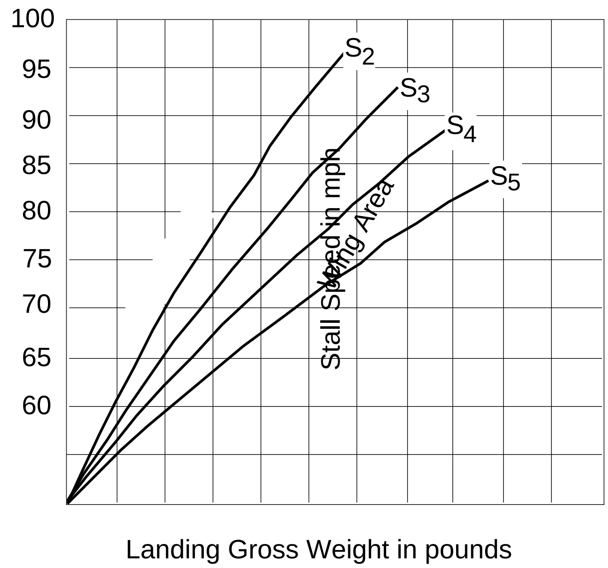


Figure 15. Full-Flap Sea Level Stall Speed versus Landing Gross Weight.

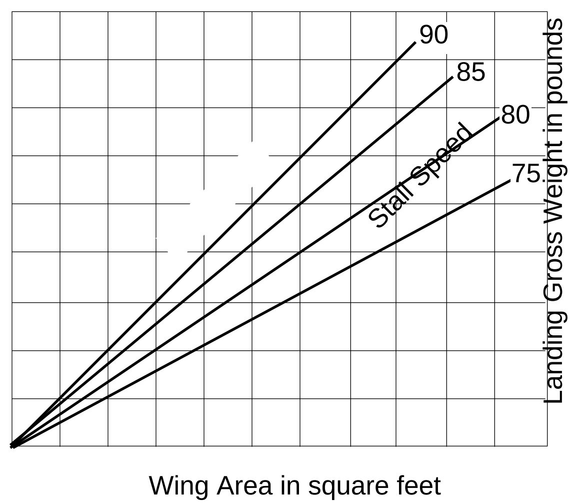


Figure 16. Landing Gross Weight versus Wing Area for Constant Full-Flap Sea Level Stall Speed.

From this plot, it is easily seen that for the chosen flap type, the combinations of weight and wing area above the contour for 85 mph will have an excessive stalling speed. We have now determined one limit beyond which the airplane will fail to meet a specification. The limiting condition is shown on the summary curve in Figure 17.

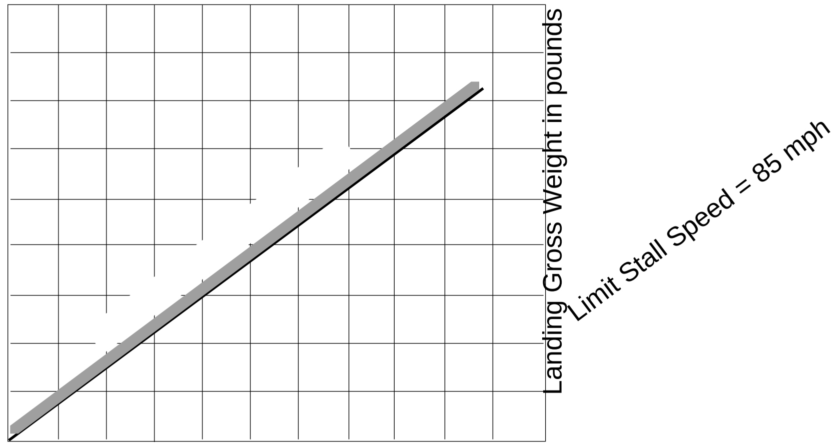


Figure 17. Summary Curve for Full-Flap Landing Stall Speed Limit.

As the term "summary curve" will be used throughout the presentation, it is desirable to define it. A summary curve is a statement of the limiting condition to meet an imposed condition for each wing area.

Obviously, it is possible to prepare Figure 17 in a more direct fashion merely by applying the standard stalling speed equation. However, the above procedure was described in detail in order to illustrate the method that must be used to determine more complicated conditions.

Next, let us consider the cruising speeds. They may be determined in much the same manner as described for stalling speeds. First, it is necessary to assemble the basic data needed to compute the power required for the range of weight and wing area combinations chosen. For the selected engine and propeller combination, the power available for any operating condition can be found. Speeds for these weights and wing areas, at the operating conditions specified, can be determined by standard performance procedures.[4] It is convenient to plot the results as Speed versus Weight for the selected values of wing area as shown in Figure 18.

As with the stalling speeds, the combinations of weight and wing area giving constant values of cruising speeds can be determined from Figure 18. These combinations can be plotted on the summary curve. The contours with the specified speed as a limit are shown in Figure 19.

[4] See References.

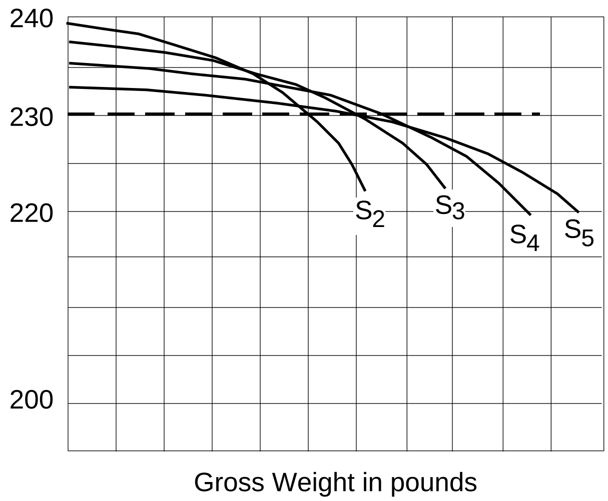


Figure 18. Cruise Airspeed versus Gross Weight at Cruise Altitude & Power.

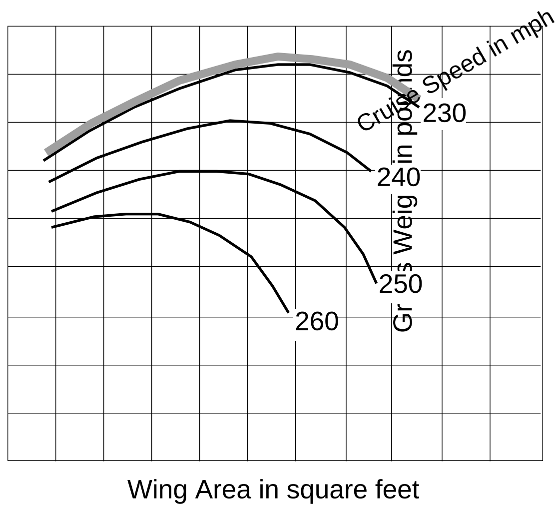


Figure 19. Cruise Speed Variation with Gross Weight and Wing Area.

It should be realized that, since there is no explicit equation for directly computing cruising speeds in terms of weight and area, it is necessary to prepare an intermediate step as shown in Figure 9 to arrive at the result presented in the summary curve.

We have now determined another of the limiting conditions for the design. If desired the same procedure may be followed to determining speeds for various powers and altitudes.

Let us now consider the takeoff field length requirement. The distance to clear a 50 foot obstacle is calculated with the assumption that an engine failure occurs at the critical point along the takeoff path according to Civil Air Regulations.[5] The computation is made by standard methods for various weights and wing areas at the optimum flap angle. The results are most conveniently plotted as Takeoff Field Length versus Gross Weight for the selected values of wing areas as shown in Figure 20.

[5] See References.

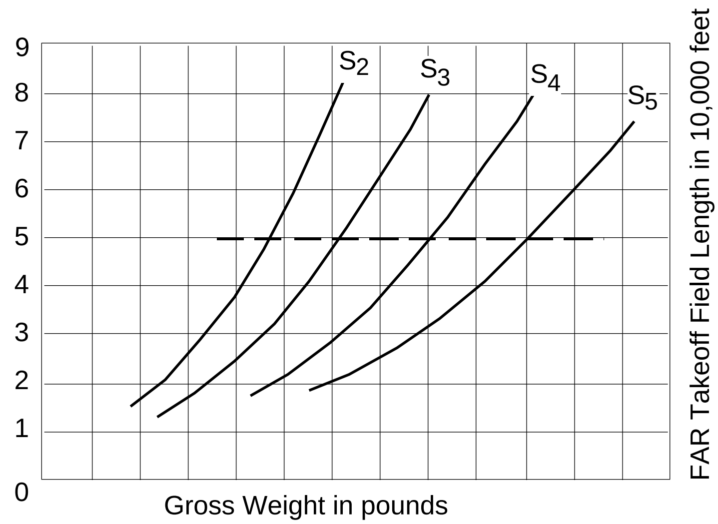


Figure 20. Takeoff Field Length versus Gross Weight

The intersection of the curves with the desired or specification field length represents the combination of weights and wing areas which will just meet the condition. Combinations for the other constant values of takeoff distances may also be determined. These lines of constant distances may be plotted as Weight versus Wing Area on the summary curve as in Figure 21. Curves of constant landing field lengths may be determined and plotted on a summary plot in the same fashion.

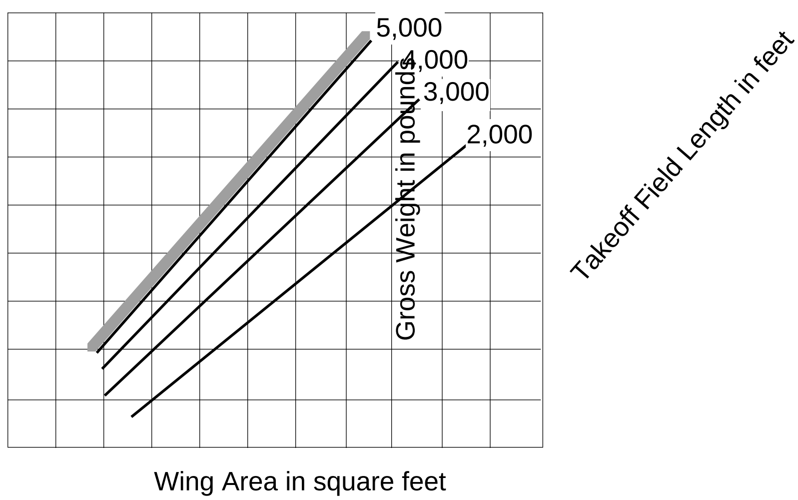


Figure 21. Summary Curve for Takeoff Field Length.

The next item on the specification list is the range. It can be handled in the same general manner as other items of performance although some additional basic data are needed. From layouts of the proposed airplane, with chosen values of wing areas and gross weights, values of weight empty can be determined. A typical plot of Weight Empty versus Gross Weight for the chosen wing areas is shown in Figure 22.

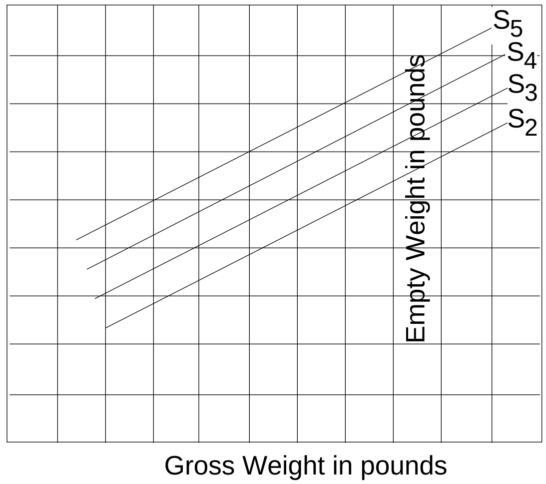


Figure 22. Weight Empty versus Gross Weight Variation with Wing Area.

From this plot and other basic data, and for the specified payload, the available fuel can be determined for each weight and wing area. From basic engine curves and previous work the fuel consumption and speeds for a given operating condition are known. With these, range can be computed for various gross weights and the wing areas selected. The values are plotted as Range versus Gross Weight for the constant values of wing area as in Figure 23.

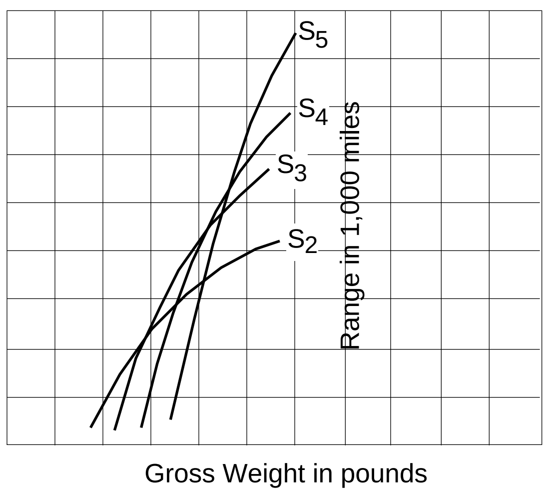


Figure 23. Range versus Gross Weight for Several Wing Areas.

From Figure 23, the combinations of weights and wing areas for constant values of range can be determined and plotted on the summary curve in Figure 24. Limiting values of weight and area to meet the specification of 1,600 miles minimum range are indicated on the plot.

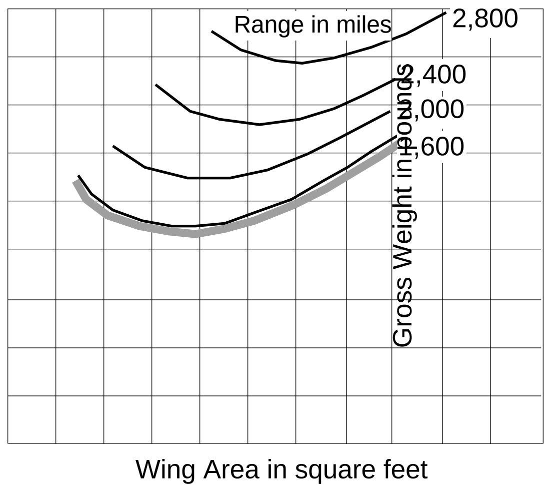


Figure 24. Summary Curve for Range.

The direct operating cost per ton-mile can readily be obtained. It is necessary to assemble all data required in the solution of the A.T.A. cost equations.[6] Since the range and cruising speeds are known, the block speeds can be determined and the dollars per hour operating cost computed. For the constant payload, the dollars per ton-mile are found and plotted versus gross weight for the chosen wing areas. Following the general plan, combinations of weights and wing areas giving constant values of dollars per ton-mile cost are plotted on a summary curve as in Figure 25.

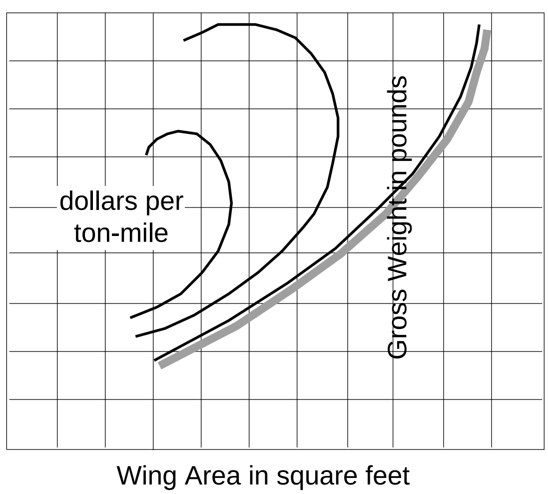


Figure 25. Summary Curve for Cost.

The foregoing discussion is concerned only with the desired specifications set forth by the operator. No mention has yet been made as to the limits imposed by Civil Air Regulations. These important limits will be considered now.

One of the Civil Air Regulations (CAR 04.1231) requires that an airplane of more than 60,000 pounds gross weight be able to maintain at 5,000 feet altitude a rate of climb in feet per minute of not less than $0.04*V_{S_0}^2$ where V_{S0} is the stalling speed in mph with the airplane in the landing configuration. This regulation will serve to describe the method of determining the limiting weights and wing areas for all C.A.R. requirements.

6 See References.

Using the basic data, power required curves can be prepared for selected weights and wing areas, imposing any conditions required by regulations. Power available curves are prepared with the engines operating as prescribed in the regulations. Actual rates of climb are computed by standard methods and plotted as Climb versus Weight for constant wing areas as in Figure 26.

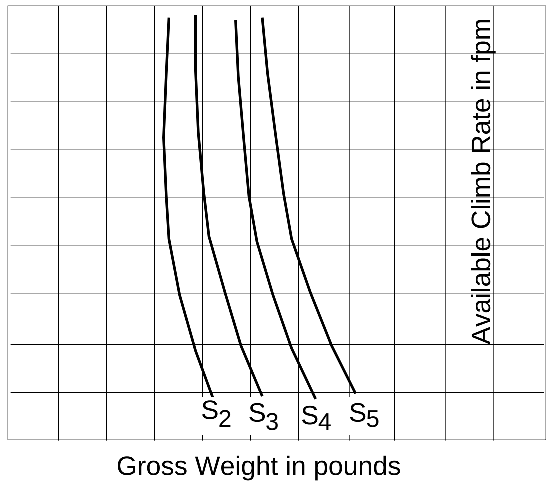


Figure 26. Actual Climb versus Gross Weight for Constant Wing Areas.

Since, from basic data, V_{S0} is known for the various weights and wing areas, the required rates of climb equal to $0.04 * V_{S_0}^2$ can be computed and plotted as shown in Figure 27. By comparing Figure 26 with Figure 27, the weights at which the required climb and the actual climb are equal can be determined for each wing area as in Figure 28. These weight and wing area combinations are plotted on the summary curve as in Figure 29. This curve represents limiting combinations to meet this regulation.

This same procedure can be followed to determine all C.A.R. limits, but the following caution must be observed. Certain of the regulations are not based on takeoff weight. When this is the case the specified limiting weights as determined by the above method must be adjusted. for instance, when the regulation is based on the landing weight, the takeoff weight may be as much greater as the amount of fuel burned or dumped. This may be handled easily by preparing curves showing Landing Weight versus Takeoff Weight for the selected wing areas.

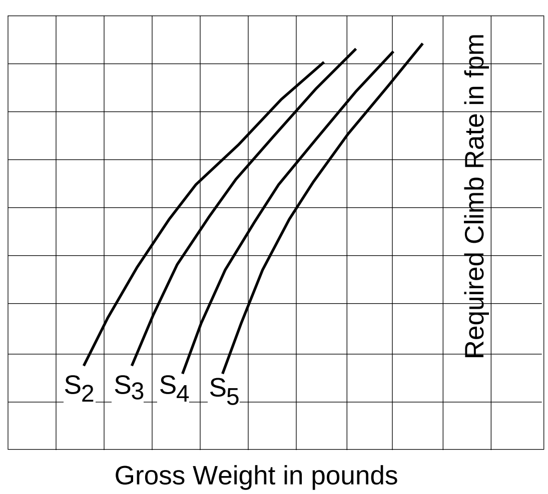


Figure 27. Required Rates of Climb versus Gross Weight.

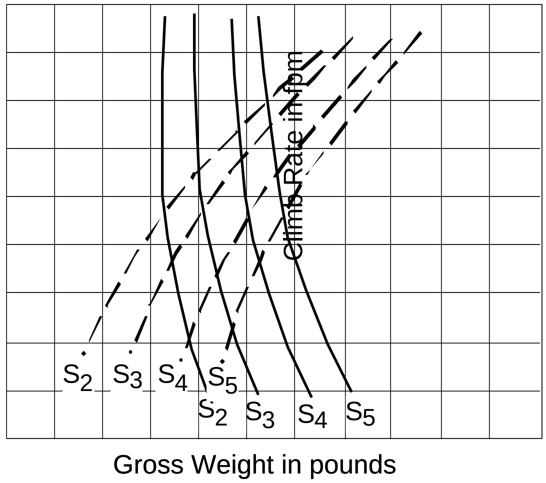


Figure 28. Actual/Required Rates of Climbs for Various Wing Areas.

So far, this discussion has covered methods for determining the limiting combinations of takeoff weights and wing areas which will meet or exceed both the specifications and Civil Air Regulations. Before continuing, it should be realized that there may be important physical limits which can greatly affect the design. For instance, certain combinations of weights and wing areas that would otherwise be suitable may be impossible or undesirable because of fuel capacity limitations, excessive center-of-gravity movement, etc. The limitations imposed by each of these physical properties can be determined and expressed as a function of weight and wing area so that they can be included on the summary plot. Time does not permit a full discussion of these various limits, and for the purposes of this paper they are assumed not to be critical.

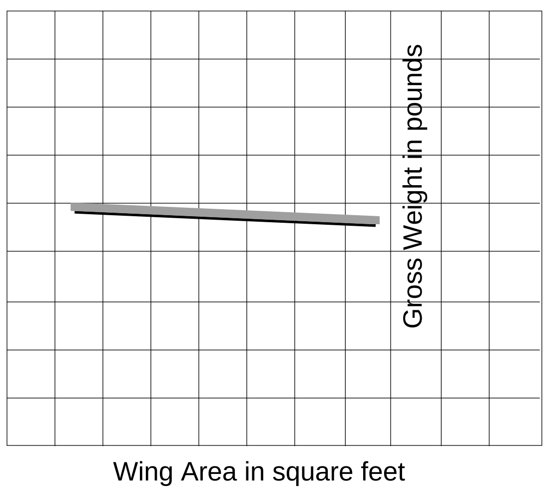


Figure 29. Summary Curve for Climb Rate.

The preceding discussion has defined certain limiting conditions independently for each requirement, and separate summary curves have been prepared. It is now desirable to present all the design limits in their relationship to each other on a single summary plot. Such a plot is shown in Figure 30.

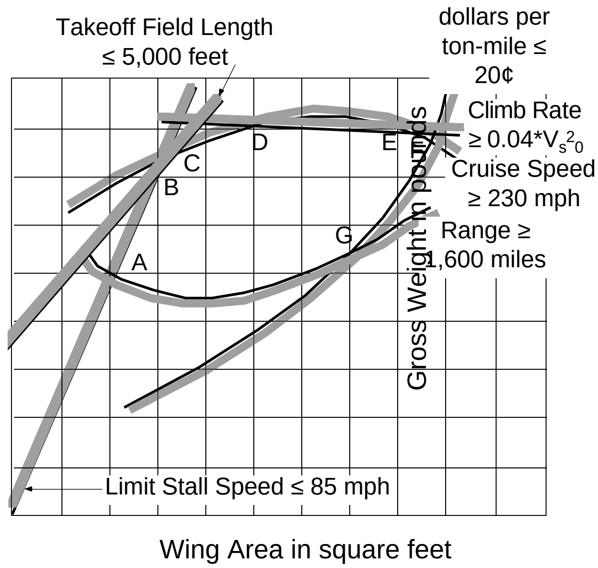


Figure 30. Summary Plot of Constraint Curves for Each Requirement.

A study of Figure 30 reveals that with the chosen design variables there are six limiting conditions affecting the design of the airplane. Each of these limits a particular combination of weight and wing area. It is seen that the 1,600 mile range requirement limits the weights and wing areas between A and G, the operating cost between B and C, the cruising speed between C and D, the $0.04*V_{S_0}^2$ CAR engine-out enroute climb between F and G, the 5,000 foot CAR takeoff field length between B and C, and the stalling speed between A and B. Only within the polygon A, B, C, D, E, F, G are found combinations of takeoff weights and wing areas which will meet or exceed the imposed requirements.

It will be noted that with the chosen design variables there is a wide variation of permissible combinations of weights and wing areas that will meet or exceed all requirements. The problem now becomes one of selecting that weight and wing area which defines an airplane having the most desirable abilities. Let us now examine the characteristics of the airplanes within the permissible range.

Figure 31 is an enlarged reproduction of the polygon and includes contours of those performance characteristics which will affect our choice. For simplicity of illustration we will assume that only three performance items will be involved. These include the range, cruising speed, and takeoff distance.

Three selections are of immediate interest. The airplane defined by point A has the greatest cruising speed, that by point B has the shortest takeoff distance, while point D has the greatest range. Since any point within the boundaries of the polygon will exceed all the imposed limitations, it is obvious that someone must decide to what extent each item or items of performance should be exceeded. The success of the proposed design, in a competitive market, is vitally affected by this decision. This fact cannot be overemphasized. Innumerable factors affecting this decision are outside the realm of the airplane designer; they concern the manufacturer's management, the sales organization, and the prospective operators. If that intangible quality of judgement could be expressed mathematically in terms of desired improvements over the minimum requirements, it would be possible to include on the summary plot contours of equally desirable characteristics. It follows that this type of analysis could lead to the determination of the point defining an airplane with the most desirable abilities.

Unfortunately, we have not yet arrived at this state of agreement on airplane performance, nor does it seem probable that such an agreement can be reached. One operator, because of his particular problems, would be concerned mainly with extending the range. The desired improvements also fluctuate with time, with the desire for increased speed giving way to lower cost of operation, or vice versa, as the economic picture changes.

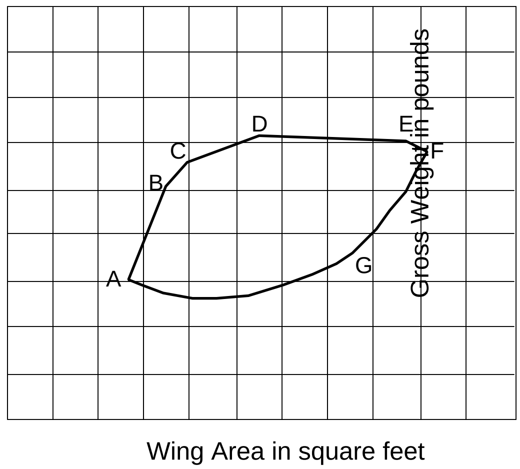


Figure 31. Blowup of Summary Curve Design Domain.

This approach to the analytical design of aircraft does, however, focus the attention on only those combinations which will meet or exceed the minimum requirements. These data are furnished so that the effect of improvement in one item of performance can be evaluated in terms of the changes in other abilities and a compromise agreement reached. Upon the judgement used in selecting the design features and the resulting performance abilities rest the ultimate success of the proposed design.

Finally, it should be noted that if the minimum requirements had been made more severe, the choice of engines, wing geometry, arrangement, etc. might have been unsatisfactory. For instance, if the range requirement had been 2,400 miles, and the cruising speed requirement been 245 miles per hour, the design variables chosen for the example would permit no combination of weight and wing area that would be enclosed by the limiting conditions. In this case different engine power, wing geometry, or arrangements would have to be investigated in an effort to determine some combinations of weight and wing area that would meet the minimum requirements.

It will be remembered that in the beginning certain design variables were chosen and held constant throughout the analysis. A different choice of engines, propellers, aspect ratio, wing thickness, taper ratio, or arrangement might have resulted in a greater possible improvement over minimum requirements. In general, even though a positive area in the summary plot is obtained, additional investigation should be undertaken until the designer has satisfied himself that he has arrived at the optimum design.

The method described in this paper is as accurate as the calculation of the performance of a single airplane.

It is not a simple shortcut method; rather, it requires the expenditure of considerable time and effort to achieve adequate solutions. It is felt, however, that as a first stop in the design of successful airplanes to meet present demanding requirements, such a process is well worth the effort expended.

CHAPTER 4
MULTI-VIEW LAYOUTS
Conceptual Design Steps Prior to a Layout

Previous chapters have helped formulate sets of equations which could be used to flesh out a specific point in the design domain and introduced a thought process conducive to gaining a feel for the important aspects of a design. In order to create a first sketch, it will be necessary to estimate several basic design parameters such as lift and drag coefficients, weights, and propulsion characteristics.

Now that theory has been examined, the design domain defined, and interesting points identified for further exploration, one or more of these points must be fleshed out in enough detail to create a configuration. Going back through the methods and inserting numbers is perhaps the best way to get started here.

Once the weight fraction method has provided a closed value of takeoff gross weight and empty weight, other configuration design parameters may be defined. Keep track of assumptions and airplane parameters which have been calculated or estimated so far:

Weights

$TOGW$

$W_{\text{start of climb}} = W_{\text{takeoff end}}$

$W_{\text{end of climb}} = W_{\text{transit to start}}$

$W_{\text{transit to end}} = W_{\text{loiter start}}$

$W_{\text{loiter end}} = W_{\text{transit from start}}$

$W_{\text{transit from end}} = W_{\text{descent start}}$

$W_{\text{descent end}} = W_{\text{reserve + landing start}}$

$W_{\text{reserve + landing end}} = W_{\text{empty}}$

W_{fuel}

W_{payload}

Aerodynamics

L/D_{cruise} T/W_{cruise}

e_{airplane}

$C_{D_{\min}} = C_{D_0}$

$C_{L_{\text{best endurance}}}$

Propulsion

$tsfc_{\text{cruise}}$

L/D_{cruise} T/W_{cruise}

Mission Performance

h_{cruise} ρ_{cruise}

V_{cruise}

$V_{\text{best endurance}}$

$R_{\text{transit to}}$

$R_{\text{transit from}}$

E_{loiter}

Geometry

λ

AR

Λ

Given one known segment in the mission, say loiter, we can solve for lift:

$$L_{\text{loiter}} = \frac{W_{\text{loiter start}} + W_{\text{loiter end}}}{2} = \frac{\rho_{\text{cruise}}}{2} V^2_{\text{best endurance}} S_{\text{reference}} C_{L_{\text{best endurance}}} \tag{117}$$

then calculate wing area.

$$S_{\text{reference}} = \frac{W_{\text{average loiter}}}{\frac{\rho_{\text{cruise}}}{2} V_{\text{best endurance}}^2 C_{L_{\text{best endurance}}}} \tag{118}$$

Also calculate wing loading at loiter.

$$\frac{W_{\text{average loiter}}}{S_{\text{reference}}} = \frac{0.5\left(W_{\text{loiter start}} + W_{\text{loiter end}}\right)}{S_{\text{reference}}} = \frac{\rho_{\text{cruise}}}{2} V_{\text{best endurance}}^2 C_{L_{\text{best endurance}}} \tag{119}$$

Back this down to sea level using calculated weight fractions from previous mission segments.

$$\frac{W_{\text{average loiter}}}{S_{\text{reference}}} = \frac{0.5\left(W_{\text{loiter start}} + W_{\text{loiter end}}\right)}{S_{\text{reference}}} = \frac{\rho_{\text{cruise}}}{2} V_{\text{best endurance}}^2 \sqrt{\frac{C_{D_0} \pi e AR}{2}} \tag{120}$$

$$\frac{W_{\text{takeoff}}}{S_{\text{reference}}} = \frac{1}{\frac{W_2}{W_1}\frac{W_3}{W_2}\frac{W_4}{W_3}\frac{W_5}{W_4}} \frac{W_{\text{average loiter}}}{S_{\text{reference}}} \tag{121}$$

Do the same with Thrust.

$$T_{\text{required}} = \frac{\rho S_{\text{reference}} C_{D_0}}{2} V^2 + \frac{\rho S_{\text{reference}}}{2} V^2 \frac{1}{\pi e AR} \left(\frac{2W^2}{\rho S_{\text{reference}} V^2}\right)^2 \tag{122}$$

$$T_{\text{required}} = \frac{\rho S_{\text{reference}} C_{D_0}}{2} V^2 + \frac{1}{\pi e AR} \left(\frac{2W^2}{\rho S_{\text{reference}} V^2}\right) \tag{123}$$

$$T_{\text{required}} = \frac{\rho S_{\text{reference}} C_{D_0}}{2} V^2 + \frac{2}{\rho \pi e AR} \left(\frac{W^2}{S_{\text{reference}}}\right) \frac{1}{V^2} \tag{124}$$

and calculate the loiter thrust-to-weight ratio.

$$\left(\frac{T_{\text{required}}}{W}\right)_{\text{average loiter}} = \frac{\rho_{\text{cruise}} \left(C_{D_0}\right)_{\text{cruise}}}{2\frac{W_{\text{average loiter}}}{S_{\text{reference}}}} V_{\text{best endurance}}^2 + \frac{2}{\rho_{\text{cruise}} \pi e AR} \left(\frac{W_{\text{average loiter}}}{S_{\text{reference}}}\right) \frac{1}{V_{\text{best endurance}}^2} \tag{125}$$

Both average loiter weight and loiter thrust required (which equals thrust available by definition at this point) must be backed down to sea level values. Weight can be adjusted by applying weight fractions as before to get TOGW.

$$TOGW = \frac{W_{\text{average loiter}}}{\frac{W_2}{W_1}\frac{W_3}{W_2}\frac{W_4}{W_3}\frac{W_5}{W_4}} \tag{126}$$

$$TOGW = \frac{W_{\text{average loiter}}}{\frac{W_2}{W_1}\frac{W_3}{W_2}\frac{W_4}{W_3}\frac{W_5}{W_4}} \tag{127}$$

Loiter thrust available at best loiter altitude will be a large fraction of maximum thrust available, say 80% or greater. Call this fraction cruise thrust setting. Then

$$\left(T_{\text{available}}\right)_{\text{loiter}} = (\text{cruise throttle setting})\left[\left(T_{\text{available}}\right)_{\text{maximum}}\right]_{\text{cruise altitude}} \tag{128}$$

$$\left[\left(T_{\text{available}}\right)_{\text{maximum}}\right]_{\text{cruise altitude}} = \left[\left(T_{\text{available}}\right)_{\text{maximum}}\right]_{\text{sea level}}\sqrt{\sigma} \tag{129}$$

And sea level takeoff thrust-to-weight ratio is

$$\frac{\left[\left(T_{\text{available}}\right)_{\text{maximum}}\right]_{\text{sea level}}}{TOGW} = \frac{\dfrac{\left[\left(T_{\text{available}}\right)_{\text{maximum}}\right]_{\text{cruise altitude}}}{\sqrt{\sigma}}}{\dfrac{W_{\text{average loiter}}}{\frac{W_2}{W_1}\frac{W_3}{W_2}\frac{W_4}{W_3}\frac{W_5}{W_4}}} \tag{130}$$

The Boeing 707, the Douglas DC-8 and the ConVAir 880 all looked similar enough that telling them apart took some work. Sometimes, seeing the airline's livery was the best clue as to which was which: If it said Pan American, it was a 707; if it said United, it was a DC-8; if it said Trans World Airlines, it was an 880. But sometimes, airplanes can be designed to very similar requirements and not look similar at all. Examples are the North American P-51 Mustang, the Curtiss P-40 Warhawk, the Supermarine Spitfire, the Hawker Hurricane, the Lockheed P-38 Lightning, the Messerschmidt Bf-109, the Focke-Wulf FW-190, and the Yak-9. All were land-based air superiority fighters designed, built and fielded during the same time period. But all looked different for more than cosmetic reasons. So let your mind go and look at everything from flying wings to flying fuselages; whatever might potentially fulfill the mission requirements you've been given.

Some FAR Engine Sizing Requirements

Roskam Volume I (pp.140-148) presents FAR Part 25 sizing requirements for engines as follows:

For takeoff with one engine inoperative:

$$\left(\frac{T}{W}\right)_{\text{takeoff}} = \left(\frac{n_{\text{engines}}}{n_{\text{engines}}-1}\right)\left[\frac{1}{\left(\frac{L}{D}\right)_{\text{takeoff}}} + \text{climb gradient}\right] \tag{131}$$

evaluated at $V = 1.2\left(V_{\text{stall}}\right)_{\text{takeoff}}$

For landing:

$$\left(\frac{T}{W}\right)_{\text{landing}} = \frac{1}{\left(\frac{L}{D}\right)_{\text{landing}}} + \text{approach gradient} \tag{132}$$

evaluated at $V = 1.3\left(V_{\text{stall}}\right)_{\text{landing}}$

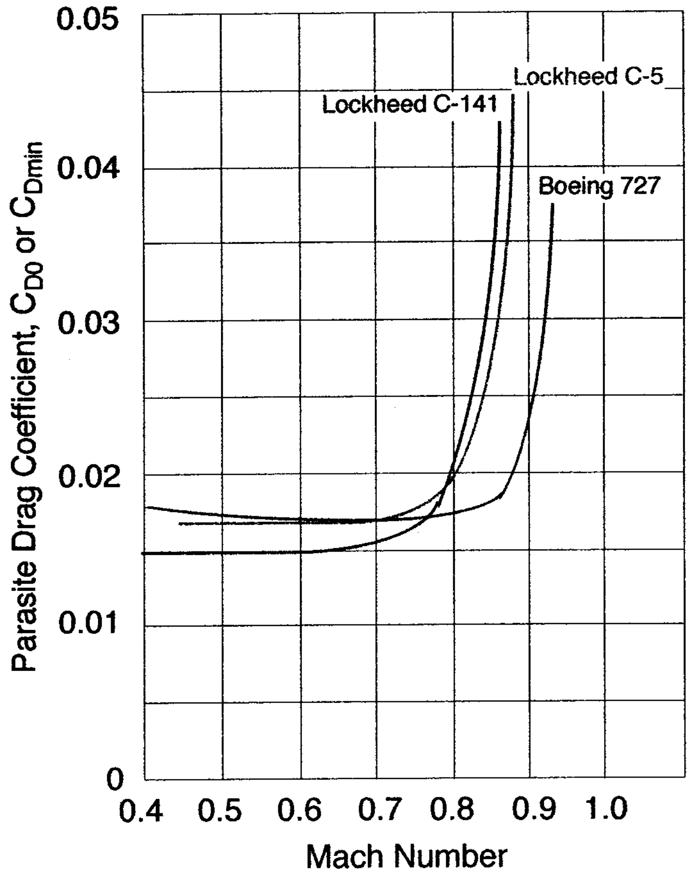
For balked landing (wave off) with one engine inoperative:

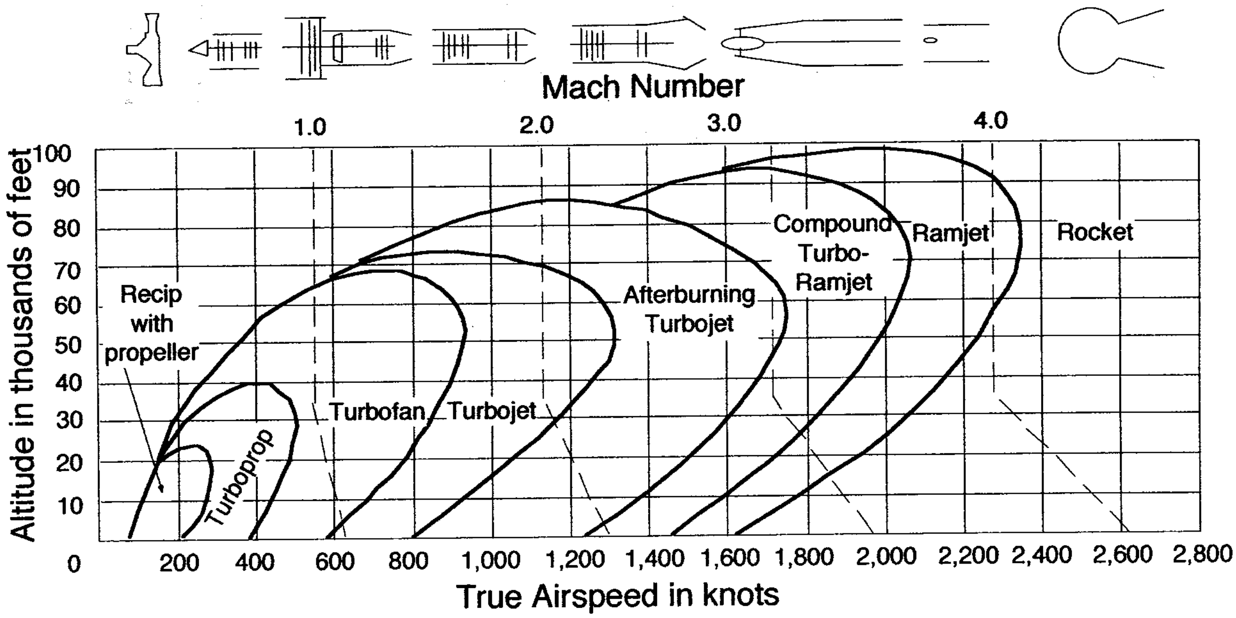
$$\left(\frac{T}{W}\right)_{\text{landing}} = \left(\frac{n_{\text{engines}}}{n_{\text{engines}} - 1}\right)\left[\frac{1}{\left(\frac{L}{D}\right)_{\text{landing}}} + \text{approach gradient}\right] \tag{133}$$

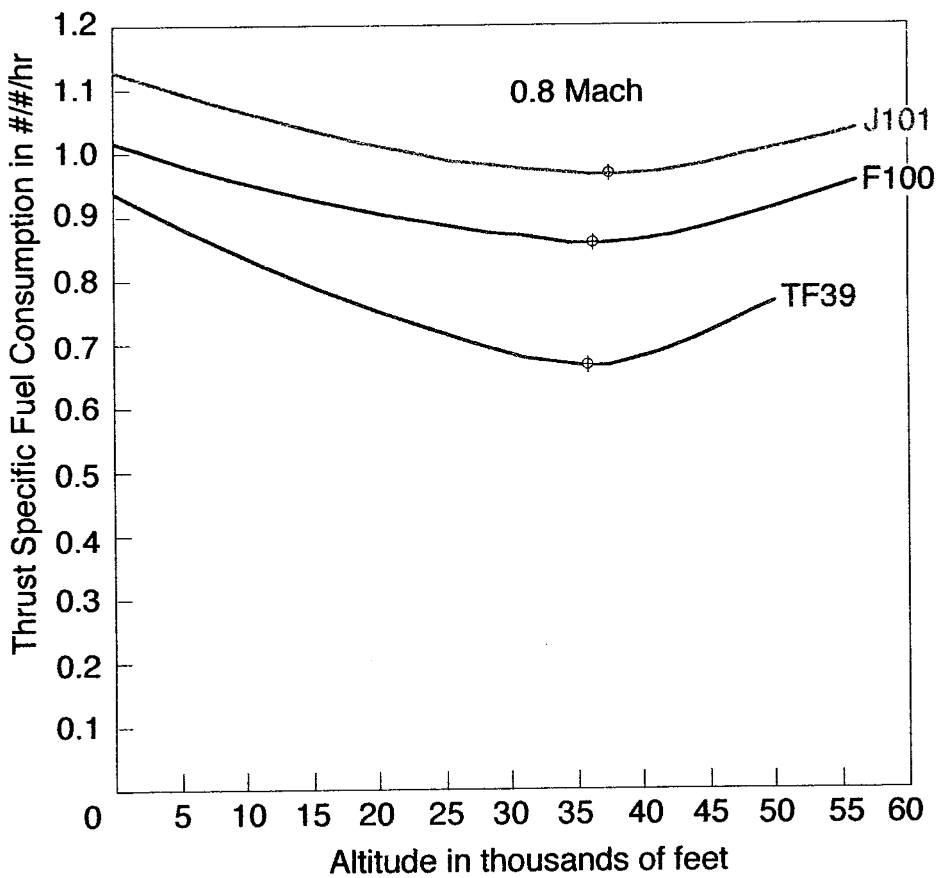
evaluated at $V = 1.5\left(V_{\text{stall}}\right)_{\text{landing}}$

CHAPTER 4
MULTI-VIEW LAYOUTS
Conceptual Design Steps Prior to a Layout

Previous chapters have helped formulate sets of equations which could be used to flesh out a specific point in the design domain. In order to start, it will be necessary to estimate several basic design parameters such as drag coefficients and propulsion characteristics.




Mach Number
1.0
2.0
3.0
4.0
Altitude in thousands of feet
100
90
80
70
60
50
40
30
20
10
0
Recip with propeller
Turboprop
Turbofan
Turbojet
Afterburning Turbojet
Compound Turbo-Ramjet
Ramjet
Rocket
200
400
600
800
1,000
1,200
1,400
1,600
1,800
2,000
2,200
2,400
2,600
2,800
True Airspeed in knots


0.8 Mach
J101
F100
TF39
Thrust Specific Fuel Consumption in #/#/hr
1.2
1.1
1.0
0.9
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
5
10
15
20
25
30
35
40
45
50
55
60
Altitude in thousands of feet

Pre-Layout Sketches

Once you've sketched one or more potential configuration concepts and done preliminary sizing of at least one, you'll want to make engineering drawings of at least one as a starting point for more detailed analyses. Figure 6 is reproduced from page 13 of a total of 20 pages of design notes, and this was the first configuration sketch made prior to beginning the multi-view drawing presented in Figure 7. The first thirteen pages addressed mission and performance requirements as well as engine specifics.

Early attempts at visualizing a configuration may take the form of partial views or two-point perspectives. Note in Figure 7 the wingtip float arrangement, augmenter bay locations, buried inlet details, and the presence of a lengthy fuselage spray dam to protect the engine inlets and augmenters from ocean spray. Note also explorations of horizontal/vertical tail locations and cockpit windows for VTOL or STOVL flight. In other words, use early sketches to explore any aspect of the layout that needs explanation and attention in a later multi-view drawing. Next, note the addition of salient cross-section shapes and the attention to hull shape.

Over the weekend and week which followed this set of calculations and sketches, the designer bought and built plastic kits of every flying boat available in 1/72 scale. It also helps overall understanding of specific design challenges to read up on interesting features of these existing aircraft and, in this case, on the basics of flying boat hull design. The results of the designer's detailed research were obvious in the details of the first layout.

The freehand sketches in Figure 6 are unsigned and undated, but the twenty page package they were in included both name, date, and title as a way of documenting who did what and when. Also documented in the package were results of conversations with other designers and users, what they recommended, and particularly what they didn't like and why.

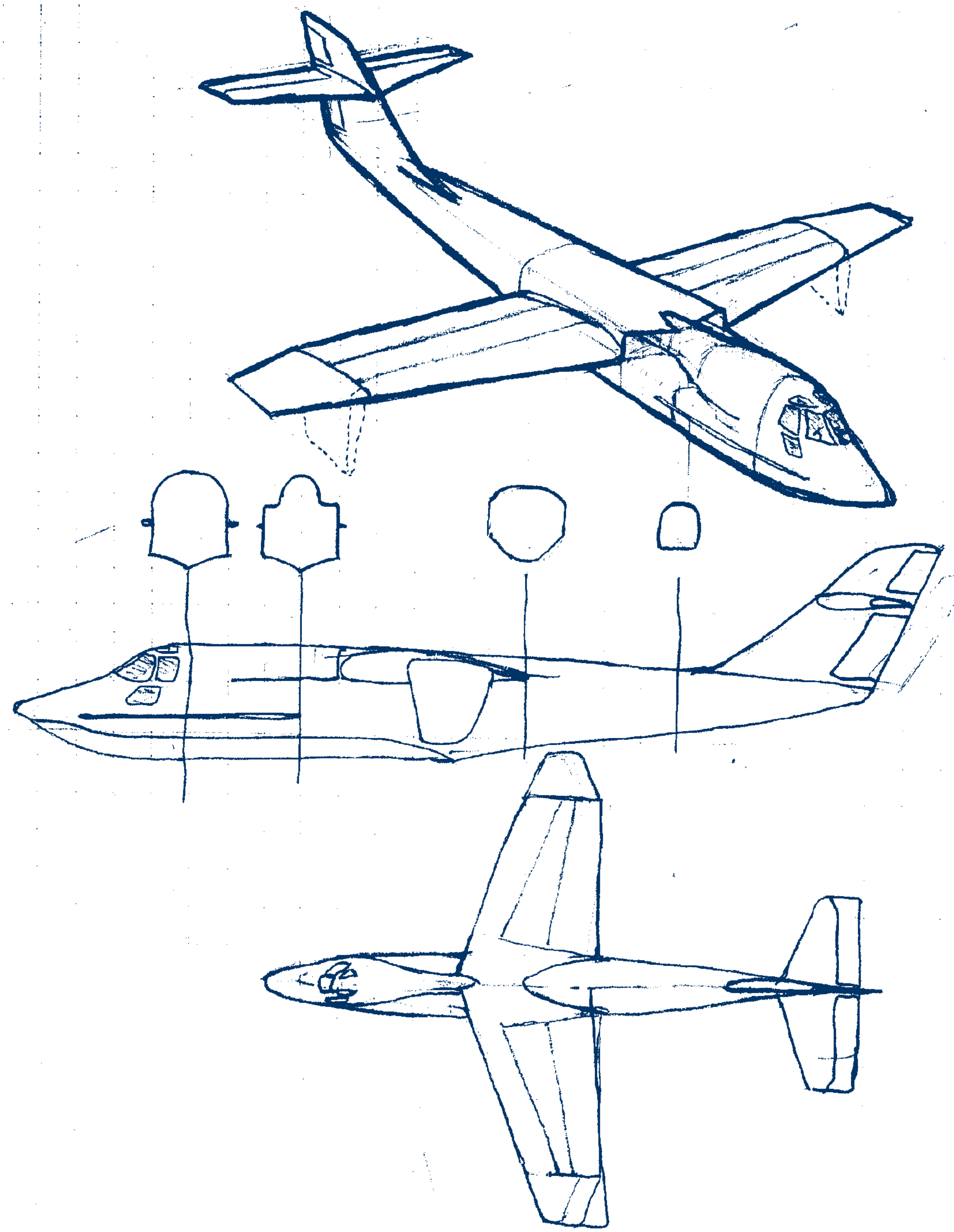


Figure 32 AFreehand Sketch to a Small Scale is Sufficient to Get Initial Ideas on Paper.

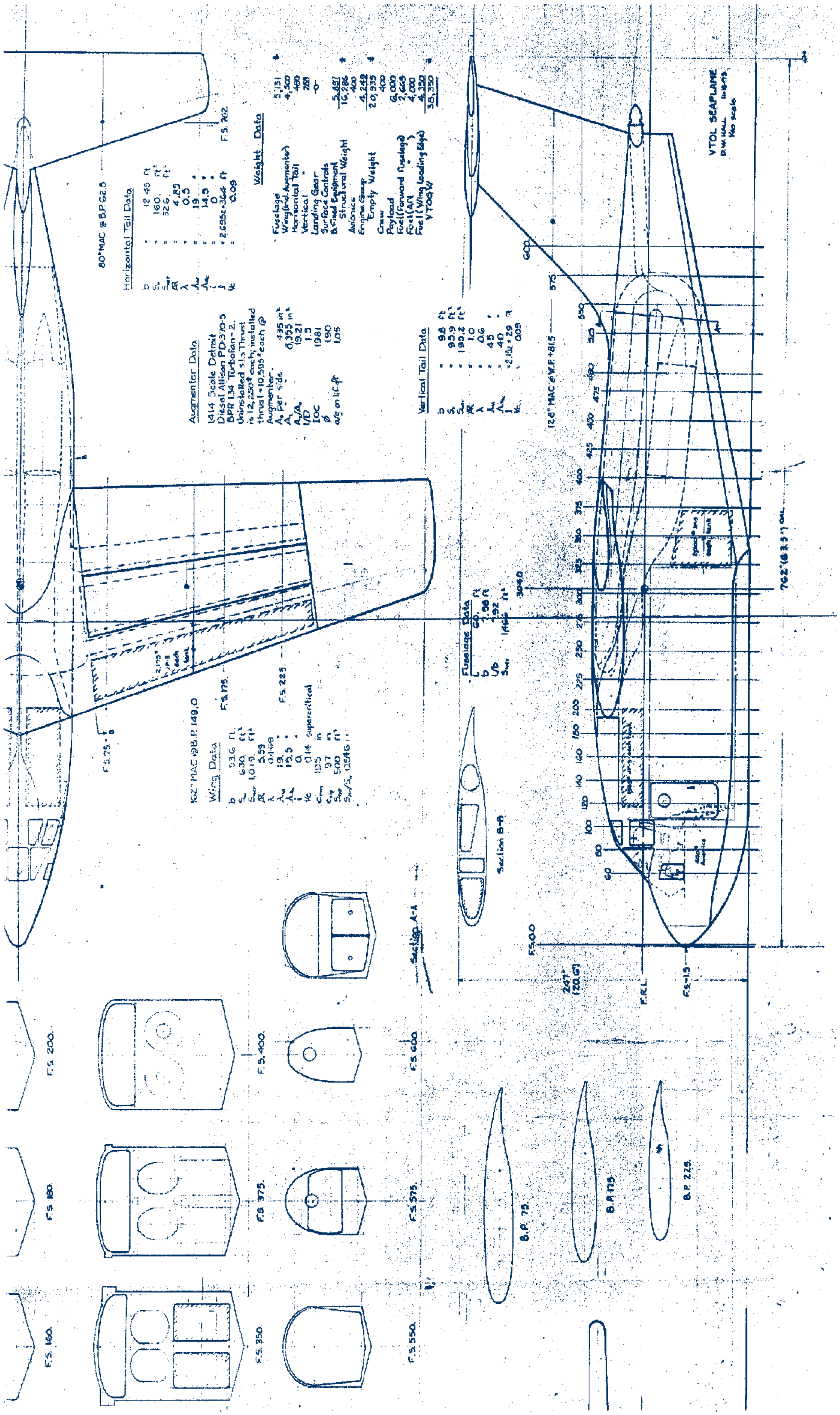


Figure 33. This April, 1973 Configuration Drawing is Typical of First Layouts.

First Layout

The rules of drafting are the lingo and grammar of a visual language engineers use to communicate ideas. As an engineer, it is important to know the rules of drafting. Despite advances in technology, most drafting rules have remained the same for generations, allowing engineers to communicate graphically. Adherence to the rules of drafting ensures that drawings from one project can be understood by anyone with a basic knowledge of drafting. Drafting rules vary between engineering disciplines. The differences may be small, but it is important to be aware of them and to be consistent with industry standards in one's engineering discipline to maintain effectiveness in communicating ideas.

Aeronautical engineers use a drawing format called the three-view. Compiled in this report are guidelines to drawing a three-view of an aircraft. If these guidelines are followed, the draftsperson will produce a drawing of an aircraft that any aeronautical engineer can use to evaluate a design. It is important to remember, though, that a three-view is only one drawing in a set of drawings that define a design. Other drawings include detail drawings and inboards. While this report concentrates on the three-view, the basics of detail drawings and inboards will also be covered.

Finally, the beginning draftsperson must recognize that mastering the graphic language of drafting takes time. Even when it is mastered, creating a set of drawings for an aircraft requires many long sessions of meticulous work. The only way to reach mastery is to practice, and with this report as a starting point, the reader can begin his or her first practice session. Remember, design is iterative; you will draw the same airplane again and again, so try to streamline your work without sacrificing quality.

Setting up

Choosing a design space. When engineers talk about a design space they often refer to the numerical parameters that define the design. The draftsperson must translate these numerical parameters into dimensions and attempt to visualize the design. The computer has become an invaluable tool to the drafter, because it allows him or her to modify drawings relatively easily. Computers also add to the draftsperson's freedom. The drafter is no longer restricted by paper size when he or she starts drawing. Most CAD suites allow the drafter to draw the aircraft full size and then scale it to fit the paper size dictated by the plotter. Since there is no need to lay out the paper with a border and title block before starting to draw, the discussion of title blocks and borders appears in the “Finishing the drawing” section.

A key point when drawing an airplane during the conceptual design phase is that the design space on paper must interact with the design space given by the parameters that define the aircraft. Before starting to draw, an analysis of the design parameters should be completed. Since designing an aircraft is an iterative process, much of the design will change after the first drawing. This should not discourage the draftsperson whose job it is to render an accurate drawing that reflects the status and shortcomings of the design at any given point.

Sketching is a powerful tool for determining the basic layout of an aircraft. Any attempt to draft an aircraft should be preceded by a sketch. Kirschbaum's *Aircraft Design Handbook* includes a treatise on sketching, as do references 2-4. Appendix E shows several sketches of a VTOL seaplane. These sketches were turned into working drawings and formal three-views. The usefulness of CAD suites comes from the ease with which they allow the configuration to be changed. However, drawing the initial layout on a computer will probably take longer than doing so by hand. This is the tradeoff that the draftsperson must accept.

Picking a datum. Datum lines define the origin of the coordinate system in which the aircraft is drawn. There are different conventions for picking datum locations, but only the most common one will be presented here. It will allow the most freedom in changing the configuration of the aircraft.

The datum can be placed at the imaginary ground plane in front of the nose of the aircraft or, perhaps, lower. Not placing the origin within the moldlines of the aircraft fuselage allows the aircraft to grow without having to relabel fuselage stations where section views are taken if the design changes. The "Section Views" section discusses fuselage station labels in more detail. Figure 35 shows correctly set up datum lines and the origin for the side view.

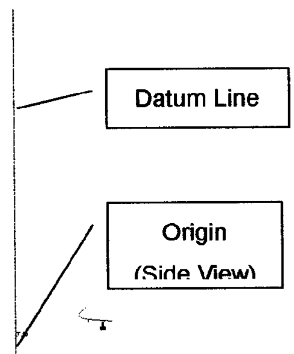


Figure 35. AThis is a Classic Drawing Set-Up with the Datum Located In Front of the Nose of the Aircraft.

Starting to Draw

Basic Layout. After the datum lines are set, the drafter is ready to start drafting. As implied by the name, three views of the aircraft will be shown. This is analogous to the three dimensional multi-view drawings seen in other engineering disciplines. As with all multi-view drawings, the views must project correctly. For most aircraft, the correct projection is orthogonal.

The most significant difference between the technical drawings found in mechanical engineering and an aircraft three-view is the way the views are projected. As with the conventional multi-view, a front view, right side view (which actually shows the left side of the aircraft) and a top view are shown. As shown in Figure 36, the top view is projected from the right side view, not from the front view. The exceptions to this rule of projection are aircraft which have large wingspans relative to their length. The typical projection scheme (top diagram in Figure 2) is often used for detail drawings which are discussed later on. Figure 37 shows the completed basic layout of an aircraft.

To clarify the shape of the aircraft, the outline of the aircraft should be drawn in a heavier line weight than the detail on the fuselage. The line weights provided in Table 6 are a good starting point. More important is the idea of using line weight to distinguish the external boundary of the aircraft. Creating all three views and making sure they project correctly takes meticulous effort, but diligence will lead to an accurate rendition of the aircraft.

Section Views. The three views of an aircraft reveal its basic geometry but they are unable to convey all of the information that defines a complex design. Section views are invaluable to show internal detail, because aircraft tend to be too complex to show such detail with hidden lines. Section views should appear for every location that reveals significant detail, like unusual structure, details of landing gear placement, the angle of the engine mounts, etc.

To create a section view, the draftsperson imagines a cutting plane through the aircraft at the location of interest. The location of the cutting plane is marked by a phantom line, which looks like a centerline with two short dashes instead of one. In mechanical drawings, arrows at the ends of the line marking the section mark the view direction. These arrows can be omitted on aircraft drawings, since the default is to look from nose to tail down the aircraft. The actual section view shows all significant visible edges that project onto the cutting plane. Figure 4 shows a section view of the Whirlwind. Further examples of section views can be found on the drawings in the appendices. Notice that section views are used to show fuselage cross-sections as well as to reveal additional information in detail drawings.

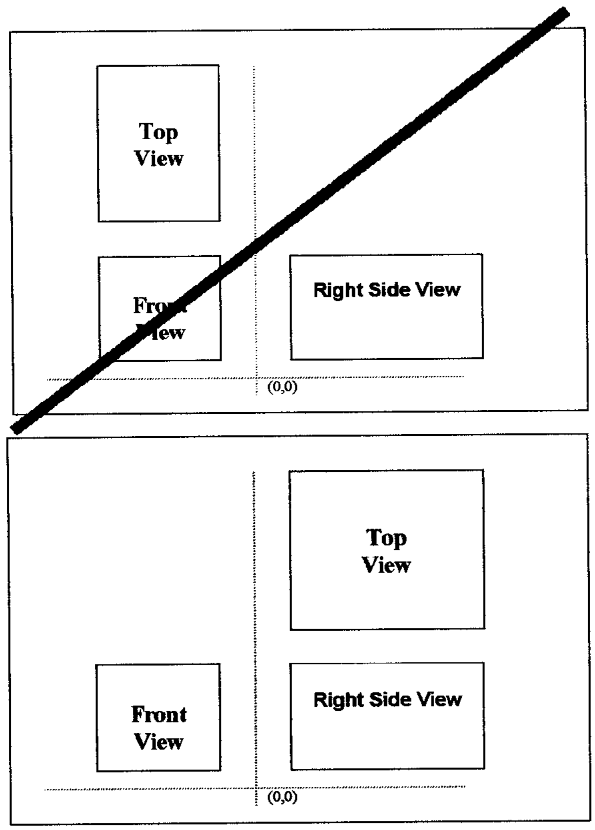


Figure 36. The Top Figure Shows the Standard Layout for a Mechanical Multi-View Drawing. For Aircraft Layouts, the Convention is Slightly Different, as Shown in the Bottom Layout.

The location of the datum becomes particularly significant when defining sections. Each section should be labeled with the distance from the datum in inches. For instance, F.S. 126 is the location of the cutting plane 126 inches aft of the datum. For example, if the side view datum line is placed at the tip of the nose and the nose has to be redesigned to accommodate a radome, either the location of the cross-sections changes or portions of the aircraft have negative stations. Relabeling fuselage stations unnecessarily adds to everyone's workload, not just the drafter's, and negative fuselage station labels invite sign errors in calculations. Both can be avoided by not placing the datum lines within the fuselage.

Table 6. Line weights for different line types on a standard three-view

Line Type	*Line Weight*
External edge	1 pt. or 0.020"
Internal edge	½ pt. or 0.010"
Hidden lines	¼ pt. or 0.005"

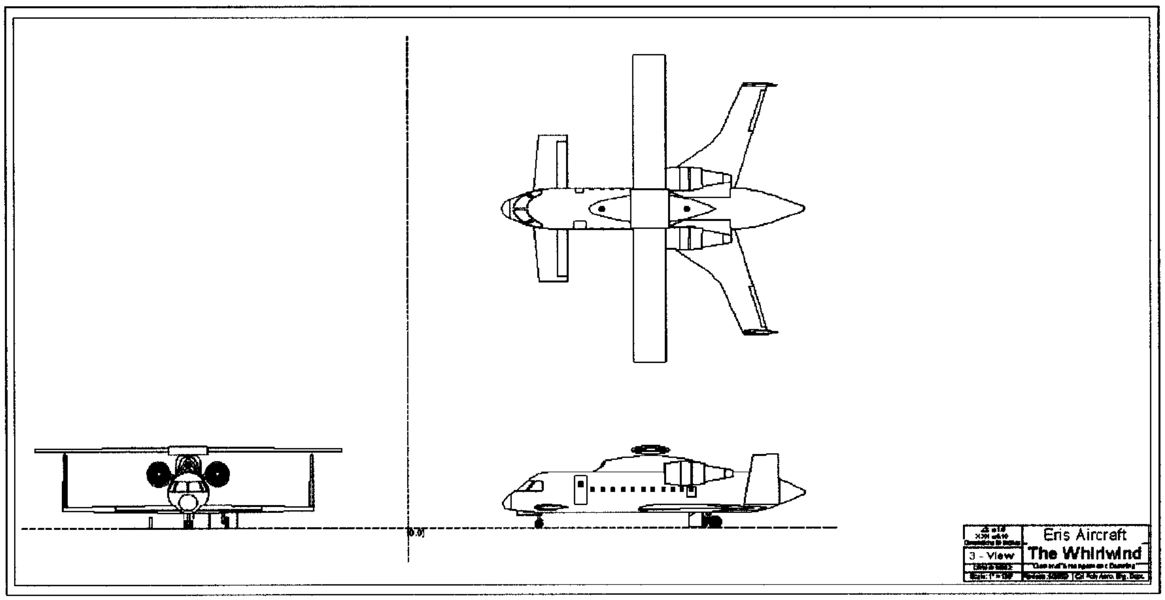


Figure 37. The Basic Layout of the Whirlwind.

Specific section views that should appear on the drawing include the airfoil sections used. If the wing uses a constant airfoil section, only one airfoil needs to be shown. If the wing has a varying section, the sections at significant changes in the wing (like at the intersection of a yehudi, structural break line, or at a sweep angle change) should be shown. Airfoil sections should also show the location of the spars running through the wing. Notice that the final three-view of the Whirlwind includes only the airfoil sections. All other section views were placed on other drawings in the set. Designers frequently show several significant cross-sections besides the airfoil sections on the three-view and the NASA Ames Model 99 HALE UAV in Appendix B is one example.

Inboard Profile. The inboard profile is a special kind of section view. To create an inboard, the fuselage is cut along a vertical plane down the centerline. All significant visible lines that project onto the cutting plane are shown. The view is identical to the side view, except that it reveals the internal arrangement of the aircraft by not showing external skins and near-side structure. A complete inboard drawing should also include a plan view, and any section views that reveal significant detail. For example, the section view included with the Whirlwind inboard in Figure 38 shows compliance with FAR cabin layout requirements. Since the object of the inboard is to show internal detail, external components of the aircraft, like wings and engines, can be omitted from the drawing. To create the plan view, an imaginary cutting plane is created along the line of maximum width parallel to the ground. The view into the airplane is as if the top has been lifted off, allowing the internal layout of the aircraft to be seen.

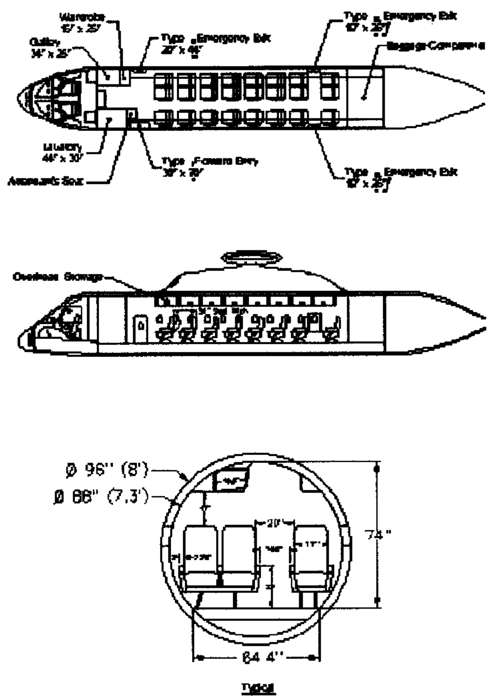


Figure 38. Shown Above are the Inboard Profile (top & middle) and a Section View (bottom) of the Whirlwind. Note that a Typical Cross-Section is Shown, So No Specific Fuselage Location is Labeled.

The inboard drawing should show the placement of oversized cargo in the cargo bay, passenger comfort, and pilot fit. For simple layouts, the inboard can be placed on the three-view; however, it often appears on a dedicated drawing as the design progresses and more is known about the layout of the airplane, particularly if several alternate internal layouts are being offered. Appendix Acontains the complete inboard profile for the Whirlwind.

Detail Drawings. Another tool for showing additional information is a detail drawing. To create a detail drawing, a single component or section is separated from the rest of the aircraft. This component is then drawn on its own in greater detail than is possible on the three-view. Detail drawings are often used to show significant structure, like the wing and wing box design. Figure 39 shows a detail drawing of the Boeing 757-200 wing. Appendix B contains a detail drawing of the NASA Ames Model 99 HALE UAV's wing. Detail drawings usually make up a significant portion of the drawings in a set; however, they are increasingly appropriate as the aircraft configuration progresses beyond the conceptual design phase. It is up to the draftsperson to take the time to make detail drawings of significant aspects of the aircraft.

Scrap Views. Scrap views are specific detail drawings of a single aspect of the aircraft. They appear as auxiliary views on other drawings, and often do not follow particular rules of projection. Scrap views illustrate specific aspects of the aircraft or the operation of onboard mechanical systems. For example, the wing detail drawing of most aircraft is often surrounded by scrap views of actuator placements and flap operation. These scrap views will show the flap stowed in the wing as well as in its extended position. Figure 40 shows scrap views of the Whirlwind landing gear placement.

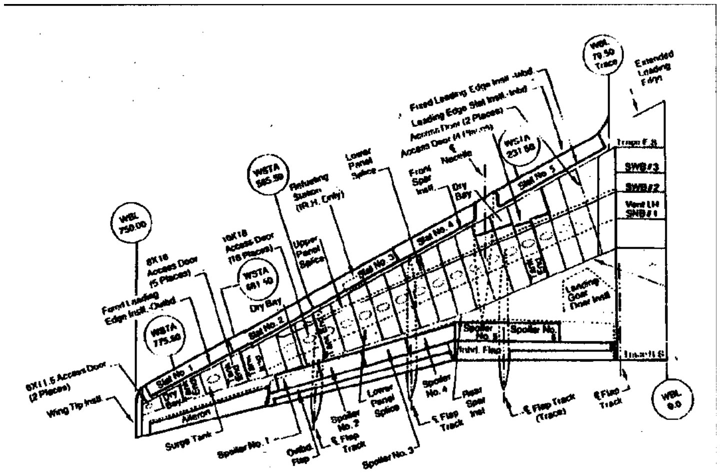


Figure 39. This is a detail drawing of the Boeing 757-200 outboard wing.

Detail . Due to the complexity of an aircraft, it is usually not feasible to use hidden lines to show every detail; however, the limited use of hidden lines to show significant details that do not warrant a separate drawing can be effective. The natural symmetry of most aircraft can be used to keep the drawing from getting too cluttered by only showing hidden detail on one half. For example, Figure 41 shows a hidden landing gear placement in the top view on one side of the aircraft. This maintains the cleanliness of the drawing on the other side and demonstrates that the placement of the gear was considered in the design. Showing hidden details on the three-view demonstrates that that issue of the design has been addressed and a solution has been found. Even if detail drawings exist, it may be worthwhile to show the hidden detail on the three-view, even if it is simplified. Generally, hidden details should be shown using dashed lines. For example, Figure 41 also shows the rotor spar, rotor shaft and control horn as hidden details. This emphasizes that the rotor has been addressed as a particular design issue of the Whirlwind. It is not possible to overemphasize that design work has addressed the salient issues.

Further Detail . Adding detail that is not significant to the design may make the layout look more convincing, or it may clutter it. For instance, adding the bolts on the hub of a tire or the latches on the overhead compartments of a commercial inboard layout are not important details to a conceptual design, but they make the design look more believable. Developing an eye for what is appropriate is a fine point of skilled draftsmanship, and must be traded off with the extra time that is needed to add detail. If extra detail is added, it must be correct and not deter from the clarity of the drawing. Nothing could be worse than having design engineers concentrate on the wrong spacing of the tires and ignore the actual virtues of your design.

To minimize the time it takes to add detail, it is useful to compile a template library of standard equipment found on aircraft that can be scaled and inserted into your design. Landing gear, cockpit layouts and cabin designs, particularly, lend themselves to coming from a template library. For example, the simplest way to check how comfortable passengers in a passenger plane will be is to insert a standard seat arrangement from your library and place at least two rows of seats in the inboard. This can also create a great permanent visual if space efficiency and passenger comfort are major tradeoffs for your design. Figure 42 shows a template made from a figure in Torenbeek's design text which can be used to size cockpits.

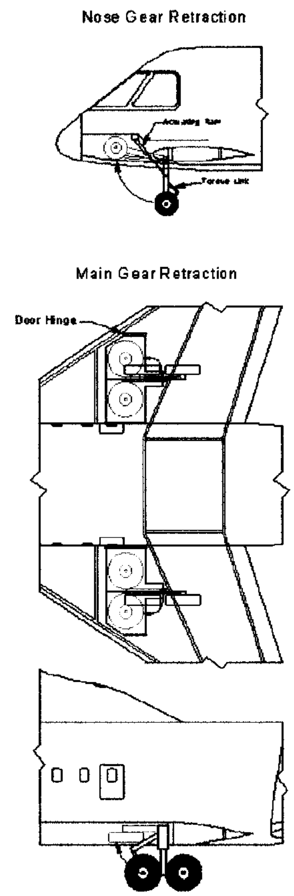


Figure 40. Shown Are Scrap Views of the Landing Gear Placement and Mechanism on the Whirlwind.

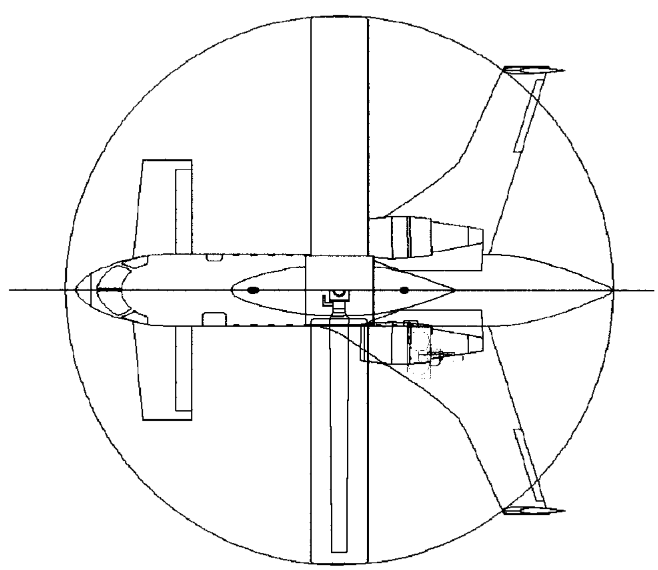


Figure 41. Note the Addition of Detail to the Above Top View. Sturctural Details of the Rotor and the Landing Gear Have Been Added to One Side, Preserving the Cleanliness of the Drawing Without Omitting Important Information.

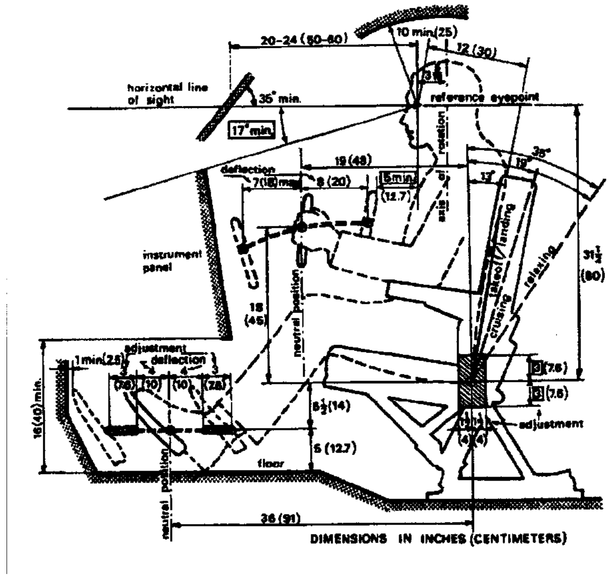


Figure 42. ADrawing from Torenbeek (ref?) Can Be Used as a Drawing Template.

Finishing the drawing

Dimensioning . The three views by themselves convey geometric proportions to present a qualitative visual of the aircraft. Dimensions are required to quantify the size characteristics of the design. For the most part, an aircraft three-view is dimensioned like a mechanical drawing, with one major exception: the top view is not dimensioned except for spanwise locations of mean geometric chords, and these dimensions should only appear on one side of the aircraft. Unusual designs may have dimensions that cannot be easily seen on the side or front views and these can also be shown on the top view. Not dimensioning the top view reserves it as the true measure of how the airplane looks.

Another convention for keeping the drawing from looking cluttered is to show dimensions only on the view that best represents them. In practice, this means an un-dimensioned top view, a front view that shows spanwise dimensions, and a side view that shows length and height dimensions. Following this practice allows each view to emphasize the information shown, rather than presenting a clutter of geometric relationships. Figure 43 shows a properly dimensioned three-view.

Data Tables. Not all the technical information that is significant to a design can be shown as dimensions. Adding data tables allows the designer to include aerodynamic and performance data, mission requirements, and weight and balance information with the drawing of the aircraft. Some designers even add a mission profile to their drawing to show the mission that the aircraft was designed for, and to annotate mission performance.

Another use of data tables is to de-clutter the dimensioned views or display standard dimensions that are obscured in the drawing. Data tables are also a great place to show non-dimensional parameters that define the aircraft like aspect ratio, taper ratio, and tail volume. The foldouts in the appendices show complete three-views with data tables, section views and hidden details as examples of how to create readable drawings.

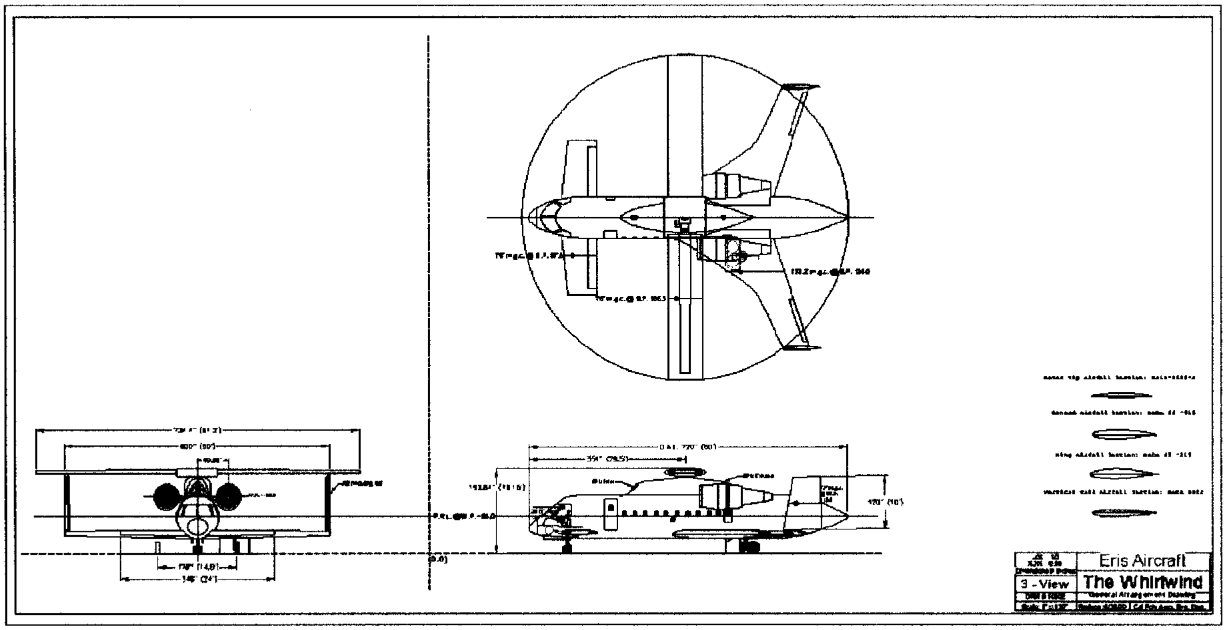


Figure 43. The Properly Dimensioned Layout of the Whirlwind Includes Airfoil Sections.

The most useful data tables to add to a three-view drawing show weight and balance information, aerodynamic data, and performance data. Showing these tables allows anyone to understand the design, both geometrically and in terms of capabilities. Tables 7 and 8 are examples of standard data tables used on the HALE UAV in Appendix B to show aerodynamic and weight and balance information.

Table 7. Aerodynamics Data Table for the NASA/Ames Model 99 HALE UAV.

Lift Coefficient	
Maximum	1.70
Average Cruise	0.81
Drag Coefficient	
Parasite	0.0068
Inviscid due to lift	0.0171
Viscous due to lift	0.0015
Total	0.0254
Lift-to-Drag Ratio at cruise	31.99
Airplane Efficiency Factor	0.9288
Wing Loading at takeoff	41.1 psf
Thrust-to-Weight Ratio at takeoff	0.538

Table 8. Partial Weight and balance Table for the NASA/Ames Model 99 HALE UAV.

Item	Weight	X-Arm	Z-Arm	Moment about X	Moment about Z
Wing	515#	179.2"	-10.0"	92,288.00"#	5,150.00"#
Horizontal	116	297.3	+32.0	34,486.80	+3,715.00
Vertical	103	299.8	+50.2	30,879.40	5,170.60
Fuselage	187	120.0	± 0	22,440.00	± 0
Nacelle	32	238.2	+18.7	7,622.40	+598.40
Landing Gear (up)					
Nose	24	66.0	-11.1	1,584.00	-266.40
Main	48	196.8	-11.6	9,446.40	-556.80

Title Block and BorderLines. Thanks to CAD suites and computers, the draftsperson can wait until he/she has finished the drawing to scale it. The easiest approach is to decide on a paper size, create the appropriate border and title block, and then insert the drawing into that border by adjusting the scale. Keep in mind that a standard scale (1:10, 1:20, 1:40, 1:50, or 1:100) will be much easier to interpret than an arbitrary scale used only to fit the drawing onto the paper. As with everything discussed here, try to balance convention with practicality. The plotter generally dictates the paper size. The maximum height of the drawing is determined by the maximum paper width the plotter can handle.

There are many types of title blocks, and it is really up to the draftsperson or their employer to choose the appropriate one. Most aircraft designers use a title block located in the bottom right corner of the drawing. Similarly open is what information to include in the title block; however, there is information that cannot be omitted, like the draftsperson's name, the scale and units, the drawing title, the company name, the date the drawing was completed, the type of drawing, and the number of the drawing in a set. Table 9 shows a generic title block that includes all of this information. Once the title block has been created, it is advisable to save it as a separate file, allowing the draftsperson to use it as a template in future drawings without having to recreate it. The finished three-view of the "Whirlwind" is a foldout in Appendix A. Note the use of data tables and how excessive white space was taken up with a pictorial of the aircraft.

Table 9. Standard Title Block.

Drawn by		Company or School	Project Logo
Date		Logo	
Checked by		Aircraft Name / Designation	
Date			
Released by		Type of Drawing	
Date		Scale (1" = ?' or ?")	Drawing # (of # in set)

Conclusion

The objective of drafting is to transmit engineering ideas. The draftsperson has three obligations.

- Create a believable image of the aircraft;
- Convey the information that defines a design; and
- Point out the significant issues of a design, and show their solutions.

Fluency in the visual language of technical drawing takes practice and diligence. It is therefore invaluable to the beginning draftsperson to study other drawings and to decide what works and what doesn't. Attached in the appendices are the complete drawings of the Whirlwind made by Robert Dinovo, and drawings of the NASA Ames Model 99 HALE UAV, the Ryan Hi-Star, and a series of sketches and working drawings of a VTOL seaplane by Dave Hall circa 1973. To explicitly show the interaction between drafting and design, an area rule study of a Reno Air Racer is also included. Appendix G is an instruction manual for creating a 3-D model of an aircraft given a 2-D layout. The references given reflect texts about the basic rules of draftsmanship, which can answer many basic technical questions. The *Aircraft Design Handbook* by Kirschbaum and Mason also includes numerous examples of aircraft three-views.

Use these sources and references to build a repertoire of drafting conventions that work to convey your ideas. Though based on a complex system of rules, every drawing also reflects the draftsperson's style, his or her dialect in the visual language. This dialect reflects which conventions work well for that drafter and, thus, allow him or her to create visually appealing and informative drawings at a practical pace. Taking drafting to a level of mastery and finding one's style takes practice and many long drawing sessions. Get started, get feedback from your peers, and take pride in your craft.

Calculations to Include with the First Layout

Geometry

Somewhere on the drawing will be a geometry summary table with measurements for flying surfaces and bodies along with estimates of wetted areas and internal volumes.

Calculations begin with reference areas for all flying surfaces. Exposed areas will be reference areas less any portions of the flying surface covered by bodies or intersections with other flying surfaces. Wetted area is defined as the portion of a flying surface exposed to the air, or approximately the exposed area multiplied by the perimeter-to-chord ratio of the chosen airfoil. A reasonable estimate of this parameter for a symmetrical airfoil is 2.035, so wetted area of a flying surface using this airfoil would be 2.035 times the exposed planform area.

Span, aspect ratio, maximum thickness-to-chord ratio, and taper ratio are as defined in aerodynamics and design texts and are always calculated using reference areas. Sweeps are measured at five stations across the mean geometric chord: leading edge, quarter chord, half chord, three-quarter chord, and trailing edge. Chords are measured parallel to the fuselage centerline (centerline of symmetry) and incidence angle is measured from the fuselage reference line, positive airfoil nose up. Twist is referenced to the root chord and is also positive airfoil nose up. Tail arms are measured from quarter chord to quarter chord and tail volumes are:

GEOMETRY

ITEM	UNITS	WING	HORIZONTAL	VERTICAL
Areas	sq.ft.			
Reference		113.08	18.74	16.76
Exposed		95.05	17.95	15.87
Wetted		198.18	36.53	31.89
Span	ft.	47.56	9.68	5.79
Aspect Ratio	-	20	5	2
Taper Ratio	-	0.3	0.3	0.3
Sweeps	° aft			
Leading Edge		6	30	30
Quarter-Chord		4	25	24
Half-Chord		3	19	18
Three-Quarter-Chord		1	14	11
Trailing Edge		0	8	3
Thickness-to-Chord Ratio	-	0.14	0.09	0.09
Section		OW 70-10-14mod	NACA0009	NACA0009
Chords	ft.			
Root		3.66	2.98	4.45
Exposed Root		3.53	2.88	4.29
Tip		1.10	0.89	1.34
Mean Geometric		2.61	2.12	3.11
Tail Arm	c	-	2.83	2.83
Tail Volume	-	-	0.469	0.023
Incidence Angle	°	3	0	0
Wash-in (positive)	°	-3	0	0

ITEM	LENGTH	WIDTH	HEIGHT	WETTED AREA	INTERNAL VOLUME
Fuselage	28.91	2.33	2.33	235. sq.ft.	142. cu.ft.
Nacelle	9.87	2.50	2.00	-0-	-0-
Payload Pod (each)	6.88	1.50	1.50	25..	8.

Figure 44. ATypical Geometry Table Looks Like This.

$$V_{\text{horizontal}} = \frac{S_{\text{horizontal}}\ell_{\text{horizontal}}}{S_{\text{reference}}c_{\text{mean}}} \tag{133}$$

$$V_{\text{vertical}} = \frac{S_{\text{vertical}}\ell_{\text{vertical}}}{S_{\text{reference}}b} \tag{134}$$

For bodies, measure their overall length, maximum height, and maximum width. Estimates of wetted area and internal volume have to come from cross-sections. Figure 45 presents a typical set of cross-sections showing entries for wetted perimeters and internal areas for the sections. Table 10 summarizes these and measurements from other cross-sections to calculate both wetted area and internal volume for the entire fuselage.

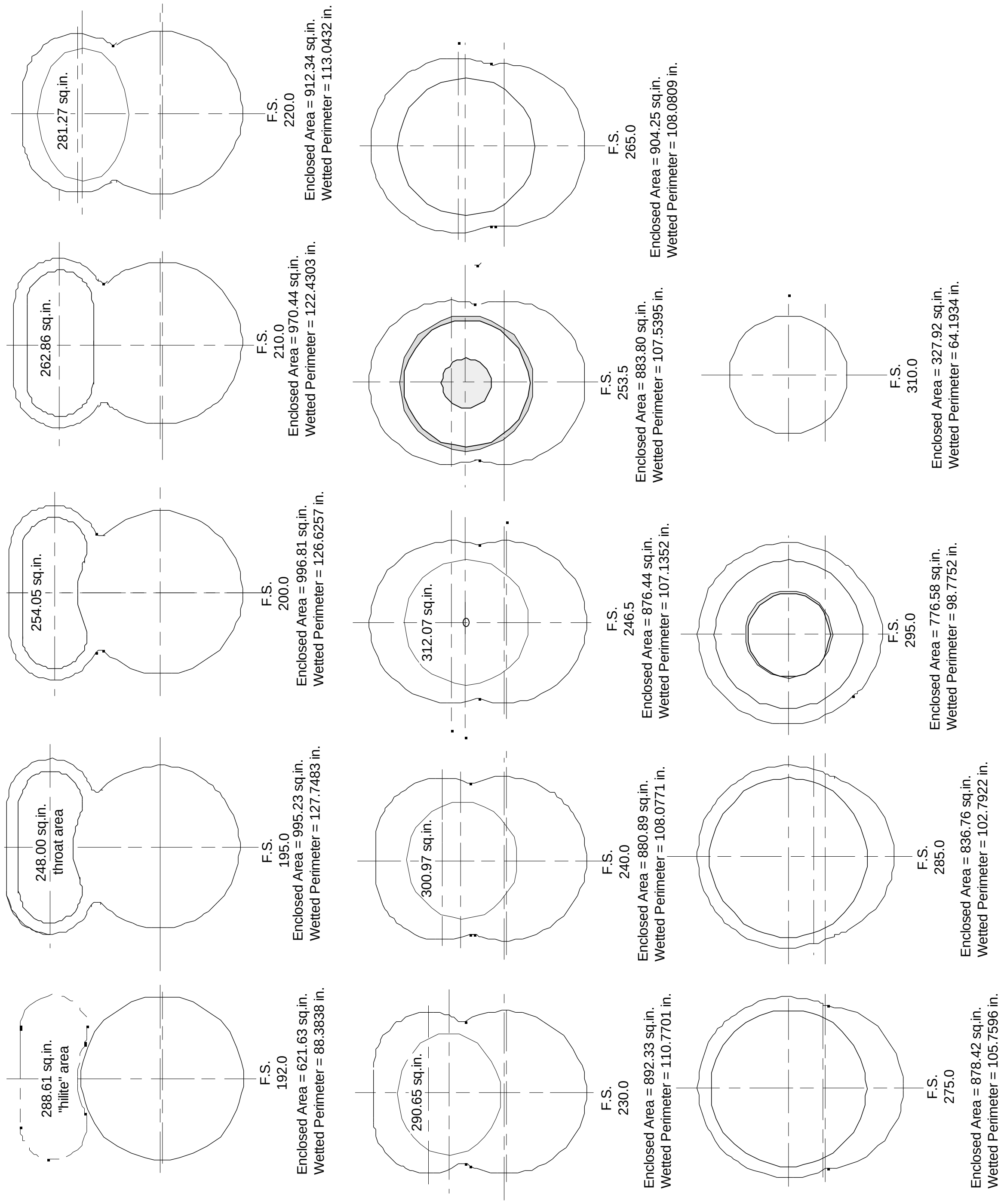


Figure 45. These are Typical Cross-Sections in the Engine Inlet Region of a Subsonic HALE Configuration.

Table 10. This Excel Spreadsheet Excerpt Shows the Calculation Sequence for Both Wetted Area and Internal Volume for the Same HALE Aircraft.

Fuselage Station	Perimeter	Area	Distance	Average Perimeter	Average Area	Wetted Area	Internal Volume	Cumulative Wetted Area	Cumulative Volume
	inches	sq.in.	inches	inches	sq.in.	sq.in.	cu.in.	sq.in.	cu.in.
		NAS4/APS Model 99							
Fuselage:									
-37.0	0.00	0.00						0	0
-20.0	48.17	184.66	17.0	24.1	92.3	409	1,570	409	1,570
0.0	67.86	366.44	20.0	58.0	275.6	1,160	5,511	1,570	7,081
40.0	87.65	611.36	40.0	77.8	488.9	3,110	19,556	4,680	26,637
45.0	95.10	678.09	5.0	91.4	644.7	457	3,224	5,137	29,860
47.3	95.32	680.48	2.3	95.2	679.3	219	1,562	5,356	31,423
50.0	95.86	688.62	2.7	95.6	684.6	258	1,848	5,614	33,271
55.0	96.48	702.15	5.0	96.2	695.4	481	3,477	6,095	36,748
60.0	96.04	702.49	5.0	96.3	702.3	481	3,512	6,576	40,259
65.0	95.20	701.86	5.0	95.6	702.2	478	3,511	7,054	43,770
72.9	93.62	697.46	7.9	94.4	699.7	746	5,527	7,800	49,298
90.0	94.51	710.72	17.1	94.1	704.1	1,609	12,040	9,408	61,338
172.0	124.47	708.03	82.0	109.5	709.4	8,978	58,169	18,387	119,506
175.0	113.13	714.36	3.0	118.8	711.2	356	2,134	18,743	121,640
179.0	105.82	719.18	4.0	109.5	716.8	438	2,867	19,181	124,507
185.0	100.24	721.88	6.0	103.0	720.5	618	4,323	19,799	128,830
192.0	97.11	722.06	7.0	98.7	722.0	691	5,054	20,490	133,884
196.0	133.37	1,093.01	4.0	115.2	907.5	461	3,630	20,951	137,514
200.0	133.45	1,093.35	4.0	133.4	1,093.2	534	4,373	21,484	141,887
210.0	129.94	1,066.11	10.0	131.7	1,079.7	1,317	10,797	22,801	152,684
220.0	120.88	1,020.43	10.0	125.4	1,043.3	1,254	10,433	24,056	163,117
225.0	119.27	1,003.58	5.0	120.1	1,012.0	600	5,060	24,656	168,177
230.0	117.03	990.61	5.0	118.2	997.1	591	4,985	25,247	173,162
240.0	114.86	999.54	10.0	115.9	995.1	1,159	9,951	26,406	183,113
246.7	114.58	996.56	6.7	114.7	998.1	769	6,687	27,175	189,800
253.5	113.97	987.17	6.8	114.3	991.9	777	6,745	27,952	196,545
264.0	114.83	1,016.26	10.5	114.4	1,001.7	1,201	10,518	29,153	207,063
275.0	111.22	965.06	11.0	113.0	990.7	1,243	10,897	30,396	217,960
285.0	109.74	942.64	10.0	110.5	953.9	1,105	9,539	31,501	227,498
295.0	103.05	841.03	10.0	106.4	891.8	1,064	8,918	32,565	236,417
310.0	64.19	327.93	15.0	83.6	584.5	1,254	8,767	33,819	245,184
	Total Wetted Area in sq.ft.							235	
	Total Internal Volume in cu.ft.								142

Performance Summary Tables

Next, include a short table summarizing basic aerodynamic coefficients and key performance data. Figure 46 shows the aerodynamics table from the Model 99 HALE example used above.

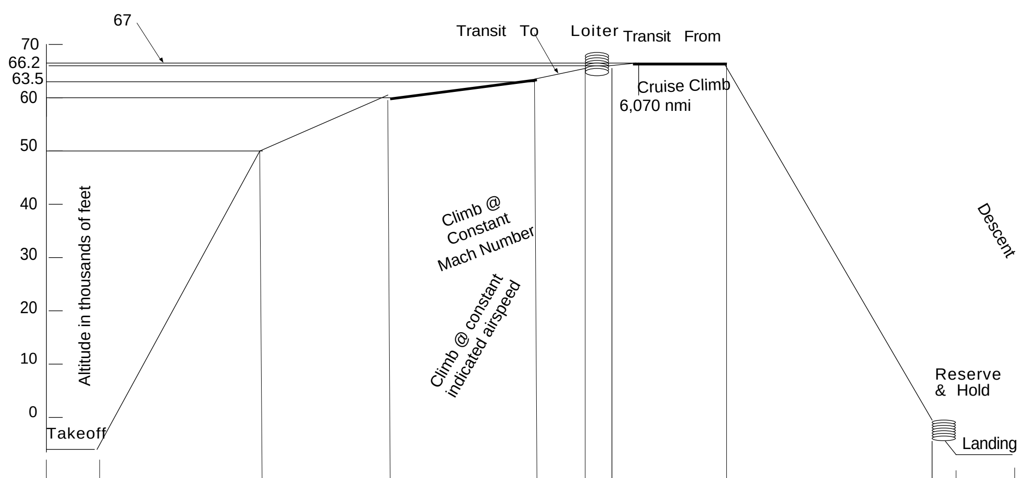
AERODYNAMICS

Lift Coefficient	
Maximum	1.70
Average Cruise	0.81
Drag Coefficient at cruise	
Parasite	0.0068
Inviscid due to lift	0.0171
Viscous due to lift	0.0015
Total	0.0254
Lift-to-Drag Ratio at cruise	31.99
Airplane Efficiency Factor	0.9288
Wing Loading at takeoff	41.1 psf
Thrust-to-Weight Ratio at takeoff	0.538

Figure 46. ABasic Aerodynamics Table Might Also Include Range and/or Endurance Summaries.

Updated drag coefficients can come from the methods in the next section or from DATCOM. Post the wing and airplane efficiency factor calculations in the next section here, too, along with revised estimates of takeoff wing loading and thrust-to-weight ratio.

If convenient and space exists on the drawing, consider including a mission profile with specific performance data shown below it each segment. Figure 47 presents the mission profile for the Model 99 turbofan HALE configuration used here and in Case Study #2.



RPV with 500# payload & fan engine:

Cumulative Distance	n.mi.	0	568 ft	60	158	3,004	3,004	5,393	8,196	8,296	8,397	284 ft
True Airspeed	knots	0	135	345	344	344	344	344	344	135	135	0
Elapsed Time	hours	0	30 sec	0.248	0.533	8.807	8.807	15.753	23.901	24.380	25.130	3 sec
Weight	pounds	4,647	4,631	4,509	4,475	3,793	3,793	3,332	2,843	2,789	2,753	2,747

Figure 47. An Annotated Mission Profile on the Drawing Assures Pertinent Configuration Performance Data Stay With the Layout.

Weight and Balance

Given a first layout and estimates for major weight constituents such as fuel and disposable payload, it is possible to check whether the aircraft center-of-gravity stays within permissible relation to the aircraft center-of-lift. Use Class II weight methods form Roskam, Torenbeek, Nicolai, Raymer, K.D.Wood, or another source to estimate component weights. The Class II methods present component weights in terms of major geometric and structural parameters such as aspect ratio, thickness-to-chord ratio, sweep, taper ratio, positioning and kinds of cutouts, and either planform area or span. At this point in the design cycle, you will want to estimate

- wing weight;
- horizontal and vertical tail weight (or empennage weight);
- fuselage weight;
- nacelle weight;
- landing gear weight;

These sum to structural weight. Then estimate

- surface controls weight;
- fixed equipment weight;
- installed power train weight;

With structural weight, these sum to empty weight. Then estimate

- disposable payload weight;
- crew and furnishings weight; and
- fuel weight.

With empty weight, these sum to takeoff gross weight. Another weight of interest is zero fuel weight, which is just that (takeoff gross weight less fuel weight). Minimum landing weight will be empty weight plus non-disposable payload items plus required fuel reserves.

Then construct a weight and balance table by locating component centers-of-gravity on the layout and measuring their x,y, and z locations (in terms of fuselage station, butt line, and water line). Sum moments, divide by weights and calculate nominal centers-of-gravity for takeoff gross weight and empty weight.

To estimate centers-of-gravity for flying surfaces, find their mean chord lines and assume the center-of-gravity in the x-direction will be around the center of the torque box or just forward of the center. For bodies such as fuselage and nacelles, assume the center-of-gravity will be at about 40% of their total length. Obviously, the lateral center-of-gravity location should be on the centerline of symmetry of the aircraft (if there is one), and the vertical location will be approximately halfway between the fuselage centerline and the wing centerline. Table 11 is reproduced from ref SAWE and provides guidance in locating component centers-of-gravity.

Table 12 presents the weight and balance table for the Model 99 HALE turbofan of Case Study 2.

Table 11. The Society of Allied Weight Engineers (SAWE) Provides These Guidelines for Component Center-of-Gravity Locations.

AIRCRAFT TYPE	COMPONENT	LONGITUDINALC.G.
commercial airliners	wing	35% MAC for 35°sweep
transports		33-48% chord at 40% half-
bombers		span for zero sweep
	horizontal tail	25-35% MAC
	vertical tail	25-55% MAC
	fuselage	40-55% fuselage length
	nacelle	40% length from nacelle
nose		
	engine section	with propulsion c.g.
	air induction	40% duct length from inlet
	landing gear	5% body length forward of aircraft c.g.
fighter	wing	45% MAC for 35° sweep
attack		60% MAC for delta planform
	tail surfaces	40% MAC
	fuselage	40-45% fuselage length
	nacelle	40% length from nacelle
		nose
	engine section	with propulsion c.g.
	air induction	40% duct length from inlet
	landing gear	5% body length forward of aircraft c.g.
general data	propulsion excluding fuel system	at manufacturer's c.g.
	fuel system	50-60% body length
	systems excluding flight controls	with fuselage c.g.
	flight controls	55-65% body length
	fuel	per layout
	payload	per layout

Table 12. This is a Typical Weight and Balance Table for a Multi-View Drawing.

Component	Weight	Fuselage Station	Butt Line	Water Line	x-Moment	y-Moment	z-Moment
	pounds	inches	inches	inches	inch-pounds	inch-pounds	inch-pounds
Wing	W	x	y	z	W*x	W*y	W*z
Horizontal							
Vertical							
Fuselage							
Nacelle							
Landing Gear							
Nose							
Mains							
Structure Weight							
Surface Controls							
Ailerons							
Flaps							
Elevators							
Rudders							
Fixed Equipment							
Propulsion Group							
Empty Weight							
Non-Disposable Payload							
Disposable Payload							
Zero-Fuel Weight							
Fuel							
TakeOff Gross Weight							

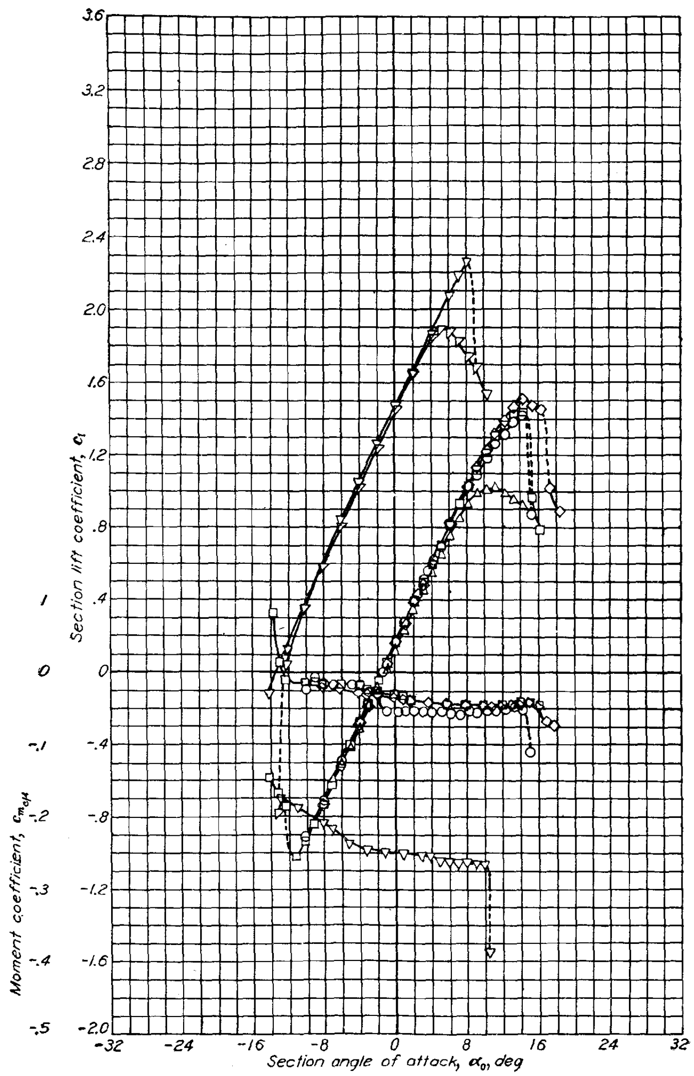
Table 13. Excel Can Be Used to Create a Weight and Balance Table.

	A B C D E	F	G	H	I	J
1	**Model 99**	**HALE**				
2		**15-Dec-92**		**D.W.Hall**		
3	**ITEM**	**WEIGHT**	**X-ARM**	**Z-ARM**	**MOMENT**	**MOMENT**
4		**pounds**	**inches**	**inches**	**ABOUT X**	**ABOUT Z**
5					**inch-pounds**	**inch-pounds**
6	Wing					
7	inboard	170	173.83	-11.00	29,551.10	-1,870.00
8	outboard	340	178.67	-10.63	60,747.80	-3,614.20
9	tip fairing	5	187.77	-9.50	938.85	-47.50
10	Horizontal					
11	structure	113	304.77	22.67	34,439.01	2,561.71
12	tip fairing	3	291.90	25.87	875.70	77.61
13	Vertical					
14	structure	100	294.60	48.67	29,460.00	4,867.00
15	tip fairing	1	316.83	88.70	316.83	88.70
16	dorsal fairing	2	255.87	24.47	511.74	48.94
17	Fuselage	187	120.00	0.00	22,440.00	0.00
18	Nacelle	32	230.00	15.00	7,360.00	480.00
19	Pods	50	181.50	-11.50	9,075.00	-575.00
20	Landing Gear					
21	Nose	24	66.70	-11.23	1,600.80	-269.52
22	Mains	48	197.00	-13.40	9,456.00	-643.20
23	Structure	**1,075**	192.35	1.03	206,772.83	1,104.54
24	Surface Controls					
25	Wing					
26	'Inboard flap	20	189.40	-10.90	3,788.00	-218.00
27	'Outboard					
28	flap #1	10	189.93	-10.83	1,899.30	-108.30
29	flap #2	10	190.50	-10.47	1,905.00	-104.70
30	aileron #1	10	191.73	-9.77	1,917.30	-97.70
31	aileron #2	10	187.77	-9.53	1,877.70	-95.30
32	Horizontal (elevators)	30	305.33	24.97	9,159.90	749.10
33	Vertical (rudder)	15	315.90	56.90	4,738.50	853.50
34	Fixed Equipment	342	72.50	-8.00	24,795.00	-2,736.00
35	Propulsion Group	668	271.00	6.57	181,028.00	4,388.76
36	Empty	**2,190**	**199.95**	**1.71**	437,881.53	3,735.90
37	Pilot & Furnishings	0	0.00	0.00	0.00	0.00
38	Payload					
39	Perseus A Nose	110	10.75	0.00	1,182.50	0.00
40	Aft Nose	90	47.30	4.27	4,257.00	384.30
41	Center Fuselage	200	80.75	6.33	16,150.00	1,266.00
42	Pods	100	154.00	-11.50	15,400.00	-1,150.00
43	Zero Fuel Weight	**2,690**	176.53	1.57	474,871.03	4,236.20
44	Fuel					
45	Wing					
46	Inboard	222	172.53	-11.43	38,301.66	-2,537.46
47	Outboard	453	180.00	-10.90	81,540.00	-4,937.70
48	Fuselage					
49	Center	482	171.40	2.53	82,614.80	1,219.46
50	Forward	800	128.50	0.23	102,800.00	184.00
51	Takeoff Gross Weight	**4,647**	**167.88**	**-0.39**	780,127.49	-1,835.50
52	Quarter MGC is at		170.87			
53	Main Gear Ground Contact Point is at		181.50			

CHAPTER 5
AERODYNAMICS
Choose an Airfoil

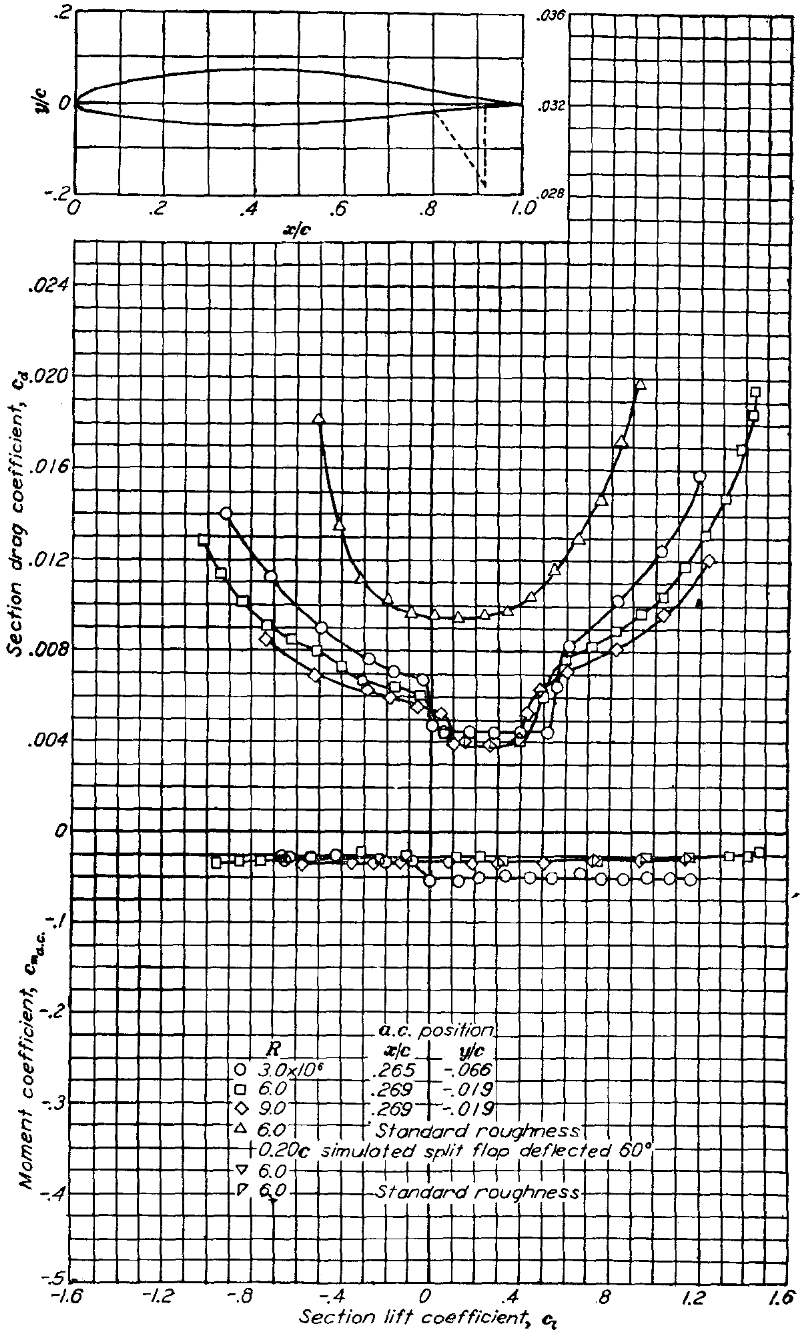
Given mission requirements for an airplane and some understanding of its size and loadings, consult airfoil books or codes and define a wing airfoil to use on the layout. An obvious source is Abbott & von Doenhoff but there are many others including specialty airfoils for specific applications. Figures 6 present data for the NACA 65_1-212 airfoil. Note the well-behaved lift-curve and the large pitching moment curve over a Reynolds Number range similar to that expected during the demanding mission segments of the AIAA mission. Section maximum lift coefficient appears to be around 1.0 with standard roughness and the drag bucket extends far enough into the lift coefficient regime of the AIAA mission estimate that some care to finish may improve performance.

F48,T14 , E135



NACA 65_1-212 Wing Section
$a = 0.6$

Figure 22a. The NACA65_1-212 Airfoil is an Initial Candidate.



NACA 65_1-212 Wing Section
$a = 0.6$ (*Continued*)

Figure 22b. The NACA65_1-212 Airfoil is an Initial Candidate.

Given wing area, aspect ratio, taper ratio and a guess at thickness-to-chord ratio, we can estimate three-dimensional wing performance in and out of ground effect with the following equations from K.D.Wood. Out of ground effect:

$$a_{\text{incompressible}} = \frac{\pi^2}{90}\left[\frac{AR}{2+\sqrt{\dfrac{AR^2}{\cos^2\Lambda}+4}}\right] \tag{112}$$

$$\frac{a}{a_{\text{incompressible}}} = \frac{2+\sqrt{4+\dfrac{AR^2}{\cos^2\Lambda}}}{2+\sqrt{4+AR^2\dfrac{1-M^2\cos^2\Lambda}{\cos^2\Lambda}}} \tag{113}$$

$$C_{D_{\text{wave}}} = \frac{C_L^2}{4}\sqrt{M_\infty^2-1}\left[\frac{AR}{AR-0.5\sqrt{M_\infty^2-1}}\right]+\frac{2\left(\frac{8}{3}\right)\left(\frac{t}{c}\right)^2}{\sqrt{M_\infty^2-1}} \tag{114}$$

$$C_L = \frac{4\alpha}{\sqrt{M_\infty^2-1}}\left[1-\frac{\sqrt{M_\infty^2-1}}{2AR}\right] \tag{115}$$

$$C_{M_{c/2}} = \frac{\alpha}{3AR\left(M_\infty^2-1\right)} \tag{116}$$

With ground effect corrections from K.D.Wood we can now re-estimate takeoff and landing performance.

$$a_{\text{wing}} = \frac{dC_L}{d\alpha}_{\text{wing}} \tag{117}$$

$$\alpha_{\text{wing}_{\text{ground effect}}} = \frac{C_{L_{\max}}}{a_{\text{wing}}k} \tag{118}$$

$$a_{\text{wing}_{\text{ground effect}}} = a_{\text{wing}}k_{\text{wing}} \tag{119}$$

$$a_{\text{horizontal}_{\text{ground effect}}} = a_{\text{horizontal}}k_{\text{horizontal}} \tag{120}$$

Use these data and other estimates to begin a layout.

Airfoil and Wing Lift

On a symmetrical airfoil at zero angle-of-attack (α), both the upper and lower surface velocities are equal at every point; otherwise, lift would be generated in one direction or the other. At non-zero angles-of-attack the velocity at any point on one side of the airfoil is different from the velocity at its corresponding point on the other side. To state this mathematically:

$$\text{At } \alpha > 0,\ V_{\text{upper}} > V_{\text{lower}} \tag{121}$$

If the subscript ∞ represents freestream conditions, then stations 1 and 2 are along the wing. Continuity says that the total pressures at these three locations are equal, or

$$p_1 + \frac{\rho_1}{2}V_1^2 = p_2 + \frac{\rho_2}{2}V_2^2 = p_\infty \tag{122}$$

If the flow is incompressible; that is, less than approximately 300 feet per second (fps), then

$$\rho_\infty = \rho_{\text{upper}} = \rho_{\text{lower}} \tag{123}$$

and

$$p_1 + \frac{\rho}{2}V_1^2 = p_2 + \frac{\rho}{2}V_2^2 = p_\infty \tag{124}$$

$$\frac{p_\infty}{p_1} = \frac{\rho}{2}V_1^2 \quad \text{and} \quad \frac{p_\infty}{p_2} = \frac{\rho}{2}V_2^2 \tag{125}$$

$$V_1 = \sqrt{\frac{2}{\rho}\frac{p_\infty}{p_1}} \quad \text{and} \quad V_2 = \sqrt{\frac{2}{\rho}\frac{p_\infty}{p_2}} \tag{126}$$

Figure 3.11 translates to Figure 3.12 which is the pressure distribution over an infinitesimally small spanwise slice of wingspan, db. The net normal force on the wing, N, has a net airfoil force equivalent, n, such that

$$dn = p\frac{dx}{ds} = pdx \tag{127}$$

where the pressure is the net pressure, or

$$p = p_{\text{upper}} - p_{\text{lower}} \tag{128}$$

So the net force can be expressed as a pressure integral over the chord length (c), or

$$n = \int_0^c p_{\text{lower}} - p_{\text{upper}} dx \tag{129}$$

where x increases in the chordwise direction. If the airfoil is at an angle-of-attack, α, then the section lift, ℓ, and drag, d, are

$$\ell = n\cos\alpha = \frac{\rho}{2}V_\infty^2 c\,\text{d}bC_\ell \quad \text{and} \quad d = n\sin\alpha = \frac{\rho}{2}V_\infty^2 c\,\text{d}bC_d \tag{130}$$

Wing lift and drag are, then,

$$L_{\text{wing}} = \int_{-\frac{b}{2}}^{\frac{b}{2}} \ell\,\text{d}b = \frac{\rho}{2}V_\infty^2 \int_{-\frac{b}{2}}^{\frac{b}{2}} cC_\ell db = \frac{\rho}{2}V_\infty^2 S_{\text{reference}}C_L \quad \text{and} \quad D_{\text{wing}} = \int_{-\frac{b}{2}}^{\frac{b}{2}} d\text{d}b = \frac{\rho}{2}V_\infty^2 \int_{-\frac{b}{2}}^{\frac{b}{2}} cC_d \text{d}b \tag{131}$$

How do we arrive at accurate estimates of finite wing lift and drag? Wind tunnel test measurements provide pressure distributions at known flow speeds and angles-of-attack for a given airfoil section. A large portion of the lift curve will be linear and the slope between any two points will yield the airfoil section lift curve slope.

$$C_{\ell_\alpha} = \frac{dC_\ell}{d\alpha} = \frac{C_{\ell_2} - C_{\ell_1}}{\alpha_2 - \alpha_1} \tag{132}$$

For a finite wing of

$$AR = \frac{b^2}{S_{\text{reference}}} \tag{133}$$

the lift and drag are both functions of the angular momentum change of air aft of the airfoil. This is measured by the downwash angle, ε, which is proportional to wing aspect ratio and lift coefficient or, for an elliptical lift distribution,

$$\varepsilon = \frac{C_L}{\pi AR} \tag{134}$$

This equation says that downwash is strongest with low aspect ratio wings producing high lift coefficients and approaches zero as aspect ratio approaches infinity. For non-elliptical lift distributions, which most wings have, the wing efficiency factor is defined in one reference work as

$$e_{\text{wing}} = 4.61\left(1 - 0.045 AR^{0.68}\right)\left(\cos \Lambda_{\text{leading edge}}\right)^{0.15} - 3.1 \tag{135}$$

This equation says that wing efficiency factor drops off precipitously with increasing aspect ratio and that wings of aspect ratios higher than about 18 will have zero or negative efficiency. That being the case,the Lockheed U-2 and ER-2, the Grob Strato 2C, the Ryan Global Hawk, and every human-powered or solar powered airplane ever built wouldn't work.

Another reference defines wing efficiency factor as

$$e_{\text{wing}} = \frac{2}{2 - AR + \sqrt{4 + AR^2\left(1 + \tan^2 \Lambda_{\text{maximum thickness line}}\right)}} \tag{136}$$

which says that wing efficiency increases with increasing aspect ratio, that a wing of aspect ratio = 1 has a 0.618 efficiency, and that there's not much point in building wings of aspect ratio higher than about 10 to 15, which is consistent with experience. Airplanes pushing the edge of the efficiency envelope would have higher aspect ratios, but there's little increase above about aspect ratio 35, which is also consistent with experience (Condor and the Scaled Composites Voyager, and the LMSC solar powered high altitude powered platform were all about aspect ratio 34).

Table 3.6 in Roskam Volume I (p. 127) provides increments to parasite drag coefficient to account for non-cruise flap and landing gear configurations:

Configuration	ΔC_{D_0}	e
clean	0	0.80-0.85
takeoff flaps	0.0100-0.0200	0.75-0.80
landing flaps	0.0550-0.0750	0.70-0.75
landing gear down	0.0150-0.0250	no effect

$$\left(C_{D_T}\right)_{\text{configuration}} = \left(C_{D_0}\right)_{\text{configuration}} + \frac{C_L^2}{\pi e_{\text{configuration}} AR} \tag{137}$$

$$\left(C_{D_T}\right)_{\text{configuration}} = \left[\left(C_{D_0}\right)_{\text{clean}} + \left(\Delta C_{D_0}\right)_{\text{landing gear}} + \left(\Delta C_{D_0}\right)_{\text{flaps}}\right] + \frac{C_L^2}{\pi e_{(\text{configuration, landing gear,flaps})} AR} \tag{138}$$

After defining airfoil and wing characteristics, the mission analysis starts with analytical description of each mission segment beginning with takeoff and continuing through the end of the transit out segment (called Transit To here). The analysis then jumps to the landing segment and works backwards through the transit back segment (called Transit From here) so that the loiter segment is the last one analyzed. In this way, mission radius of action requirements are satisfied, both beginning and ending conditions for the loiter segment are known and loiter time becomes a residual. This method begins with a known point design.

Initial Wing Definition

Sizing calculations so far have defined transit and loiter segments in enough detail to use them to estimate wing reference area. Given aspect ratio, wingspan will be

$$b = \sqrt{AR * S_{\text{reference}}} \tag{139}$$

Roughly define wing shape in terms of reference area, wingspan, taper ratio, and thickness-to-chord ratio, then locate the wing mean geometric chord. Aerodynamic theory says that a taper ratio between 0.4 and 0.6 creates an approximately elliptical lift distribution. Taper ratio is defined as

$$\lambda = \frac{c_{\text{tip}}}{c_{\text{root}}} \tag{140}$$

Lay out the wing in the planview and find the mean geometric chord as the area centroid of a trapezoid. Measure root and tip chords from the layout and use to calculate an exact value of mean geometric chord.

$$c_{\text{mean}} = \frac{2c_{\text{root}}}{3}\left(\frac{1+\lambda+\lambda^2}{1+\lambda}\right) \tag{141}$$

Next, choose an airfoil taking into account requirements already estimated for:

- Maximum lift for takeoff with partial flaps
- Maximum lift for landing with full flaps
- Cruise lift at transonic Mach Numbers

Thickness-to-chord ratio will affect transonic drag coefficient of flying surfaces—the thinner the better for conventional airfoil sections and somewhat thicker for supercritical airfoil sections. Tradeoff with structural need for torque box depth. From Torenbeek, pp. 246-249:

$$t/c \le 0.3\left\{\frac{2}{\gamma M_{\text{critical}}^2}\left[\left(1-\left(\frac{2+(\gamma-1)}{\gamma+1}\right)M_{\text{critical}}^2\right)^{\frac{\gamma}{\gamma-1}}\right]\right\}^{\frac{2}{3}} \tag{142}$$

Adding wing sweep effects:

$$(t/c)_{\text{average}} \le \frac{0.3}{M_{\text{drag divergence}}}\left\{\left[\frac{1}{M_{\text{drag divergence}}\cos\Lambda_{c/4}} - M_{\text{drag divergence}}\cos\Lambda_{c/4}\right]^{\frac{1}{3}}\left\{1-\left[\frac{5+\left(M_{\text{drag divergence}}\cos\Lambda_{c/4}\right)^2}{5+\left(M^*\right)^2}\right]^{3.5}\right\}\right\}^{\frac{2}{3}} \tag{143}$$

The parameter M^* has no physical significance and is used as a measure of aerodynamic sophistication in obtaining supercritical flow. $M^* = 1.00$ for conventional airfoils with maximum thickness-to-chord at about 30% chord. $M^* = 1.05$ for high speed (peaky) airfoils. $1.12 \le M^* \le 1.15$ for supercritical airfoils.

Improved Drag Coefficient Estimation

The following method was developed before the advent of readily-accessible computers and is published in one appendix of K.D.Wood's design text. Start with a standard definition of wing efficiency factor

$$e_{\text{wing}} = \frac{\dfrac{1}{\pi AR}}{\dfrac{dC_{D_T}}{dC_L^2}} \tag{144}$$

then add decrements for the effects of bodies on the wing.

$$\frac{1}{e_{\text{airplane}}} = \frac{1}{e_{\text{wing}}} + \Sigma\left[\Delta\left(\frac{1}{e}\right)_{\text{parts}}\right] \tag{145}$$

The last term can be found in Figure 23 if we know characteristic dimensions and areas of the bodies.

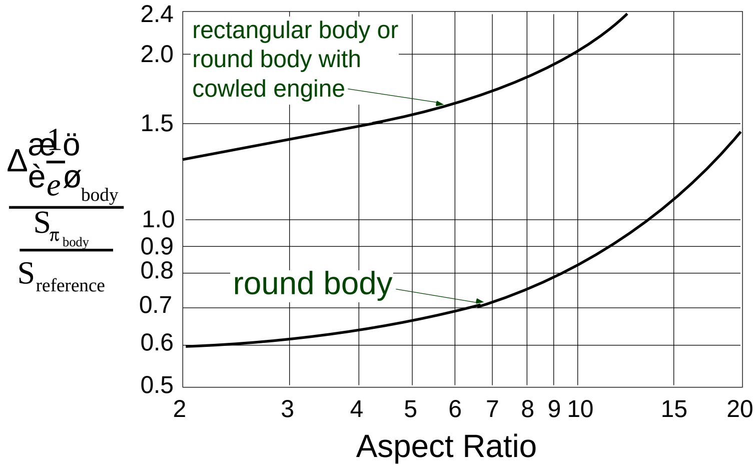


Figure 23. Body Effect on Airplane Efficiency Factor Referenced to Wing Aspect Ratio.[7]

The unfamiliar term in Figure 23, S_π, can be related to more familiar drag terms by the equation below:

$$f = C_f S_{\text{wetted}} = C_{D_f} S_{\text{reference}} = \frac{D_f}{q} = \Sigma C_{D_\pi} S_\pi \tag{146}$$

$$C_{D_f} = \frac{f}{S_{\text{reference}}} = C_{D_{P_{\min}}} \tag{147}$$

$$C_{D_0} = C_{D_{P_{\min}}} + 0.0058\left[1 + 1.5\left(\frac{t}{c}\right) + 125\left(\frac{t}{c}\right)^4\right] \tag{148}$$

This drag-buildup method calculates $C_{D_\pi} S_\pi$ for each body and uses that to estimate the total parasite drag coefficient. Use the area, S_π, to adjust efficiency factor. Table 6 presents K.D.Wood's method for estimating subsonic, incompressible parasite drag coefficient.

[7] NACATR-540.

Table 9. Approximate Parasite Drag of Airplane Components at Low Speed (<0.4 Mach) and Reynolds Numbers > 1,000,000.[8]

Part	Description	Length for Reynolds Number Calculations	Area for Drag Calculations	CDPRange
Wing	usual roughness with 0.10≤t/c≤0.20	chord	wing reference area	0.004-0.010
Flaps	60% span deflected 30°	chord	wing reference area	0.02-0.03
Tail	usual roughness with 0.08≤t/c≤0.12	chord	tail area	0.006-0.008
Fuselage	smooth streamlined body	length	maximum cross-section area	0.03-0.08
	large transport	length	maximum cross-section area	0.07-0.10
	bomber	length	maximum cross-section area	0.08-0.12
	small plane with nose engine	length	maximum cross-section area	0.09-0.20
Boat Hull	very low drag	boat hull length	maximum cross-section area	0.04-0.08
	designed for best water performance	boat hull length	maximum cross-section area	0.08-0.20
Nacelle	above wing, small airplane	length	maximum cross-section area	0.07-0.12
	in wing, large airplane	length	maximum cross-section area	0.04-0.07
	turbine engine	length	maximum cross-section area	0.04-0.07
External tanks	centered on wingtip	length	maximum cross-section area	0.05-0.07
	below wingtip	length	maximum cross-section area	0.07-0.10
	inboard below wing including support	length	maximum cross-section area	0.15-0.30
Bomb	below wing including support	length	maximum cross-section area	0.20-0.30
Float	best streamlining	float length	maximum cross-section area	0.05-0.08
	designed for best water performance	float length	maximum cross-section area	0.12-0.25
Landing Gear	nose wheel and strut	wheel diameter	wheel width*wheel diameter	0.50-0.80
	two well-faired wheels with pants	wheel diameter	wheel width*wheel diameter	0.15-0.30
	wheels and struts exposed	wheel diameter	wheel width*wheel diameter	0.30-0.50

This is an easy method to mechanize in a simple computer program after calculating fits for the curves in Figure 23. Polynomial curve fits that I've used in the past follow; however, Figure 23 more precise fits than the previous two equations should be used. For bodies with rectangular cross-sections:

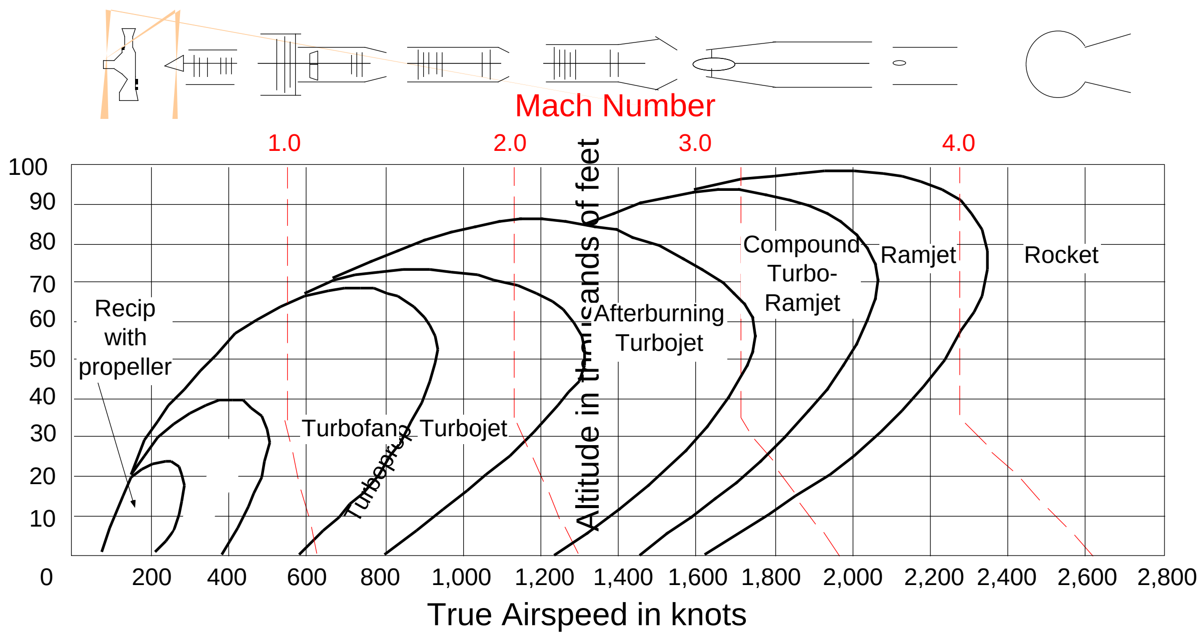
$$\left[\left(\frac{1}{e}\right)_{\text{part}}\right]_{\text{rectangular}} = 1.15 + 0.05917AR + 0.00875AR^2 - 0.00042AR^3 \tag{149}$$

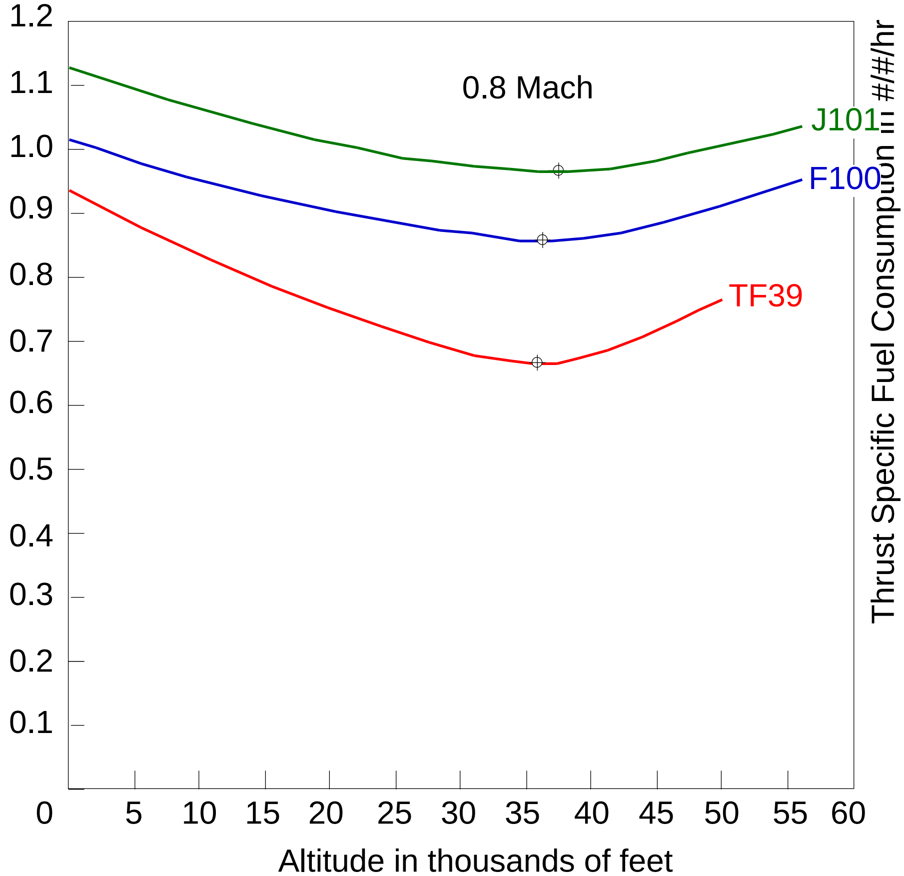
For bodies with round cross-sections:

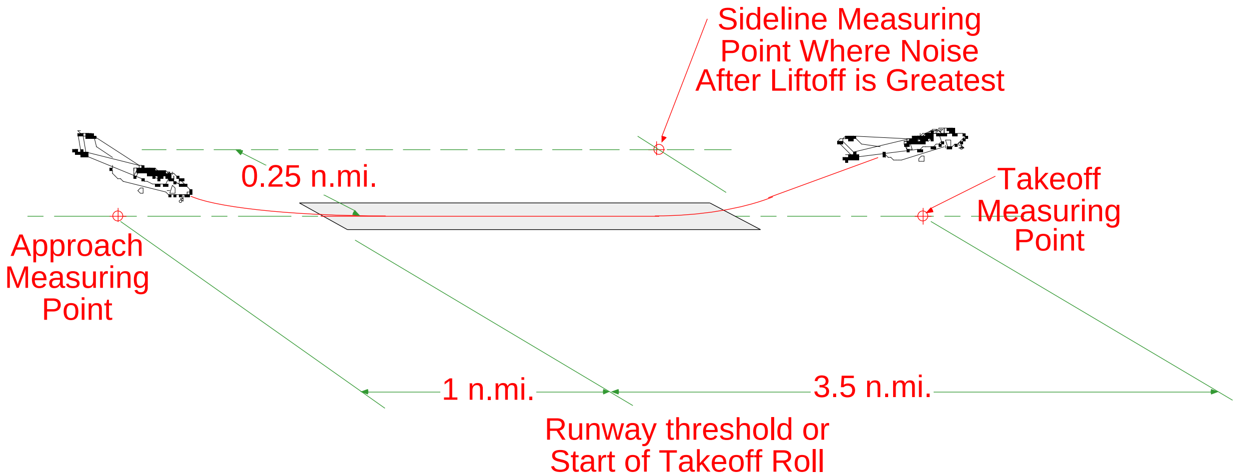
$$\left[\left(\frac{1}{e}\right)_{\text{part}}\right]_{\text{round}} = 0.5884 + 0.02002AR + 0.00027AR^2 + 0.000054AR^3 \tag{150}$$

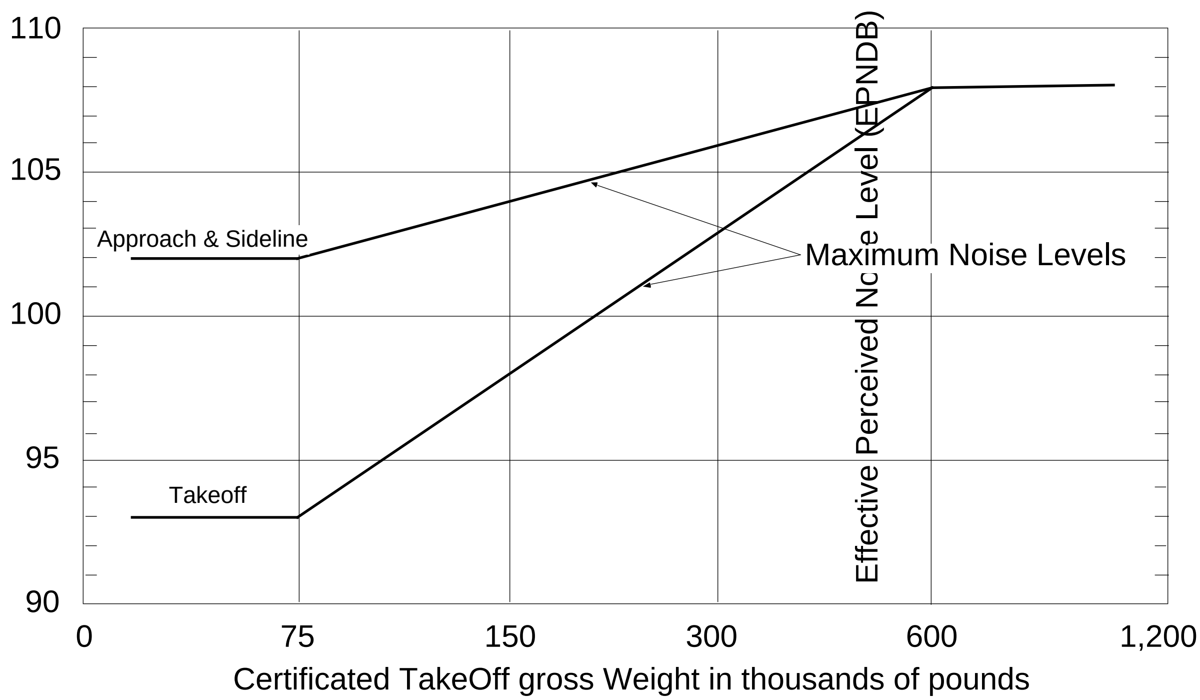
[8] Wood, p. A114.

CHAPTER 6
PROPULSION




1.2
1.1
1.0
0.9
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
5
10
15
20
25
30
35
40
45
50
55
60
0.8 Mach
J101
F100
TF39
Thrust Specific Fuel Consumption in #/#/hr
Altitude in thousands of feet


Sideline Measuring Point Where Noise After Liftoff is Greatest
0.25 n.mi.
Approach Measuring Point
Takeoff Measuring Point
1 n.mi.
3.5 n.mi.
Runway threshold or Start of Takeoff Roll


110
105
100
95
90
Approach & Sideline
Takeoff
Maximum Noise Levels
Effective Perceived Noise Level (EPNDB)
0
75
150
300
600
1,200
Certificated TakeOff gross Weight in thousands of pounds

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4. Shupe, H.W., and Machovina, P.E., *Engineering Geometry and Graphics* , McGraw-Hill, New York, NY, 1956
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APPENDIX A
AMODERN EXAMPLE OF THE CHERRY& CROSHERE METHOD

by
David W. Hall, P.E.
(originally written November 15, 1998)

Even though the foregoing Cherry and Croshere summary plot method is dated in terms of their example, the basic methodology is well-suited to parametric analysis of today's airplanes. Included here is an example of their approach applied to a design problem of interest to NASA's ERAST program. Note that the basic equations can be coded into a spreadsheet or computer program.

Mission to be Examined

One high altitude and long endurance (HALE) mission of current military and civilian science interest is to provide a platform to stand off from a location a given distance at a specified altitude and cruise speed to provide either military surveillance capability or a repositionable suborbital science platform. In this case, the airplane must be capable of transiting approximately 2,000 nautical miles (n.mi.) to its loiter area within five hours after taking off from a 3,500 foot runway at sea level on a standard day. It must carry its full payload of 500 pounds throughout the mission. On return to its airport of origin, it must have a landing speed of 100 knots or less and a landing distance of no more than 3,500 feet. The vehicle may be uninhabited, but cost estimates must reflect ground station requirements if this option is chosen. In addition, the aircraft must be capable of meeting FAR part 23 or Part 25, depending on TakeOff Gross Weight and applicable civilian airport sound level requirements

Step One: Quantify Requirements and State as Inequalities

$S_{takeoff} \leq 3{,}500$ feet

$S_{transit} \geq 2{,}000$ n.mi.

$V_{landing} \leq 100$ knots

$S_{landing} \leq 3{,}500$ feet

$W_{payload} \geq 500$ pounds

Step Two: Begin Constructing Applicable Equations

Takeoff distance here is from Roskam I-3.8.

$$\left(s_{FL}\right)_{takeoff} = \frac{37.5}{\sigma * \left(C_{L_{\max}}\right)_{takeoff} * \frac{T_{sls}}{W}} * \frac{W}{S_{ref}} \leq 3{,}500\, feet$$

Note various ways of expressing drag.

$$T_{req} = T_{av} = D_{cruise} = q_{cruise} * S_{ref} * C_{D_T}$$

$$T_{req} = 0.5 * \rho_{cruise} * V_{cruise}^2 * S_{ref} * \left(\left(C_{D_0}\right)_{cruise} + \frac{\left(C_L\right)_{cruise}^2}{\pi * e * AR} \right)$$

$$\left(C_L\right)_{cruise} = \frac{2 * W}{\rho_{cruise} * S_{ref}} * \frac{1}{V_{cruise}^2}$$

$$\left(T_{av}\right)_{cruise} = 0.5 * \rho_{cruise} * \left(C_{D_0}\right)_{cruise} * S_{ref} * V_{cruise}^2 + \frac{4 * W^2}{\pi * e * AR * \rho_{cruise} * S_{ref}} * \frac{1}{V_{cruise}^2}$$

$$T_{cruise} = \sigma_{cruise} * T_{sls} * n_{engines}$$

$$\left(\frac{L}{D}\right)_{cruise} = \left(\frac{W}{T_{cruise}}\right)$$

$$T_{av} = W * \frac{\left(C_{D_T}\right)_{cruise}}{\left(C_L\right)_{cruise}}$$

Breguet range for turbofans is

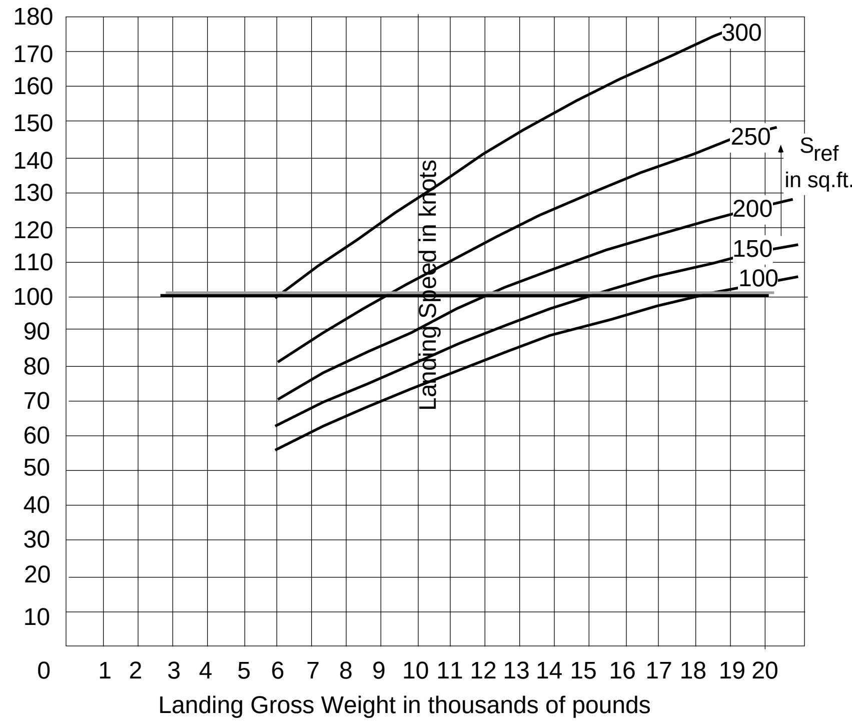
$$R = \frac{V_{cruise}}{tsfc} * \left(\frac{L}{D}\right)_{cruise} * \ln\left(\frac{W_{initial}}{W_{initial} - W_{final}}\right)_{cruise}$$

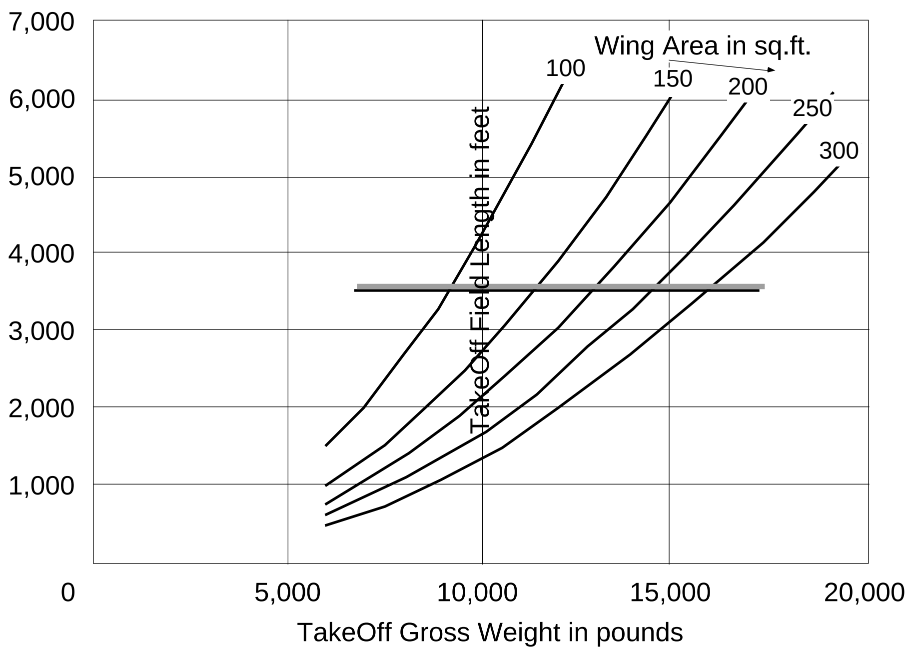
Landing speed is a multiple of stall speed.

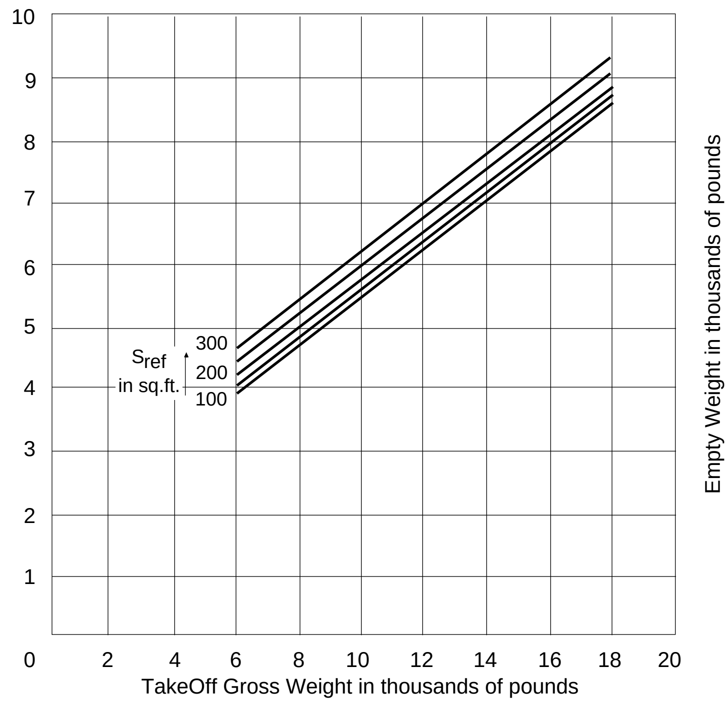
$$V_{landing} = 1.1 * \left(V_{stall}\right)_{fullflaps} = 1.1 * \sqrt{\frac{2}{\rho * C_{L_{max}}}} * \sqrt{\frac{W_{landinggross}}{S_{ref}}}$$

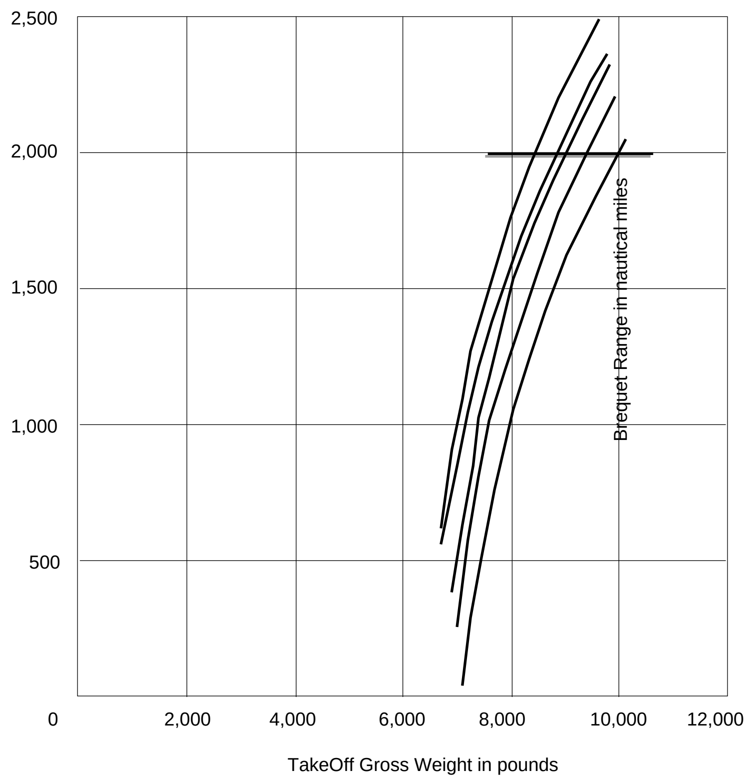
Here’s an opportunity to examine the choice of high lift system.

Step Three: Plot

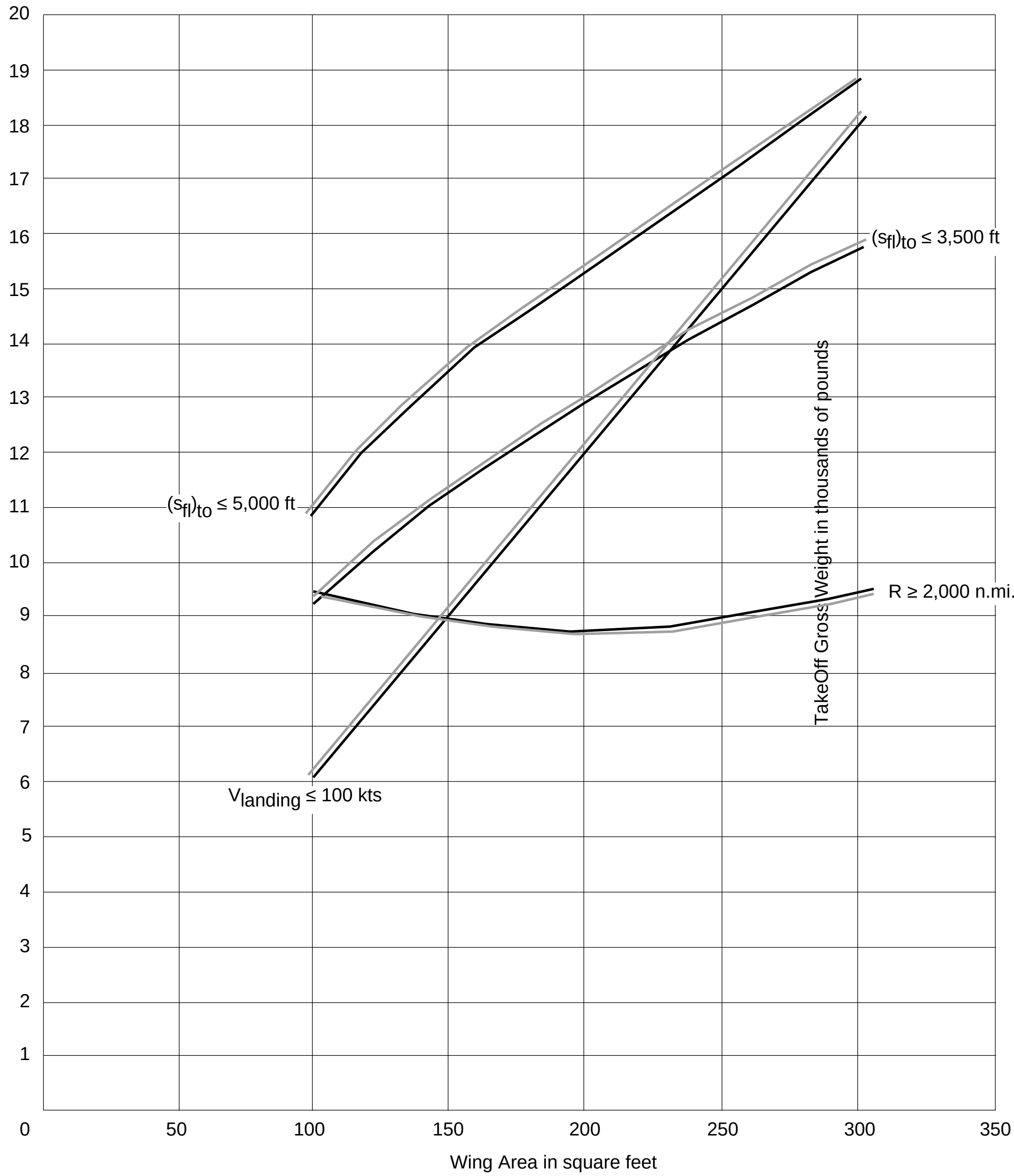

180
170
160
150
140
130
120
110
100
90
80
70
60
50
40
30
20
10
0
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20
Landing Gross Weight in thousands of pounds
Landing Speed in knots
300
250
200
150
100
S_{ref}
in sq.ft.


7,000
6,000
5,000
4,000
3,000
2,000
1,000
0
5,000
10,000
15,000
20,000
TakeOff Gross Weight in pounds
TakeOff Field Length in feet
Wing Area in sq.ft.
100
150
200
250
300


10
9
8
7
6
5
4
3
2
1
0
2
4
6
8
10
12
14
16
18
20
300
200
100
S_{ref}
in sq.ft.
TakeOff Gross Weight in thousands of pounds
Empty Weight in thousands of pounds


2,500
2,000
1,500
1,000
500
0
2,000
4,000
6,000
8,000
10,000
12,000
Brequet Range in nautical miles
TakeOff Gross Weight in pounds

Step Four: Cross-Plot as Summary Curves



I've left out several sets of calculations to keep the example simple. The key step is to cross-plot all the constraint curves onto a common set of axes, in this case takeoff gross weight versus wing area, but it could just as easily have been thrust-to-weight ratio versus wing loading. This latter makes comparison between any two platforms in any performance class straightforward.