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Landing Gear Integration in Aircraft Conceptual Design
by
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Abstract

The design of the landing gear is one of the more fundamental aspects of aircraft design. The design and integration process encompasses numerous engineering disciplines, e.g., structure, weights, runway design, and economics, and has become extremely sophisticated in the last few decades. Although the design process is well-documented, no attempt has been made until now in the development of a design methodology that can be used within an automated environment. As a result, the process remains to be a key responsibility for the configuration designer and is largely experience-based and graphically-oriented. However, as industry and government try to incorporate multidisciplinary design optimization (MDO) methods in the conceptual design phase, the need for a more systematic procedure has become apparent.

The development of an MDO-capable design methodology as described in this work is focused on providing the conceptual designer with tools to help automate the disciplinary analyses, i.e., geometry, kinematics, flotation, and weight. Documented design procedures and analyses were examined to determine their applicability, and to ensure compliance with current practices and regulations. Using the latest information as obtained from industry during initial industry survey, the analyses were in terms modified and expanded to accommodate the design criteria associated with the advanced large subsonic transports. Algorithms were then developed based on the updated analysis procedures to be incorporated into existing MDO codes.
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Nomenclature

Symbols

\( A \) Area

\( ACN \) Aircraft classification number

\( AR \) Aspect ratio

\( C \) Clearance, moment component

\( C_L \) Lift coefficient

\( CBR \) California bearing ratio

\( D \) Drag, diameter

\( ESWL \) Equivalent single wheel load

\( E \) Energy, modulus of elasticity

\( F \) Fillet radius, force

\( F.S. \) Factor of safety

\( H \) Height

\( I \) Moment of inertia, second area moment

\( KE \) Kinetic energy

\( L \) Lift, length

\( M \) Bending moment

\( MTOW \) Maximum takeoff weight

\( N \) Landing load factor, axial force

\( P \) Pressure, load

\( R \) Radius of centerline curve, offset

\( S \) Safety margin, wing area, stroke, offset

\( T \) Thrust, torque

\( V \) Speed, volume, shear force

\( W \) Weight, width, offset

\( X \) Distance

\( a \) Acceleration
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<td>$b$</td>
<td>Wheelbase</td>
</tr>
<tr>
<td>$c$</td>
<td>Constant</td>
</tr>
<tr>
<td>$cg$</td>
<td>Center of gravity</td>
</tr>
<tr>
<td>$d$</td>
<td>Displacement</td>
</tr>
<tr>
<td>$e$</td>
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<tr>
<td>$f$</td>
<td>Constant</td>
</tr>
<tr>
<td>$g$</td>
<td>Gravitational acceleration</td>
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<tr>
<td>$h$</td>
<td>Height</td>
</tr>
<tr>
<td>$k$</td>
<td>Modulus of subgrade reaction, machinability factor</td>
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<tr>
<td>$l$</td>
<td>Length, direction cosine, radius of relative stiffness</td>
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<td>$m$</td>
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**Greek Letters**

- $\Gamma$: Dihedral angle
- $\Lambda$: Sweep angle
- $\Psi$: Turnover angle
- $\alpha$: Angle of attack, repetition factor
- $\beta$: Steering angle
- $\delta$: Main assembly offset angle
- $\phi$: Roll angle, retraction angle
- $\eta$: Efficiency factor
\[\varphi\quad\text{Castor angle, rotation angle}\]
\[\mu\quad\text{Poisson's ratio}\]
\[\pi\quad\text{Pi}\]
\[\theta\quad\text{Pitch angle, axle inclination angle}\]
\[\rho\quad\text{Sea-level air density, radius of gyration}\]
\[\sigma\quad\text{Stress}\]
\[\tau\quad\text{Stress}\]

**Acronyms**

ACN-PCN Aircraft-Pavement Classification Number
ACSYNT AirCraft SYNThesis
FAA Federal Aviation Administration
ICAO International Civil Aviation Organization
LCN Load Classification Number
MDO Multidisciplinary Design Optimization
NASA National Aeronautic and Space Administration
PCA Portland Cement Association
FLOPS Flight Optimization System
DIS Dynamic Integration System
Chapter 1  Introduction

1.1. Introduction

The design of the landing gear, which is considered “the essential intermediary between
the aeroplane and catastrophe” [1], is one of the more fundamental aspects of aircraft
design. The design and integration process encompasses numerous engineering disciplines,
e.g., structures, weights, runway design, and economics, and has become sophisticated in
the last few decades.

The landing gear design process is well-documented by Conway [1] and more recently
by Currey [2] and is experience-based and graphically-oriented in nature. As such, it is a
key responsibility of the configuration designer during initial concept studies. However, as
industry and government try to incorporate multidisciplinary design optimization (MDO)
methods in the conceptual design phase, the need for a more systematic procedure has
become apparent. Accordingly, NASA Ames provided Virginia Tech with a two-year
research grant to develop a landing gear design methodology that can be implemented
within an MDO environment, with a special emphasis on design considerations for
advanced large subsonic transports. The result of this research project, known as Landing
Gear Integration in Aircraft Conceptual Design, is the topic of this report.

1.2. Overview

Several design considerations that must be addressed are briefly discussed to illustrate
the complexity involved in the development of such a methodology. The list is made up of
an ever-increasing, and sometimes conflicting, number of requirements, e.g., component
maximum strength, minimum weight, high reliability, low cost, overall aircraft integration,
airfield compatibility, etc., and truly reflect the multidisciplinary nature of the task.

The weight of the landing gear, which typically ranges from three to six percent of the
maximum aircraft takeoff weight, is also a design consideration. With advances in flight
science technologies, which result in reduced structural and mission fuel weights, the
landing gear may become an increasingly large weight fraction in future large aircraft.
Figure 1.1 illustrates the issue. Several typical weight estimating equations are compared
with data tabulated by Roskam[3]. The figure also contains recent estimates from Boeing and Airbus for their proposed new large aircraft designs. In particular, note the difference in trends between the ACSYNT and Douglas and Torenbeek equations for weights above a million pounds. It is particularly interesting to note that the Airbus estimate agrees with both the ACSYNT and Douglas equations. However, which trend is correct? Because the curves cross at this point, it is impossible to tell which estimate is appropriate. Thus, one design objective of the study is to be able to estimate the weight of the landing gear early in the design phase using a first principals analysis. A major reduction in the landing gear weight may be hard to realize because landing gears are one of the few non-redundant load-paths in an aircraft, and any reduction in reliability from current fail-safe standard is not acceptable [4].

![Graph showing comparison of weight equations with aircraft weights data.](image)

Figure 1-1. Initial comparison of weight equations with aircraft weights data.

The location of the aircraft center of gravity (cg) is critical in the design and location of the landing gear. The nose and main assemblies must be located within specific distances from the aircraft cg, in both the longitudinal and lateral directions, such that the aircraft is in
no danger of tipping back\(^1\) or turning over on its side over the full range of cg locations under static or dynamic conditions. Another issue to be considered is the distribution of the aircraft weight, which is dependent on the distances between the aircraft cg and the nose and main assembly. Between 85 and 92 percent of the MTOW must be maintained on the main assemblies such that the brakes can provide sufficient energy to slow down the aircraft within a given runway length [5].

Airfield compatibility has become one of the primary considerations in the design of landing gears due to the high cost associated with infrastructure modification, e.g., pavement reinforcement and runway and taxiway expansion[6].\(^2\) Pavement bearing strength, which varies from one airport to another due to variations in subgrade materials, dictates the number and arrangement of tires needed to produce the required flotation characteristics. Flotation is defined as the capability of the runway pavement and other surfaces, e.g., taxiway and apron, to support the aircraft. In addition, the disposition of the landing gear is constrained by the runway and taxiway geometry as found at the airports to be served. Since the ground track is dependent on the dimensions of the wheelbase and track, an increase in these dimensions could bring the aircraft over the edge of the pavement during certain maneuvers, e.g., a 180-degree turn and centerline-tracing taxiing, and cause the aircraft to bog down in soft soil [7].

\(^1\) Some aircraft have tail props to ensure that the aircraft does not tip back while parked at the gate. From the AIAA case study series on the 727: "When the first National Airlines 727-200 rolled to a stop as it was delivered in Miami, the pilot touched the brakes, the airplane nose went down and then recoiled up, lifting the nose gear off the concrete about 6 to 8 inches. The gasp in the crowd were heard 3,000 miles away in Seattle. ... As far as we know, no 727-200 has ever sat on its tail and maybe we overreacted to the National incident, but that's why you will nearly always see a 727 with its rear airstairs down when parked. There are some rare cases where we attach lead to the radome bulkhead for extreme loading conditions." Note that the cg range of the 727-200 ranges from 8 to 42\% of the mac.

\(^2\) The prototype B-36 had single large main wheels, 110 inches in diameter. They were the largest aircraft wheels ever made. They required a 22 1/2 inch thick runway, thus limiting the prototype to three specially strengthened runways, those at Fort Worth, Eglin AFB and Fairfield-Suisan AFB (later Travis AFB also). A multi-wheel gear could not be obtained until adequate brakes could be designed. Finally, a four-wheel gear using 56 inch diameter tires was perfected for the B-36A. A 13 1/2 inch thick runway was needed, and 22 primary and a further 22 alternate air fields could handle the production bombers. (source: Meyers K. Jacobson and Ray Wagner, B-36 in action, squadron/signal publications Aircraft No. 42, 1980, the initial TOGW of the B-36 was 265,000 lbs, and grew to 360,000 lbs.).
The soundness of a landing gear concept depends on the efficacy of overall system integration. Ground clearance, particularly between the engine nacelle and the static groundline, plays a key role in determining the length of the landing gear and the permissible takeoff rotation angle. Insufficient allowance can result in costly modifications, e.g., lengthening of the strut with concomitant stowage constraints or complicated strut shrinkage mechanisms, or repositioning of the under-wing engines, that effectively rule out future growth options.* The landing gear stowage issue must also be addressed as the number of main assembly struts increases with the increase in aircraft weight [8]. Trade-off studies concerning space availability, structural integrity, and weight penalties resulting from local structural reinforcements are needed to arrive at an optimum design.

With the financial challenges arising from the deregulation of the air-travel industry, airlines need to reduce operating costs to remain competitive. As a result, airlines are demanding that aircraft manufacturers produce new designs with high reliability and low maintenance requirements. Recent technologies, e.g., carbon-carbon heat sinks, radial tires, and high-strength steel, are being introduced. In addition, simplified design and improved manufacturing techniques, e.g., die-forging and three-dimensional machining [9], are being used to reduce the part-count associated with the landing gear system.

1.3. Objectives

The development of an MDO-capable design methodology is focused on providing the conceptual designer with tools to help automate the disciplinary analyses, i.e., geometry, kinematics, flotation, and weight. Documented design procedures and analyses as found and referenced by Curry [2] and Torenbeek [3] were examined to determine their applicability, and to ensure compliance with current practices and regulations. Although in most cases the documented analyses were developed for a specific type of aircraft, the essential fundamentals remain unchanged for any type of aircraft. Thus, using the latest information as obtained from industry during an initial industry survey [App. A], the analyses were developed to accommodate the design criteria associated with possible advanced large subsonic transports. Algorithms were then developed based on the updated analysis procedures as a package to be incorporated into existing MDO codes.

* This was the case with the Boeing 727.
Chapter 2  Aircraft Center of Gravity

2.1. Introduction

The precise location of the aircraft cg is essential in the positioning of the landing gear, as well as for other MDO applications, e.g., flight mechanics, stability and control, and performance. Primarily, the aircraft cg location is needed to position the landing gear such that ground stability, maneuverability, and clearance requirements are met. Given the fact that none of the existing conceptual design-level cg estimation procedures has the degree of responsiveness and accuracy required for MDO applications, a new approach is formulated to provide a reliable range of cg locations that is better suited for MDO applications.

The connection between the landing gear and the cg has become even more critical with the adoption of advanced control systems. As pointed out by Holloway[10] in 1971, and illustrated here in Fig. 2.1, once the aft cg limit is no longer based on stability but on the ability to generate the required nose down pitching moment, the wing tends to move forward relative to the cg and the landing gear may “fall off” the wing. Thus, the tip-back angle may become an important consideration in determining the aft cg limit. Sliwa identified this issue in his aircraft design studies.[11]

2.2. Current Capabilities

Although not expected to determine the location of the aircraft cg, current aircraft sizing programs, as typified by Jayaram et al. [12] and McCullers [13], do provide some rudimentary estimates. These codes use estimated component weights obtained from statistical weight equations, and either user-specified or default component cg locations to arrive at the overall aircraft cg location. However, as demonstrated by Chai et al. [14], the lack of responsiveness and accuracy have rendered current approaches inadequate for MDO application.
Figure 2.1 Typical tail sizing chart with tip back limit becoming the aft cg limit for relaxed static stability aircraft (after Holloway, et al., [10]).

The lack of responsiveness is attributed to the fact that each aircraft component is assigned a specific location within the airframe. Typically, these approaches do not estimate the operational range of cg locations. The cg location is a complicated function of the configuration, loading, and fuel state, with an allowable range limited by a number of operational factors [15]. Although a range of cg locations can be established by varying the configuration, equipment arrangement, and payload and fuel states individually, the process is difficult. The accuracy limitations arise because the codes assume that the user has the experience and knowledge required to make adjustments to the component weight and cg estimates. Unfortunately, this approach is not suitable for use in automated procedures required in MDO.

Evidently, what is needed is a new approach which is capable of establishing a maximum permissible cg range for a given configuration. This available cg range can then be compared with the desired operational cg range obtained from performance, control, and operational requirements. If the desired cg range is within the available cg range, the
concept is viable and can be balanced. If not, the configuration must be changed, either by the designer or an MDO procedure if an automated process is being used.

2.3. *Alternate Method*

Component location flexibility at the conceptual design phase is actively exploited as a means to improve the responsiveness and accuracy of current cg estimation procedures. In the proposed procedure, aircraft components are assigned a range of cg locations based on the geometry, as well as physical and functional considerations, associated with each component. By arranging the cg of the components at their fore- and aft-most limits, the maximum permissible cg range of a particular layout can be established. This cg range can then be used by an MDO procedure to determine the forward and aft aircraft cg limits required to meet performance and stability and control considerations. Adjusted for uncertainty, this maximum permissible cg range can be used as a constraint for the operational cg range during the optimization.

2.3.1. *Establishment of Component CG Range*

The assignment of component cg range is based on the geometry, planform, and the type of components involved. In the case of the primary components, e.g., fuselage, wing, and empennage, the location of these items remains relatively unchanged once the concept is frozen. Consequently, the cg range is expected to be centered near the volumetric center of the component and is unlikely to shift too much. For ease of identification, the primary components will be referred to as the *constrained* items.

As for secondary components, e.g., equipment and operational items, the location of each component varies from one aircraft concept to another, depending on the philosophy and preference of the airframe manufacturer. Note that as long as the stowage and functionality constraints are not violated, these components can be assigned to any available space throughout the aircraft due to their compactness. Consequently, the
corresponding cg range is defined by the forward and aft boundaries of the stowage space within which the item is located. Accordingly, these components are termed the unconstrained items.

Although the payload and passenger amenity, i.e., furnishings and services, are confined within the cargo holds and cabin, operational experience has shown that the cg location of these items varies according to the loading condition and cabin layout as specified by the airlines, respectively. Similarly, the cg location of the fuel varies as a function of time as the fuel is being consumed during the duration of the mission. Given the added freedom in terms of the loading pattern, these components are also classified as unconstrained items.

2.3.2. Generic Component Layout

The proposed aircraft component cg ranges are listed in Table 2.1 and represented graphically in Fig. 2.2. The ranges are based on the layout of existing commercial transports [16 and 17] and can be modified to accommodate any unique layout of the aircraft concept under consideration.

The locations of the front and rear spar for the wing and empennage are dictated by space required for housing the control surfaces and the associated actuation systems, where values of 15 and 65 percent chord, respectively, are typically used. As in the conventional cantilever wing and empennage construction, the majority of the structure, i.e., bulkheads, ribs, and fuel tanks, are located between the front and rear spars. Thus, it can be expected that the cg of the wing is most likely to be located between the two, along the respective mean aerodynamic chords (mac). In addition, given the physical arrangement of the fuel tanks, the cg of the fuel and the fuel system can be expected to be located near the same vicinity.
Table 2.1 Generic component location for conventional civil transports

<table>
<thead>
<tr>
<th>Component</th>
<th>Type</th>
<th>Component cg range</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>Constrained</td>
<td>Between fore and aft spars along wing <em>mac</em></td>
</tr>
<tr>
<td>Fuselage</td>
<td>Constrained</td>
<td>40 to 50 percent fuselage length</td>
</tr>
<tr>
<td>Horizontal tail</td>
<td>Constrained</td>
<td>Between fore and aft spars along horizontal tail <em>mac</em></td>
</tr>
<tr>
<td>Vertical tail</td>
<td>Constrained</td>
<td>Between fore and aft spars along vertical tail <em>mac</em></td>
</tr>
<tr>
<td>Engines/Nacelles</td>
<td>Constrained</td>
<td>45 to 60 percent engine length</td>
</tr>
<tr>
<td>Nose gear</td>
<td>Constrained</td>
<td>Between fore and aft wheelwell bulkheads</td>
</tr>
<tr>
<td>Main gear</td>
<td>Constrained</td>
<td>Between fore and aft wheelwell bulkheads</td>
</tr>
<tr>
<td>Fuel system</td>
<td>Unconstrained</td>
<td>Between fore and aft spars along wing <em>mac</em></td>
</tr>
<tr>
<td>Hydraulics</td>
<td>Unconstrained</td>
<td>Between fore and aft wing spars along aircraft centerline;</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Between aft pressure bulkhead and tip of tailcone</td>
</tr>
<tr>
<td>Electrical system</td>
<td>Unconstrained</td>
<td>Between forward pressure bulkhead and nose wheelwell;</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Between fore and aft wing spars along aircraft centerline</td>
</tr>
<tr>
<td>Avionics</td>
<td>Unconstrained</td>
<td>Between forward pressure bulkhead and nose wheelwell</td>
</tr>
<tr>
<td>Instrumentation</td>
<td>Unconstrained</td>
<td>Between forward pressure bulkhead and nose wheelwell</td>
</tr>
<tr>
<td>Environmental</td>
<td>Unconstrained</td>
<td>Between fore and aft wing spars along aircraft centerline</td>
</tr>
<tr>
<td>Flight control</td>
<td>Unconstrained</td>
<td>Between aft spar and trailing-edge along surface <em>mac</em></td>
</tr>
<tr>
<td>Auxiliary power</td>
<td>Unconstrained</td>
<td>Between aft pressure bulkhead and tip of tailcone</td>
</tr>
<tr>
<td>Furnishings</td>
<td>Unconstrained</td>
<td>45 to 60 percent cabin length</td>
</tr>
<tr>
<td>Services</td>
<td>Unconstrained</td>
<td>45 to 60 percent cabin length</td>
</tr>
<tr>
<td>Passengers</td>
<td>Unconstrained</td>
<td>45 to 60 percent cabin length</td>
</tr>
<tr>
<td>Cargo</td>
<td>Unconstrained</td>
<td>45 to 55 percent forward and aft cargo holds</td>
</tr>
<tr>
<td>Fuel</td>
<td>Constrained</td>
<td>Between fore and aft spars along wing <em>mac</em></td>
</tr>
<tr>
<td></td>
<td></td>
<td>Between fore and aft wing spars along aircraft centerline</td>
</tr>
</tbody>
</table>
The cg of the fuselage depends on the structural arrangement of the pressure bulkheads, frames, and the aft-body taper ratio. Other factors include local structural reinforcement around the landing gear wheelwells, cargo holds, and the layout of the cabin, e.g., a forward upper-deck as found on the Boeing Model 747 or a double-decker as found on the proposed ultra-high-capacity transports. Taking these factors into consideration, the proposed procedure assumes that the cg of the fuselage is most likely to be located between 40 and 50 percent of the fuselage length.

Figure 2.2 Ranges of available component cg locations

The cg of the engine group varies according to the dimensions of the engine, nacelle, and engine pylon. To account for weight-affecting factors such as compressor fan diameter, the shape of the nacelle, thrust reverser and pylon structure arrangement, forward and aft cg limit of 45 and 60 percent of the length of the engine, respectively, were assigned.
Regardless of the configuration of the landing gear, the cg of the landing gear will be
confined between the landing gear wheelwells in flight. Thus, the forward and aft cg limits
of the landing gear are assumed to coincide with the forward and aft stowage volume
boundaries of the nose and main assembly wheelwells.

Hydraulics is divided into the wing and empennage group, with the weight
proportional to the ratio of the respective control surface area to the total control surface
area. The wing group is assumed to be located beneath the wing torsion box, which results
in a cg range that is defined by the fore and aft wing spars along the aircraft centerline. On
the other hand, the cg range of the empennage group is limited to the space behind the aft
pressure bulkhead. Besides providing the stowage volume for the empennage hydraulics,
the tail cone space also houses the auxiliary power unit.

Similarly, flight controls are divided into the wing and empennage group, with the
weight proportional to the ratio of the local control surface area to the total control
surface area. The proposed procedure assumes that the weight of the leading-edge control
surfaces is negligible and that the trailing-edge control surfaces are in the retracted
position. Thus, the cg of the flight controls are bounded by the rear spar and the trailing
edge of each surface, along the respective macs.

The electrical system is divided into the battery and generator groups, assuming that
the weight is distributed evenly between the two. The battery group is to be located
between the forward pressure bulkhead and the nose wheelwell, although it can also be
located in the cavity between the nose wheelwell and the forward cargo hold. The
generator group is to share the wing-body fairing cavity as being used to stow the wing
hydraulics, i.e., under the wing torsion box. Due to functionality constraints, avionics and
instrumentation are assumed to be located in the same compartment which houses the
batteries. Similarly, environmental control packs are to share the wing-body fairing cavity
with the electrical generator and wing hydraulic groups.
Given that the aircraft is fully loaded, the cg of the furnishings, services, and passengers is limited to between 45 and 60 percent of the cabin length. This assumption takes into account the distribution of the passengers and the corresponding arrangement of the furnishings and passenger services in different cabin layouts. To accommodate the variable nature of the cargo loading operation, which is affected by the type and weight of the baggage and bulk materials, forward and aft cg limits of 45 and 55 percent, respectively, of both forward and rear cargo holds were assigned.

2.3.3. Validation of Analysis

A simple spreadsheet software, where the component cg range data as presented in Table 2.1 are stored and a macro is defined for calculation purposes, is created to establish the forward and aft limits of the permissible aircraft cg range. A detailed description of the spreadsheet can be found in Chapter Nine. The Boeing Models 737, 747, 767, and McDonnell Douglas DC-10 were used to validate the proposed cg estimation procedure as outlined above. Estimated component weights were obtained from ACSYNT(AirCraft SYNThesis) [12] and used for all four aircraft, while component cg ranges were determined using the generic layout as detailed in the previous section. Essentially, the four aircraft are treated as conceptual aircraft. The objective here is to determine if the maximum permissible cg range as established by the new approach can enclose the actual operational cg range. Actual [18] and estimated aircraft cg ranges determined using the spreadsheet are listed in Table 2.2, both sets of data are shown in Fig. 2.3 for ease of comparison.

<table>
<thead>
<tr>
<th>Aircraft</th>
<th>Estimated, % mac</th>
<th>Actual, % mac</th>
</tr>
</thead>
<tbody>
<tr>
<td>B737 (forward/aft)</td>
<td>0.0/68.0</td>
<td>12.0/30.0</td>
</tr>
<tr>
<td>B767 (forward/aft)</td>
<td>-4.0/67.0</td>
<td>11.0/32.0</td>
</tr>
<tr>
<td>DC10 (forward/aft)</td>
<td>-7.0/46.0</td>
<td>8.0/18.0</td>
</tr>
<tr>
<td>B747 (forward/aft)</td>
<td>4.0/63.0</td>
<td>13.0/33.0</td>
</tr>
</tbody>
</table>
As shown in Fig. 2.3, the new approach is capable of producing a permissible aircraft cg range that brackets in the actual operational cg range for all four aircraft. In addition, the estimated cg range offers a generous margin at either end-limit of the band representing the actual operational cg range. Since both the weight and location of the components are based on statistical information, the margin would ensure that the operational cg range remains within the obtainable range even when the uncertainty is included. Evidently, the proposed cg estimation procedure is able to meet the flexibility and reliability requirements that are essential for MDO applications.
Chapter 3  Landing Gear Concept Selection

3.1. Introduction

The design and positioning of the landing gear are determined by the unique characteristics associated with each aircraft, i.e., geometry, weight, and mission requirements. Given the weight and cg range of the aircraft, suitable configurations are identified and reviewed to determine how well they match the airframe structure, flotation, and operational requirements. The essential features, e.g., the number and size of tires and wheels, brakes, and shock absorption mechanism, must be selected in accordance with industry and federal standards discussed in the following chapters before an aircraft design progresses past the concept formulation phase, after which it is often very difficult and expensive to change the design [19]. Three examples of significant changes made after the initial design include the DC-10-30, which added the third main gear to the fuselage, the Airbus A340, where the main gear center bogie increased from two to four wheels in the -400 series, and the Airbus A-300, where the wheels were spread further apart on the bogie to meet LaGuardia Airport flotation limits for US operators.

Based on the design considerations as discussed in this chapter, algorithms were developed to establish constraint boundaries for use in positioning the landing gear, as well as to determine whether the design characteristics violate the specified requirements. The considerations include stability at takeoff/touchdown and during taxiing, braking and steering qualities, gear length, attachment scheme, and ground maneuvers.

3.2. Configuration Selection

The nose wheel tricycle undercarriage has long been the preferred configuration for passenger transports. It leads to a nearly level fuselage and consequently the cabin floor when the aircraft is on the ground. The most attractive feature of this type of undercarriages is the improved stability during braking and ground maneuvers. Under normal landing attitude, the relative location of the main assembly to the aircraft cg produces a nose-down pitching moment upon touchdown. This moment helps to reduce the angle of attack of the aircraft and thus the lift generated by the wing. In addition, the braking forces, which act behind the aircraft cg, have a stabilizing effect and thus enable the pilot to make full use of the brakes. These factors all contribute to a shorter landing field length requirement.
The primary drawback of the nose wheel tricycle configuration is the restriction placed upon the location where the main landing gear can be attached. With the steady increase in the aircraft takeoff weight, the number of main assembly struts has grown from two to four to accommodate the number of tires required to distribute the weight over a greater area. However, stability and performance constraints as identified by Holloway et al. [10] and Sliwa [11] effectively eliminate all but a few locations where the main assembly can be attached. The attachment limitation phenomenon is known as the location stagnation [App. A] and can become a major concern for future large aircraft, where additional tires and struts are required to alleviate the load being applied to the pavement. Typically, a large trailing-edge extension, i.e., the Yehudi, is employed to alleviate at least in part the location stagnation problem. The Yehudi can result in weight and aerodynamic penalties due to local structural reinforcement and increased wetted area, respectively. However, the increased root chord also allows an increase in absolute root thickness for a given t/c. This advantage may outweigh other penalties.

3.3. Landing Gear Disposition

The positioning of the landing gear is based primarily on stability considerations during taxiing, liftoff and touchdown, i.e., the aircraft should be in no danger of turning over on its side once it is on the ground. Compliance with this requirement can be determined by examining the takeoff/landing performance characteristics and the relationships between the locations of the landing gear and the aircraft cg.

3.3.1. Angles of Pitch and Roll During Takeoff and Landing

The available pitch angle (θ) at liftoff and touchdown must be equal, or preferably exceed, the requirements imposed by performance or flight characteristics. A geometric limitation to the pitch angle is detrimental to the liftoff speed and hence to the takeoff field length. Similarly, a geometric limitation to the roll angle (ϕ) could result in undesirable operational limit under cross-wind landing condition.

1Apparently known as a “Yehudi”, this inboard trailing edge extension actually first appeared on the Boeing B-29 to solve a fuselage-nacelle interference problem. Douglas used it first on a swept wing transport on the DC-8, and it was not adopted by Boeing until the 707 design went to the -320 model. The name was first used to describe the wind tunnel part that was made on the spot during the wind tunnel test. “Who’s Yehudi” was a running gag on a popular radio show at the time, as well as the name of a popular violinist (letter from Bill Cook, retired Boeing engineer and author of The Road to the 707).
For a given aircraft geometry and gear height \((h_g)\), the limit for the takeoff/landing pitch angle follows directly from Fig. 3.1. The roll angle at which the tip of the wing just touches the ground is calculated using the expression [5, p. 350]

\[
\tan \phi = \tan \Gamma + \frac{2h_g}{s - t} - \tan \theta \tan \Lambda
\]  

(3.1)

In this case, \(\Gamma\) is taken as the dihedral angle, \(s\) is the wing span, \(t\) is the wheel track, and \(\Lambda\) is the wing sweep. Similar conditions may be deduced for other parts of the aircraft, except that \(\Gamma\), \(\Lambda\) and \(s\) in Eq. (3.1) must be replaced with appropriate values. For example, the permissible roll angle associated with nacelle-to-ground clearance is determined with the following values: \(\Gamma\) measured from the horizon to the bottom of the nacelle in the front view, \(\Lambda\) measured from the chosen landing gear location to the engine in the top view, and \(s\) the distance between the engines.

3.3.1.1. Pitch Angle Required for Liftoff

The takeoff rotation angle is prescribed in preliminary design, and then estimated. The final values for \(\theta\) and \(\phi\) are found as the detailed performance characteristics of the aircraft become available. The pitch angle at liftoff (\(\theta_{LOF}\)) is calculated using the expression [5, p. 350]

\[
\theta_{LOF} = \alpha_{LOF} + \frac{d\theta}{dt} \left( \frac{2l_1}{V_{LOF}} + \sqrt{\frac{h_2}{g \frac{dC_L}{d\alpha}}} \right)
\]  

(3.2)

where \(\alpha_{LOF}\) is the highest angle of attack anticipated for normal operational use, \(V_{LOF}\) is the liftoff speed, \(g\) is the gravitational acceleration \(C_{L,LOF}\) is the lift coefficient, and \(dC_L/d\alpha\) is the lift-curve slope. As shown in Fig. 3.1, the dimension of \(l_1\) and \(l_2\) are defined by the line connecting the tire-ground contact point upon touchdown and the location of the tail bumper, if one is present. For large transports, the typical value for the rate of rotation \((d\theta/dt)\) is taken as four degrees per second [5].
Figure 3.1 Geometric definitions in relation to the pitch and roll angles [5]
The detailed aerodynamic data required to use Eq. (3.2) is not always available at the conceptual design stage. In most aircraft the aft-body and/or tail bumper is designed such that the aircraft cannot rotate by more than a specified number of degrees at liftoff. Typically, the value is between 12 and 15 degrees [2]. In addition to the tail scrape problem, the aircraft cg cannot rotate over and aft of the location of the main assembly, a phenomenon known as tail tipping and is critical during landing.

3.3.1.2. Pitch and Roll Angles During Landing

With the flaps in the fully-deflected position, the critical angle of attack of the wing during landing is smaller than in takeoff. Consequently, the pitch angle during landing is generally less than that during takeoff. In the absence of detailed information, the pitch angle on touchdown ($\theta_{TD}$) may be assumed equal to $\theta_{LOF}$. As for the roll angle upon touchdown, an upper limit of between five [20] and eight [5] degrees is generally applied to large transport aircraft.

3.3.2. Stability at Touchdown and During Taxiing

Static stability of an aircraft at touchdown and during taxiing can be determined by examining the location of the applied forces and the triangle formed by connecting the attachment locations of the nose and main assemblies. Whenever the resultant of air and mass forces intersects the ground at a point outside this triangle, the ground will not be able to exert a reaction force which prevents the aircraft from falling over. As a result, the aircraft will cant over about the side of the triangle that is closest to the resultant force/ground intersect.

Assuming first that the location of the nose assembly is fixed, the lower limit of the track of the landing gear, identified as constraint I in Fig. 3.2, is defined by the line passing through the center of the nose assembly and tangential to the circle with a radius of 0.54 times the height of the aircraft cg ($h_{cg}$) from the static groundline, centered at the fore-most cg location [5]. The constant 0.54 is based on static and dynamic instability considerations at touchdown and during taxiing. Conversely, if the location of the main assembly is
assumed to be fixed, the aft-most limit of the nose assembly mounting location, identified as constraint II in Fig. 3.2, is defined as the intersection of the aircraft centerline and the line that passes through the center of the main assembly, tangential to the circle with a radius of 0.54 times of the height of aircraft cg.

3.3.2.1. Condition at Touchdown

The most unfavorable condition at touchdown would be a landing with the aircraft cg at its aft-most and highest location, which can lead to the tail scrape and tail tipping phenomenon mentioned previously. Assuming there are no retarding forces, i.e., spin-up load, a vertical force acting at a distance behind the aircraft cg is needed to produce a moment that will pitch the nose downward. Thus, the minimum allowable offset between the aft-most cg and the main assembly mounting locations, identified as constraint III in Fig. 3.2, is determined using the following expression [5, p. 352]

\[ l_m \geq (h_{cg} + e_s) \tan \theta_{TD} \]  

(3.3)
where $e_s$ is the total static deflection of the shock strut and tire, and $\theta_{rd}$ is the pitch angle at touchdown. Note that the offset distance is dependent on the value of the pitch angle, whose value is similar to the pitch angle at liftoff, i.e., between 12 and 15 degrees. For a low-wing passenger aircraft, $h_{qs}$ can be approximated assuming a full load of passengers and no wing fuel [2]. This generally results in a vertical $cg$ position at the main passenger-deck level.

### 3.3.2.2. Sideways Turnover Angle

Forces acting sideways on the airplane in cross-wind landing condition or a high-speed turn during taxiing could cause the aircraft to turnover on its side. It is thus desirable to keep the turnover angle ($\psi$) as small as possible. The angle is determined using the expression [2, p. 38]

$$
\tan \psi = \frac{h_{cg}}{l_n \sin \delta} 
$$

where

$$
\tan \delta = \frac{t}{2(l_m + l_n)}
$$

and $\delta$ is defined as the angle between the aircraft centerline and the line connecting the center of the nose and main assembly. The dimensions used in the above equations are given in Fig. 3.3. For land-based aircraft, either the maximum allowable overturn angle of 63 degrees [2] or the stability considerations at takeoff and touchdown and during taxiing, whichever is the most critical, determines the lower limit for the track of the main assembly.

### 3.3.3. Braking and Steering Qualities

The nose assembly is located as far forward as possible to maximize the flotation and stability characteristics of the aircraft. However, a proper balance in terms of load distribution between the nose and main assembly must be maintained. When the load on the nose wheel is less than about eight percent of the maximum takeoff weight (MTOW),
controllability on the ground will become marginal, particularly in cross-wind conditions. This value also allows for fuselage length increase with aircraft growth. On the other hand, when the static load on the nose wheel exceeds about 15 percent of the MTOW, braking quality will suffer, the dynamic braking load on the nose assembly may become excessive, and a greater effort may be required for steering [5]. Note that these figures should be looked upon as recommendations instead of requirements.

3.3.4. Gear Length

Landing gear struts should be of sufficient length such that adequate clearance between the runway and all other parts of the aircraft, e.g., the aft-body, wingtips, and engine nacelles, is maintained when the aircraft is on the ground. For a low-wing aircraft with wing-mounted engines, the above requirement proves to be one of the most challenging design issues in terms of permissible roll angle at touchdown. Although engine nacelle-to-ground clearance has not been explicitly defined, a similar requirement for propellers was

* There are exceptions. The DC-9-50 has 3% on the nosewheel.
specified in FAR Part 25 and can be used as an absolute minimum: a seven-inch clearance between the propellers and the ground in level takeoff or taxiing attitude, whichever is most critical. To date, the smallest offset on jet transports is found on the Boeing Model 747/GE90 testbed, where the GE90 engine nacelle clears the ground by a mere 13-inch clearance [21]. As for operational aircraft, the Boeing Model 737-300, -400, and -500 exhibit a 15-inch nacelle-to-ground clearance [22]. The length of the nose wheel strut is generally based on the requirement that the fuselage should be horizontal or tilted slightly nose-down when the aircraft is on the ground.

Besides the clearance considerations, allowance must also be considered for future stretching of the aircraft, which generally involves adding plugs forward and aft of the wing spars. Provided that the attitude of the aircraft will remain the same, the increase in the aft fuselage length would thus reduce the maximum permissible takeoff rotation angle, which can result in costly modifications and thus effectively rule out future growth options. Boeing abandoned further stretches of the Model 727 partially because of the difficulties encountered while attempting to maintain an adequate tail scrape angle, whereas Douglas was able to reduce the required tail scrape angle on the MD-11 by only structural changes, increasing the wing incidence by three degrees over that of its 22-foot shorter DC-10-30 forebear.

3.3.5. Landing Gear Attachment

From considerations of surrounding structure, the nose and main assembly are located such that the landing and ground loads can be transmitted most effectively, while at the same time still comply with the stability and controllability considerations. For a wing-mounted assembly, the trunnion is generally attached to the rear wing spar and the landing gear beam and the loads are transmitted directly to the primary wing-fuselage bulkheads. With the inclusion of fuselage-mounted assemblies in the multiple main-strut configurations, a secondary frame would then be added at a distance behind the rear wing-spar, where loads are transmitted forward to the primary wing-fuselage bulkhead through the keel and by shear in the fuselage skin. As for the nose assembly, structural considerations may be conclusive in deciding the mounting location, i.e., at the proximity
of forward cabin bulkhead to minimize weight penalty due to local structural reinforcement.

3.4. Ground Operation Characteristics

Besides ground stability and controllability considerations, the high costs associated with airside infrastructure improvements, e.g., runway and taxiway extensions and pavement reinforcements, have made airfield compatibility issues one of the primary considerations in the design of the landing gear [23]. In particular, the aircraft must be able to maneuver within a pre-defined space as it taxes between the runway and passenger terminal. For large aircraft, this requirement effectively places an upper limit on the dimension of the wheelbase and track.

3.4.1. Aircraft Turning Radii

As shown in Fig. 3.4, turning radii are defined as the distances between the center of rotation and various parts of the aircraft. The center of rotation is located at the intersection of the lines extending from the axes of the nose and main assemblies. For aircraft with more than two main struts, the line extending from the main assembly group is located midway between the fore and aft gears. The turning radii are a function of nose gear steering angle (β); the greater the angle, the smaller the radii. The upper limit for this angle is determined by the methods available to provide the steering action, which generally limits the angle to ±60 degrees [2].

The turning radius corresponding to an 180-degree turn \( r_{180°\text{turn}} \) as identified in Fig. 3.4 is determined using the expression

\[
r_{180°\text{turn}} = b \tan(90 - \beta) + \frac{t}{2}
\]

(3.6)

where \( b \) and \( t \) are the wheelbase and track, respectively. Given the aircraft design group classification as listed in Table 3.1, the minimum turning diameter, i.e., twice of the 180-degree turn radius, should be less than the corresponding runway pavement width.

* This value may be low. The B737 has ±75°, the DC-8 ±74 1/2 °, the DC-9 ±80°, and the B767 ±65°.
Table 3.1 FAA airplane design group classification for geometric design for airports [7]

<table>
<thead>
<tr>
<th>Airplane design group</th>
<th>Wingspan, ft</th>
<th>Runway width, ft</th>
</tr>
</thead>
<tbody>
<tr>
<td>III</td>
<td>79.0 &lt; s &lt; 118.0</td>
<td>100.0</td>
</tr>
<tr>
<td>IV</td>
<td>118.0 &lt; s &lt; 171.0</td>
<td>150.0</td>
</tr>
<tr>
<td>V</td>
<td>171.0 &lt; s &lt; 197.0</td>
<td>150.0</td>
</tr>
<tr>
<td>VI</td>
<td>197.0 &lt; s &lt; 262.0</td>
<td>200.0</td>
</tr>
</tbody>
</table>

With the greater wheelbase and track dimensions as exhibited by large aircraft, the 180-degree turn maneuver can no longer be achieved with the conventional nose-steering scheme alone. As a result, combined nose and main assembly steering systems have been introduced on the newer large aircraft, e.g., Boeing Models 747 and 777, to reduce the turning radii. Other advantages provided by this feature include reduced tire wear and scuffing of the pavement surface in a sharp turn. Note that at the conceptual design phase of an aircraft, Eq. (3.6) is sufficient in producing a first-cut estimate. The resulting turning radii, which are based on nose-steering scheme, are slightly larger than the ones corresponding to combined nose and main assembly steering scheme, and thus provide a built-in safety margin.

* However, the first generation DC-8 also incorporated this feature.
3.4.2. Centerline-guidance Taxiing

The size of the fillets at runway and taxiway intersections depend not only on wheelbase, radius of centerline curve, width of taxiway, and total change in direction, but also on the path that the aircraft follows. There are two options in which an aircraft can be maneuvered on a turn: one is to establish the centerline of the taxiway as the path of the nose gear; the other is to assume that the nose gear follows a path offset outward of the centerline during the turn. The former is selected as the critical design case since it is the most demanding of the two in terms of piloting skill, i.e., difficult to keep the nose wheel, which is below and behind the pilot's field of view, on the centerline while taxiing, and thus requires a greater area of pavement during the maneuver as safety margin.

As shown in Fig. 3.5, the maximum castor angle ($\phi$), i.e., the angle formed between the tangent to the centerline and the longitudinal axis of the aircraft, will occur at the end of the turn, where the nose wheel is at the point of tangency. The angle is approximated by [7, p. 318]
\[
\sin \phi = \frac{b}{R}
\] (3.7)

where \( R \) is the radius of centerline curve. A re-check should be made at this point to make sure that the design castor angle is within the permissible range of the steering angle.

For a given wheelbase and track dimension, the required fillet radius \((F)\) is calculated using the expression [5, p. 318]

\[
F = \sqrt{R^2 + b^2 - 2R b \sin \phi} - \frac{t}{2} - S
\] (3.8)

where \( S \) is the minimum distance required between the edge of the outboard tire and the edge of the pavement. Given the aircraft design group classification number as determined from Table 3.1 and the corresponding FAA design values as presented in Table 3.2, the upper limit for the wheelbase and track of the aircraft can be determined using Eqs (3.7) and (3.8).

<table>
<thead>
<tr>
<th>Table 3.2 FAA recommended taxiway exit geometry [7]</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Group III</strong></td>
</tr>
<tr>
<td>Centerline radius, ft</td>
</tr>
<tr>
<td>Fillet radius, ft</td>
</tr>
<tr>
<td>Safety margin, ft</td>
</tr>
</tbody>
</table>
3.5. Landing Gear Disposition Constraints

Landing gear location constraints as discussed in the above sections are superimposed on the three-view of a notional aircraft for illustrative purposes. As shown in Fig. 3.6a, the main assembly must be located such that when the shock strut is at the fully-extended position, the tire-ground contact point is below constraints II and IV in the vertical direction and outboard of constraint I in the lateral direction. In the top view as shown in Fig. 3.6b, the main assembly must also be located aft of constraint IV in the longitudinal direction and outboard of constraint V in the lateral direction. As for the nose assembly, it must be
located between constraints I and II in the longitudinal direction. And finally, as shown in Fig. 3.6c, the fully-extended tire-ground contact point is below constraint II in the vertical direction and aft of constraint II in the lateral direction.

---

**Figure 3.6 Landing gear attachment location constraints [5]**

---
c) Side view

Figure 3.6 Landing gear attachment location constraints (cont'd)
Chapter 4  Tires, Wheels, and Brakes

4.1. Introduction

The number of tires required for a given aircraft design gross weight is largely determined by the flotation characteristics, which will be discussed in detail in Chapter Seven. Assuming that the number and distribution pattern of the tires is already known, this chapter provides guidelines to the selection of the tires, wheels, and brakes that will meet the performance and safety requirements [2 and 20].

As a part of the landing gear configuration definition process, tires, wheels, and brakes selection algorithms were developed based on the procedure as discussed in this chapter. Specified selection criterion, e.g., minimum size, weight, or pressure, are used to select suitable tires and wheels from manufacturer’s catalog [24] and industry standards [25], while statistical database was used to size the brakes as required to meet the braking requirements.

4.2. Type, Size and Inflation Pressure of the Tire

The tire selection process involves listing all candidates that meet the performance requirements. A list of tires and wheels used on commercial transports can be found in Appendix D. The primary consideration is the load-carrying capacity of the tire during the speed regime normally applicable for landing or takeoff cycles. In addition, the number of plys and type of construction, which determines the weight of the tire and its operational life, is important from an economic standpoint. Other considerations include the inflation pressure of the tire and the size of the wheel. The former must be chosen in accordance with the bearing capacity of the airfield from which the aircraft is designed to operate from, whereas the latter must have sufficient space to house the brake assembly.
4.2.1. Basic Tire Constructions

Radial tires have gained growing acceptance since their introduction despite a somewhat cautious approach at the beginning, which is attributed to lack of applicable standards, concerns about the mixability with bias tires, and retreadability of refurbished tires. Intermixing of radial and bias tires, or even with radial tires of different construction, is possible only if the loading is no more uneven than currently encountered with mixing of bias tires only. As for retreadability, it should be noted that multiple retreating is not necessarily a benefit to the airlines; instead, it could be an indication of low tire performance in terms of tread wear. Thus, the concern here is not as much how often the tire can be retreated, but how to extend the average total carcass life.

In radial construction, shear stresses in the rubber matrix are minimized and loads are efficiently distributed throughout the tire. Even if the same basic materials used in bias tires are used in the radials, the amount of material required for a particular application can be reduced. As a result, weight savings of up to 20 percent have been realized [26]. In addition, minimized slippage between the tire and the contact surface and the near optimal tuning of belt stiffness that comes with the radial construction all contributed to improved wear performance. In fact, some radial tires currently achieve twice as many landings per tread as conventional bias tires [26].

Operational experience has also shown that radial tires offer a greater overload bearing capacity and withstand under-inflation better. An approximately 10-percent increase in the footprint area improves the flotation characteristics and reduces hydroplaning on wet runways [26]. In addition, radial tires do not fail as suddenly as bias tires do. Warning signs such as external deformation and out-of-roundness exhibited prior to catastrophic failure provide indications of a potential blowout to maintenance personnel, and thus enhance operational safety.
4.2.2. Size of the Tire

The choice of the main wheel tires is made on the basis of the static loading case. The total main gear load \( F_m \) is calculated assuming that the aircraft is taxiing at low speed without braking. As shown in Figure 4.1, equilibrium gives [5, p. 356]

\[
F_m = \frac{l_m}{l_m + l_n}W
\]  

(4.1)

where \( W \) is the weight of the aircraft and \( l_m \) and \( l_n \) are the distance measured from the aircraft cg to the main and nose gear, respectively. The design condition occurs at MTOW with the aircraft cg at its aft limit. For single axle configurations, the total load on the strut is divided equally over the tires, whereas in tandem configurations, the load per wheel depends on the location of the pivot point; to reduce overloading of the front wheels during braking, the pivot is usually positioned such that the distance between it and the front and rear wheel axles is about 55 and 45 percent of the truck beam, respectively [5].

The choice of the nose wheel tires is based on the nose wheel load \( F_n \) during braking at maximum effort, i.e., the steady braked load. Using the symbols shown in Fig. 4.1, the total nose gear load under constant deceleration is calculated using [5, p. 358]

\[
F_n = \frac{l_m}{l_m + l_n}(W - L) + \frac{h_{cg}}{l_m + l_n}\left(\frac{a_x}{g}W - D + T\right)
\]  

(4.2)

where \( L \) is the lift, \( D \) is the drag, \( T \) is the thrust, and \( h_{cg} \) is the height of aircraft cg from the static groundline. Typical values for \( a_x/g \) on dry concrete vary from 0.35 for a simple brake system to 0.45 for an automatic brake pressure control system [5]. As both \( D \) and \( L \) are positive, the maximum nose gear load occurs at low speed. Reverse thrust decreases the nose gear load and hence the condition \( T = 0 \) results in the maximum value [5, p. 359]

\[
F_n = \frac{l_m + h_{cg}(a_x/g)}{l_m + l_n}W
\]  

(4.3)

The design condition occurs at MTOW with the aircraft cg at its forward limit.
To ensure that the rated loads will not be exceeded in the static and braking conditions, a seven percent safety factor is used in the calculation of the applied loads [2]. In addition, to avoid costly redesign as the aircraft weight fluctuates during the design phase, and to accommodate future weight increases due to anticipated aircraft growth, the calculated loads are factored upward by another 25 percent prior to tire selection [2].

4.2.3. Inflation Pressure

Provided that the wheel load and configuration of the landing gear remain unchanged, the weight and volume of the tire will decrease with an increase in inflation pressure. From the flotation standpoint, a decrease in the tire contact area will induce a higher bearing stress on the pavement, thus eliminates certain airports from the aircraft's operational bases. Braking will also become less effective due to a reduction in the frictional force between the tires and the ground. In addition, the decrease in the size of the tire, and hence the size of the wheel, could pose a problem if internal brakes are to be fitted inside the wheel rims. The arguments against higher pressure are of such a nature that commercial operators generally prefer the lower pressures in order to maximize tire life and minimize runway stress [26].
4.3. Wheel Design

The design of the aircraft wheel is influenced primarily by its requirement to accommodate the selected tire, to be large enough to house the brake, and to accomplish the above tasks with minimum weight and maximum life. As shown in Figure 4.2, two basic configurations of wheel design are currently available: A-frame and bowl-type [27]. The former is structurally the most efficient and therefore the lightest that can be achieved. However, this design has a limited space for housing the brake as compared to the bowl-type design. Consequently, as the braking energy requirement increases with aircraft weight and hence the size of the heat sink required, it might be necessary to resort to a bowl-type design even though it has a weight penalty [27].

![Diagram of wheel design]

Figure 4.2 Basic configuration of wheel design [27]

Continued heavy dependence on forged aluminum alloy wheels is foreseen by industry, whereas steel and magnesium alloy wheels are no longer given serious consideration due to weight and corrosion problems, respectively [28]. Although practicable, titanium wheels are still quite expensive. Most of the premium for titanium wheels results from the expense for the forging process, which could be 10 to 11 times
those of aluminum alloy [28]. In addition, current titanium forging tolerances have yet to reach the precision obtainable for aluminum material, thus machining of all surfaces is required to control weight and obtain the desired form.

Based on statistical data, the wheel assembly weight is determined as a function of the rated per wheel static load \( F \) and average tire outer diameter \( D \) [2, p. 145]

\[
f_w = \frac{FD}{1000}
\]

(4.4)

Given the type of material to be used, the wheel assembly unit weight is obtained from Figure 4.3 with the weight factor \( f_w \) as determined from Eq. (4.4).

Figure 4.3  Aircraft wheel assembly weight [2]

4.4. Brake Design

Besides the primary task of stopping the aircraft, brakes are used to control speed while taxiing, to steer the aircraft through differential action, and to hold the aircraft stationery when parked and during engine run-up. Since the heat sinks account for a significant fraction of total landing gear weight, there is a continual effort to reduce their weight through the application of advanced materials, namely, carbon [19].
4.4.1. Heat Sink Material

Material characteristics of steel and carbon are compared in Table 4.1. As shown in the table, carbon’s high specific heat and thermal conductivity make it highly desirable as a heat absorber. The former ensures a reduction in brake weight, while the latter ensures that the heat transfer throughout the heat sink occurs more uniformly and at a faster rate. In addition, carbon retains much of its specific strength, which is defined as the ultimate tensile strength divided by density, at high temperature while steel loses almost all of its strength.

Table 4.1 Heat sink materials comparison [2]

<table>
<thead>
<tr>
<th>Property</th>
<th>Steel</th>
<th>Carbon</th>
<th>Desired</th>
</tr>
</thead>
<tbody>
<tr>
<td>Density, lb/in$^3$</td>
<td>0.283</td>
<td>0.061</td>
<td>High</td>
</tr>
<tr>
<td>Specific heat at 500°F, Btu/lb•°F</td>
<td>0.13</td>
<td>0.31</td>
<td>High</td>
</tr>
<tr>
<td>Thermal conductivity at 500°F, Btu/h•ft$^2$•°F</td>
<td>24.0</td>
<td>100.0</td>
<td>High</td>
</tr>
<tr>
<td>Thermal expansion at 500°F, 1.0E-6 in•°F/in</td>
<td>8.4</td>
<td>1.5</td>
<td>Low</td>
</tr>
<tr>
<td>Thermal shock resistance index, x105</td>
<td>5.5</td>
<td>141.0</td>
<td>High</td>
</tr>
<tr>
<td>Temperature limit, °F</td>
<td>2,100</td>
<td>4,000</td>
<td>High</td>
</tr>
</tbody>
</table>

Long service life and low maintenance requirements for carbon brakes prove to be another plus from an economic standpoint. It was estimated that carbon would permit up to five to six times more landings as compared to steel between refurbishment and would require fewer man-hours for overhaul [27]. To illustrate the economic advantage of using carbon brakes, it was estimated that a total weight saving of 1,200 pounds could be achieved on the Concorde using carbon brakes. This is equivalent to five percent of its estimated transatlantic payload [29].

The primary drawback of carbon brakes is that a greater volume is required to absorb the same amount of energy in comparison to steel brakes. Some problems with carbon brakes include sudden loss of strength due to oxidation of the carbon, temporary loss of braking due to moisture contamination, and high initial cost. However, these issues have
largely been resolved in favor of the performance and economic aspects of carbon heat sinks [16]. In fact, advanced transports such as the Boeing Model 777 and the emerging ultra-high-capacity aircraft all feature carbon brakes.

4.4.2. Brake Sizing

The primary consideration in brake development is the size and weight of the brake required to meet the kinetic energy generated under the design landing weight, maximum landing weight, and rejected takeoff (RTO) conditions. Brake capacity requirements for these braking conditions are listed in Table 4.2.

<table>
<thead>
<tr>
<th>Specifications</th>
<th>100 stops at average of 10 ft/s² deceleration</th>
</tr>
</thead>
<tbody>
<tr>
<td>Design landing weight</td>
<td></td>
</tr>
<tr>
<td>Maximum landing weight</td>
<td>5 stops at average of 10 ft/s² deceleration</td>
</tr>
<tr>
<td>Rejected takeoff</td>
<td>1 stop at average of 6 ft/s² deceleration</td>
</tr>
</tbody>
</table>

The total kinetic energy is determined using the expression [20]

\[
KE = 0.0443WV^2
\]  

(4.5)

where \( V \) is the power-off stalling speed in knots. Assuming that the power-off stalling speed is 1.2 times of the stalling speed \( (V_s) \), it can be approximated using the expression [5, p. 577]

\[
V = 1.2V_s = 1.2\sqrt{\frac{2W}{1.13\rho SC_{L,max}}}
\]  

(4.6)

where \( \rho \) is the standard sea-level air density, \( S \) is the reference wing area, and \( C_{L,max} \) is the maximum wing lift coefficient. The constant 1.13 takes into account the speed loss in the FAA stall maneuver [5]. As illustrated above, the kinetic energy absorption requirements increase as the square of the velocity and hence the landing speed is significant.
The procedure used to size a steel brake is given here for illustrative purposes. Similar data for carbon are not available, but scaling factors of 1.28 and 0.40 can be used to relate the steel volumes and weights, respectively, to those values for carbon [30]. Kinetic energy levels expected under the normal landing weight, maximum landing weight, and RTO conditions are first calculated using Eq. (4.5) and the appropriate aircraft weights. Brake assembly weights \( W_{brake} \) corresponding to each kinetic energy level are obtained from Figure 4.4 and averaged to arrive at a compromise value. The required heat sink volume \( V_{brake} \) is then approximated using the expression

\[
V_{brake} = 3.3W_{brake} - 84.2
\]  

(8.7)

where the constant coefficients are determined using linear regression analysis on statistical database[2].

![Figure 4.4 Brake assembly weight vs. kinetic energy level [2]](image)

Given the tire wheel diameter as determined during the tire selection process, heat sink inner and outer diameters and the volume per inch width constant are selected from Table 4.3. Dividing the total volume by the constant then gives the necessary heat sink width.
The envelope for the heat sink and torque plate carrier is established by adding 0.75 inch on the inside diameter and the end facing the wheel centerline. Finally, the piston housing envelope is approximated by adding two inches on the actuation side of the heat sink as shown in Figure 4.5 [2].

<table>
<thead>
<tr>
<th>Rim dia., in</th>
<th>Inner dia., in</th>
<th>Outer dia., in</th>
<th>Volume/inch width, in²</th>
</tr>
</thead>
<tbody>
<tr>
<td>14.0</td>
<td>7.375</td>
<td>12.000</td>
<td>70.4</td>
</tr>
<tr>
<td>15.0</td>
<td>8.125</td>
<td>13.000</td>
<td>80.9</td>
</tr>
<tr>
<td>16.0</td>
<td>8.750</td>
<td>13.750</td>
<td>88.4</td>
</tr>
<tr>
<td>17.0</td>
<td>9.500</td>
<td>14.750</td>
<td>100.0</td>
</tr>
<tr>
<td>18.0</td>
<td>10.125</td>
<td>15.750</td>
<td>114.3</td>
</tr>
<tr>
<td>19.0</td>
<td>10.750</td>
<td>16.500</td>
<td>123.1</td>
</tr>
<tr>
<td>20.0</td>
<td>11.500</td>
<td>17.500</td>
<td>136.7</td>
</tr>
<tr>
<td>21.0</td>
<td>12.250</td>
<td>18.500</td>
<td>150.9</td>
</tr>
<tr>
<td>22.0</td>
<td>12.875</td>
<td>19.500</td>
<td>168.5</td>
</tr>
<tr>
<td>23.0</td>
<td>12.750</td>
<td>20.375</td>
<td>176.3</td>
</tr>
<tr>
<td>24.0</td>
<td>14.375</td>
<td>21.375</td>
<td>195.2</td>
</tr>
<tr>
<td>25.0</td>
<td>15.125</td>
<td>22.375</td>
<td>212.1</td>
</tr>
</tbody>
</table>

Figure 4.5 Key elements of carbon brakes. [See [2], pg 138 for more detailed view.]
Chapter 5  Shock Absorber Design

5.1. Introduction

The basic function of the shock absorber is to absorb and dissipate the impact kinetic energy to the extent that accelerations imposed upon the airframe are reduced to a tolerable level [2 and 20]. Existing shock absorbers can be divided into two classes based on the type of the spring being used: those using a solid spring made of steel or rubber and those using a fluid spring with gas or oil, or a mixture of the two that is generally referred to as oleo-pneumatic. The high gear and weight efficiencies associated with the oleo-pneumatic shock absorber make it the preferred design for commercial transports [2].

Based on the analysis procedure as outlined in this chapter, algorithms were developed to determine the required stroke and piston length to meet the given design conditions, as well as the energy absorption capacity of the shock absorber.

5.2. Oleo-Pneumatic Shock Strut Design

The basic weight support function of the oleo-pneumatic shock struts, which have a high efficiency under dynamic conditions both in terms of energy absorption and dissipation, is provided by a compressed cylinder of air and oil. A single-acting shock absorber, which is the most commonly used design for commercial transports, is shown in Fig. 5.1. This type of shock strut absorbs energy by first forcing a chamber of oil against a chamber of dry air or nitrogen and then compressing the gas and oil. During the compression process, the oil and gas either remain separated or are mixed depending on the type of design. After the initial impact, energy is dissipated as the air pressure forces the oil back into its chamber through recoil orifices.

Although the compression orifice could be merely a hole in the orifice plate, most designs have a metering pin extending through it, and by varying the pin diameter the orifice area is varied. This variation is adjusted so that the strut load is fairly constant under dynamic loading. If this can be made constant, the gear efficiency would be 100 percent. In practice, this is never obtained and efficiencies of 80 to 90 percent are more usual [4]. Since only the efficiency factor is of interest in the conceptual design phase, no additional discussion on the design of the metering pin will be provided.
5.2.1. Stroke Calculation

The first step in calculating the stroke \( S \) is to select the design reaction factor \( N \), sometimes called the landing load factor. This factor should not be confused with the aircraft load factor, which results from maneuvers or atmospheric disturbances. For a transport-type aircraft the landing load factor varies from 0.7 to 1.5, with 1.2 being the most widely used value [2].

Sink speed \( V_s \) is usually legislated by the procuring authority and/or the regulations pertaining to a particular category of aircraft. The FAA requires that a transport-type aircraft be able to withstand the shock of landing at 10 ft/s at the design landing weight and 6 ft/s at maximum gross weight [19]. In practice, sink speeds of this magnitude rarely occur due to ground effects and flare-out of the aircraft prior to touchdown.

The total energy \( E \) of the aircraft at the instant of touchdown, which consists of kinetic and potential energy, is approximated using the expression [2, p. 35]
\[ E = \frac{WV^2}{2g} + (W - L)(S + S_t) \] (5.1)

where \( W \) is the aircraft weight, \( V \) is the sink speed, \( g \) is the gravitational acceleration, \( L \) is the wing lift, and \( S_t \) is the tire deflection. \( S \) is shock absorber stroke, which is the value we are trying to find. Given that the kinetic energy capacity of the shock absorber and tire must be equal to the total energy, Eq. (5.1) becomes \[2, p. 35\]

\[ \eta_s SNW + \eta_t S_t NW = \frac{WV^2}{2g} + (W - L)(S + S_t) \] (5.2)

where \( \eta_s \) and \( \eta_t \) are the shock absorber and tire absorber efficiency factors, respectively. The former is generally assumed to be 0.47 and the latter 0.8 for an oleo-pneumatic strut \[2\]. To maintain an adequate safety margin, an extra one inch of stroke is usually added to the calculated stroke.

5.2.2. Compression Ratios

Compression ratios are the ratios of the pressure under one condition divided by the pressure under another condition, e.g., fully compressed to static. Two compression ratios are normally considered: static to fully extended and fully compressed to static. For transport-type aircraft, where floor height variation is important, a ratio of 4:1 for the static to extended case and 3:1 for the compressed to static case would be satisfactory \[2\]. Assuming a static pressure \((P_2)\) of 1,500 psi, which enables standard compressors to be used for servicing and provides enough margin to allow for aircraft growth, pressures at the extended \((P_1)\) and compressed \((P_3)\) positions are calculated using the compression ratios given above. Note that the piston area \((A)\), and subsequently the displacement volume \((d)\), are both a function of the static pressure, that is

\[ A = \frac{F}{P_2} \] (5.3)

and

\[ d = SA \] (5.4)

where \( F \) is the maximum static load per strut.
5.2.3. The Load-stroke Curve

The energy absorbed by the strut during its stroke is obtained by integrating the area beneath the load-stroke curve, which relates the magnitude of the applied ground loads to the stroke traversed. Standard notation for shock strut sizing uses the subscript 1 to denote the fully extended position, 2 to denote the static position, and 3 to denote the compressed position. To accommodate excess energy produced in a heavy or semi-crash landing, shock absorbers are designed such that the piston is not fully bottomed even at the compressed position, \( i.e., V_3 \neq 0 \). The reserve air volume, which is assumed to be 10 percent of the displacement [2], allows the shock strut at a predetermined load to move through extra travel, absorbing the excess energy by the work done. Hence, the air volume at the fully-extended position is approximated as [2, p. 100]

\[
V_1 = V_3 + d
\]  

(5.5)

Pressures between the extended and static positions are defined by the isothermal compression curve, which is representative of normal ground handling activity [2, p. 100]

\[
R V_1 = P_x V_x = \text{const}
\]  

(5.6)

Given the relationships of Eqs (5.4) and (5.5), the pressure at stroke \( X \) is obtained using the expression [2, p. 100]

\[
P_x = \frac{P_1 V_1}{V_x} = \frac{R(V_3 + d)}{V_1 - XA} \quad S_{\text{extend}} < X < S_{\text{static}}
\]  

(5.7)

Pressures obtained using Eq. (5.7) are then multiplied by the piston area to arrive at the design loads as shown on the load-stroke curve.

A polytropic, \( i.e., \) real-gas, compression curve should be considered for pressures between the static and compressed positions. It is representative of dynamic compression cases such as landing impact and bump traversal and is based upon \( P V^n \) being constant [2], hence

\[
P_x = P_2 \left( \frac{V_2}{V_1 - XA} \right)^n \quad S_{\text{static}} < X < S_{\text{compress}}
\]  

(5.8)

The constant \( n \) can either be 1.35 or 1.1; the former is used when the gas and oil are separated and the latter when they are mixed during compression. The distance from the
static to the fully compressed position is largely a matter of choice. Statistical data indicate that transport-type aircraft typically have further compression beyond the static position of about 16 percent [2] of the total stroke, a figure which tends to give a hard ride while taxiing. However, with the static position being so far up the load-stroke curve, where a large amount of energy is absorbed with a relatively small stroke travel, aircraft weight variations do not result in substantial gear deflections. That is, the built-in margin minimizes the need of redesigning the baseline shock strut for uses on future growth versions of the aircraft. Again, the pressures obtained using Eq. (5.9) are multiplied by the piston area to arrive at the design loads. At this point the values of $P_1$ and $P_3$ should be checked to ensure that the former is greater than 60 psi to avoid sticking due to friction between the piston and the cylinder wall, while the latter is less than 6,000 psi to prevent seal leakage [2].

5.2.4. Internal Cylinder Length

As specified by MIL-L-8552, the distance between the outer ends of the bearings shall be not less than 2.75 times the internal cylinder/piston outside diameter ($D$). Thus the minimum piston length is given by [2, p. 111]

$$L_{pist} = S + 2.75D$$

(5.9)

where

$$D = \sqrt{\frac{4A}{\pi}}$$

(5.10)

5.2.5. Sample Calculation

The load-stroke curve of a notional single-acting shock absorber is generated for illustrative purposes. Based on the design requirements as stated in Table 5.1, Eq. (5.3) gives a piston cross-sectional area of 33.3 in$^2$, while Eq. (5.4) places the total displacement at 666.7 in$^3$. Using the 16 percent extension figure, the static position at which the gas law switches from isothermal to polytropic gas law is estimated to be 3.2 inches from the fully compressed position, i.e., $X$ at 16.8 inches. Loads corresponding to isothermal and polytropic compression were determined using Eqs (5.7) and (5.8), respectively, and presented in Table 5.2. The corresponding load-stroke curves are shown in Fig. 5.2.
Table 5.1 Shock absorber sizing parameters

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Design value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Total stroke</td>
<td>20.0 in</td>
</tr>
<tr>
<td>Static position</td>
<td>16 percent of total stroke</td>
</tr>
<tr>
<td>Static load</td>
<td>50,000 lb</td>
</tr>
<tr>
<td>Static pressure</td>
<td>1,500 lb</td>
</tr>
<tr>
<td>Compression ratio</td>
<td>4:1 static to extended</td>
</tr>
<tr>
<td></td>
<td>3:1 compressed to static</td>
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Table 5.2 Calculations of isothermal and polytropic compression

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<th>$X$, in.</th>
<th>$V$, in$^3$</th>
<th>$P_{iso}$, psi</th>
<th>$P_{poly}$, psi</th>
<th>$P_{comb}$, psi</th>
<th>$F_{comb}$, lb</th>
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<td>0.0</td>
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<td>459.2</td>
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<td>578.9</td>
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<td>694.8</td>
<td>592.1</td>
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<td>858.2</td>
<td>692.4</td>
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<td>833.3</td>
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<td>14.0</td>
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<td>2234.2</td>
<td>1406.5</td>
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<td>73820.0</td>
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<td>4500.6</td>
<td>10740.1</td>
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<td>188176.7</td>
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</table>

Figure 5.2 The load-stroke curve
Chapter 6  Kinematics

6.1. Introduction

Kinematics is the term applied to the design and analysis of those parts used to retract and extend the gear [2]. Particular attention is given to the determination of the geometry of the deployed and retracted positions of the landing gear, as well as the swept volume taken up during deployment/retraction. The objective is to develop a simple deployment/retraction scheme that takes up the least amount of stowage volume, while at the same time avoiding interference between the landing gear and surrounding structures. The simplicity requirement arises primarily from economic considerations. As shown from operational experience, complexity, in the forms of increased part-count and maintenance down-time, drives up the overall cost faster than weight [5]. However, interference problems may lead to a more complex system to retract and store the gear within the allocated stowage volume.

Based on the analysis as outlined in this chapter, algorithms were developed to establish the alignment of the pivot axis which permits the deployment/retraction of the landing gear to be accomplished in the most effective manner, as well as to determine the retracted position of the assemblies such that stowage boundary violations and structure interference can be identified.

6.2. Retraction Scheme

For safety reasons, a forward-retracting scheme is preferable for the fuselage-mounted assemblies. In a complete hydraulic failure situation, with the manual release of uplocks, the gravity and air drag would be utilized to deploy and down-lock the assembly and thus avoid a wheels-up landing [2]. As for wing-mounted assemblies, current practice calls for an inboard-retraction scheme which stows the assembly in the space directly behind the rear wing-spar. The bogie undercarriage may have an extra degree of freedom available in that the truck assembly can rotate about the bogie pivot point, thus requiring a minimum of space when retracted. As will be illustrated in the following section, deployed/retracted
position of the landing gear, as well as possible interference between the landing gear and surrounding structures, can easily be identified using the mathematical kinematic analysis.

6.3. Mathematical Kinematic Analysis

A mathematical kinematic analysis, which is more effective and accurate than the graphical technique, was selected to determine the axis of rotation that will, in one articulation, move the landing gear assembly from a given deployed position to a given retracted position. As shown in Fig. 6.1, a new coordinate system, termed the kinematic reference frame here, is defined such that the origin is located at the respective landing gear attachment locations with the axes aligned with the aircraft reference frame. The aircraft coordinate system-based origin permits constraints established in the kinematic reference frame, e.g., assembly clearance envelope, retraction path, and swept volume, be translated into the aircraft reference frame and checked for interference with surrounding structures.

6.3.1. The Pivot Axis and Its Direction Cosines

In the determination of the alignment of the landing gear pivot axis, it is assumed that the axle/piston centerline intersection is brought from its deployed position to a given location within the stowage volume. For wing-mounted assemblies, the retracted position of axle/piston centerline intersection is assumed to coincide with the center of the stowage volume. In the case of fuselage-mounted assemblies with a forward-retracting scheme, the retracted position is assumed to be at the center of the cross-sectional plane located at the forward third of the stowage length.*

---

* Note: to reduce structural cut-away, many forward retracting gears have shrink mechanisms. In particular, it appears that the Airbus A 330 and A340 aircraft may have shrink struts on the main gear. This consideration is neglected in the current analysis, but probably should be considered.
6.3.1.1. The Fuselage-mounted Assembly

For fuselage-mounted assemblies with a forward retracting-scheme, the pivot axis is defined by the cross product of the space vectors corresponding to the deployed and retracted position of a point location on the truck assembly. As shown in Fig. 6.2, the cross product of two vectors \( \mathbf{V}_1 \) and \( \mathbf{V}_2 \) representing the deployed and retracted positions of a given point location, here taken as the axle/piston centerline intersection, is orthogonal to both vectors, i.e., in the direction of the pivot axis. Thus,

\[
\mathbf{V} = \mathbf{V}_1 \times \mathbf{V}_2
\]

(6.1)

From standard vector operation, the direction cosines of the fuselage-mounted assembly is given as

\[
\begin{align*}
  l &= \frac{X}{\sqrt{X^2 + Y^2 + Z^2}} \\
  m &= \frac{Y}{\sqrt{X^2 + Y^2 + Z^2}} \\
  n &= \frac{Z}{\sqrt{X^2 + Y^2 + Z^2}}
\end{align*}
\]

(6.2)
and the angle between the two vectors, i.e., the angle of retraction ($\phi_{full}$) in this case, is calculated using the expression

$$\cos\phi_{full} = l_1l_2 + m_1m_2 + n_1n_2$$

(6.3)

where $l$, $m$, and $n$ are the respective direction cosines of the deployed and retracted space vectors.

Figure 6.2 Fuselage-mounted assembly pivot axis alignment

6.3.1.2. The Wing-mounted Assembly

The determination of the wing-mounted assembly pivot axis involves the deployed and retracted positions of two points on the assembly. Essentially, the problem consists of bringing the line segment between the two points from its deployed position to its retracted position [31]. For ease of visualization, a twin-wheel configuration is used here to illustrate the procedure involved in determining the alignment of the desired pivot axis. Identical procedure is used for other configurations as well.

As shown in Fig. 6.3, the axle/piston centerline intersection is selected as the first point (point A), while the second point (point B) is conveniently located at a unit distance along
the axle, inboard from the first point location. Retracted positions of the first and second
points are given as point A' and B', respectively.

\[ \mathbf{V}_4 = \mathbf{P}' + \mathbf{Z} \]

\[ \mathbf{V}_4 = \mathbf{P}' + \mathbf{Z} \]

Figure 6.3 Vector representation of the wing-mounted landing gear

Of the four point positions required in the analysis, the positions of point A and A' are
readily determined from the geometry of the landing gear and the stowage volume,
respectively. From simple vector algebra

\[ \mathbf{V}_2 = \mathbf{V}_1 + \mathbf{j} \]  \hspace{1cm} (6.4)

where subscripts 1 and 2 denote the space vector corresponding to the deployed positions
of points A and B, respectively. Similarly,

\[ \mathbf{V}_4 = \mathbf{V}_3 + \mathbf{U}_r \]  \hspace{1cm} (6.5)

where subscript 3 and 4 denote the retracted positions of point A and B, respectively, and
\( \mathbf{U}_r \) defines the orientation of the unit vector in its retracted position and is unknown.
To solve for \( U_r \), it is assumed that no devices are used to shorten the length of the strut during the retraction process, i.e., that the magnitudes of \( V_2 \) and \( V_4 \) remain constant,

\[
X_1^2 + (Y_1 + 1)^2 + Z_1^2 = (X_3 + X_U)^2 + (Y_3 + Y_U)^2 + (Z_3 + Z_U)^2
\]  

(6.6)

and that the magnitude of the retracted unit vector remains at unity

\[
X_U^2 + Y_U^2 + Z_U^2 = 1
\]  

(6.7)

The angle of inclination \( \theta \) of \( U_r \) in the \( yz \)-plane, which is one of the design variables that can be used to position the retracted truck assembly to fit into the available stowage space, is given as

\[
\tan \theta = \frac{Y_U}{Z_U}
\]  

(6.8)

The vector components of \( U_r \), and subsequently \( V_4 \), can then be determined by solving Eqs (6.6), (6.7), and (6.8) simultaneously.

As shown in Fig. 6.4, the pivot axis that will permit the achievement of the desired motion is defined by the cross product of the space vectors between the deployed and retracted positions of the two point locations, in this case points A and B,

\[
V = V_B \times V_A
\]  

(6.9)

where

\[
V_A = (X_3 - X_1)\hat{i} + (Y_3 - Y_1)\hat{j} + (Z_3 - Z_1)\hat{k}
\]  

(6.10)

and

\[
V_B = (X_4 - X_2)\hat{i} + (Y_4 - Y_2)\hat{j} + (Z_4 - Z_2)\hat{k}
\]  

(6.11)

Thus, the direction cosines of the wing-mounted assembly and the angle of rotation can be determined using Eqs (6.2) and (6.3), respectively. Note that the subscripts in Eq. (6.3) will be 1 and 3 in this case, i.e., the vectors corresponding to the deployed and retracted positions of point A, respectively.

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6.3.2. Retracted Position of a Given Point Location

In addition to determining the required pivot axis and angle of retraction, the analytic method is used to establish the retraction path and the stowed position of the landing gear assembly. Note that the drag and side struts are excluded in the analysis since the retraction of these items involves additional articulation, e.g., folding and swiveling, that cannot be modeled by the analysis.

Define point $A$ as an arbitrary point location on the landing gear assembly. Given the angle of rotation and the direction cosines of the pivot axis as determined above, the retracted position of point $A$, denoted here as $A'$, can be determined by solving the following system of linear algebraic equations [2, pp. 193-194]

$$
\begin{bmatrix}
X_{A'} \\
Y_{A'} \\
Z_{A'}
\end{bmatrix} = c_1 \begin{bmatrix} 
 l(X_A + mY_A + nZ_A) - X_A \\
 m(X_A + mY_A + nZ_A) - Y_A \\
 n(X_A + mY_A + nZ_A) - Z_A
\end{bmatrix} + c_2 \begin{bmatrix}
 mZ_A - nY_A \\
 nX_A - lZ_A \\
 lY_A - mX_A
\end{bmatrix} \begin{bmatrix}
 X_A \\
 Y_A \\
 Z_A
\end{bmatrix}
$$

where

$$
c_1 = 1 - \cos \phi \quad c_2 = \sin \phi \quad 0 < \phi < \phi_{\text{full}}
$$

Figure 6.4 Wing-mounted assembly pivot axis alignment
Similarly, the retraction path and swept volume of the assembly, as shown in Fig. 6.5, can be established by calculating several intermediate transit positions at a given interval of degrees. The above information can then be used to identify possible interference between the landing gear and surrounding structures during deployment/retraction.

Figure 6.5 Retraction path and swept volume of the landing gear

6.4. Integration and Stowage Considerations

For future large aircraft, interference between the landing gear assembly and the surrounding structure is one of the more important considerations in the development of kinematics. With the large number of doors required to cover the stowage cavity on such aircraft, a complex deployment/retraction scheme for both the landing gear and doors is required to ensure that no interference will occur under all conditions. Additionally, the availability of stowage volume can become a major integration problem as the number of tires increases with aircraft takeoff weight. Given the conflicting objectives between maximizing the volume that can be allocated for revenue-generating cargoes and providing adequate landing gear stowage space, a trade-off study involving crucial design parameters,
e.g., pivot axis alignment, angle of retraction, and bogie rotation, is needed to arrive at a satisfactory compromise with surround structures.

6.4.1. Truck Assembly Clearance Envelope

Clearances are provided to prevent unintended contact between the tire and the adjacent parts of the aircraft during operation, particularly in the case when the tire is damaged and continues to spin when stowed. As shown in Fig. 6.6, the maximum grown outside diameter \( D_G \) and section width \( W_G \) are determined using the expressions [25, p. 8]

\[
D_G = D + 2(1.115 - 0.074AR)H \quad (6.14)
\]

and

\[
W_G = 1.04W \quad (6.15)
\]

where \( D \) is the specified rim diameter, \( H \) is the maximum section height, \( W \) is the maximum section width, and \( AR \) is the tire aspect ratio defined as

\[
AR = \frac{H}{D} \quad (6.16)
\]

The values for the radial and lateral clearance, i.e., \( C_R \) and \( C_W \), respectively, are calculated using the expressions [25, p. 9]

\[
C_R = \begin{bmatrix}
0.073 \\
0.060 \\
0.047 \\
0.037 \\
0.029 \\
\end{bmatrix} W_G + 0.4 \quad \text{at} \quad \begin{bmatrix}
250MPH \\
225MPH \\
210MPH \\
190MPH \\
160MPH \\
\end{bmatrix} \quad (6.17)
\]

and

\[
C_W = 0.019W_G + 0.23 \quad (6.18)
\]

The constant coefficients found in Eqs (6.14), (6.15), and (6.16) are based on the maximum overall tire dimensions, plus growth allowance due to service and the increase in diameter due to centrifugal force.
Based on the clearance as determined above, the minimum radial and lateral distance between the tire and surrounding structures are calculated as follows [25, p. 9]

\[
R_x = \frac{D_G}{2} + C_R \quad (6.18)
\]

\[
W_x = \frac{W_G}{2} + C_W \quad (6.19)
\]

\[
S_x = \frac{C_W + C_R}{2} \quad (6.20)
\]

Given the minimum allowable distances obtained using Eqs (6.18), (6.19), and (6.20), a clearance envelope is established around the truck assembly. Then, using the kinematic analysis as outlined in the previous section, the boundary of the envelope is re-established in the retracted position. Note that the envelope is represented in the kinematic coordinate system, while the boundaries of the landing gear wheelwell are in the aircraft coordinate system. Recall that the origin of the kinematic reference frame is defined in the aircraft coordinate system. Thus, simple algebraic manipulation would bring both sets of data under the same coordinate system, whether it be the airframe or the kinematic reference frame. Stowage boundary violations can then be identified by comparing both sets of data for discrepancies.
Chapter 7  Aircraft Flotation Analysis

7.1. Introduction

The configuration of the landing gear has a direct impact on ground flotation, a term used to describe the capability of pavement and other surfaces to support an aircraft [32]. The number and arrangement of the wheels, along with the aircraft weight and its distribution between the nose and main assemblies, dictates the required pavement thickness for a particular aircraft. In addition, the type of the pavement found at the airports to be served by the aircraft also need to be considered. As shown in Fig. 7.1, existing runway and apron pavements can be grouped into two categories: flexible and rigid [7]. A flexible pavement, more commonly known as asphalt, may consist of one or more layers of bituminous materials and aggregate, i.e., surface, base, and subbase courses, resting on a prepared subgrade layer. On the other hand, rigid pavement may consist of a slab of portland cement concrete placed on a layer of prepared soil. The thickness of each of the layers must be adequate to ensure that the applied loads will not damage the surface or the underlying layers.

![Diagram of pavement cross-sections](image)

**Figure 7.1** Theoretical pavement cross-sections [33]

Based on the analyses as outlined in this chapter, a program was developed to determine the required flexible and rigid pavement thickness for a particular aircraft. Results obtained from the program were validated with actual design data to ensure that a high degree of reliability can be placed upon the program itself.
7.2. Design Pavement Thickness

Various flotation analyses have been developed over time in different countries and by different government agencies and organizations. Some agencies and organizations and the corresponding design methods are listed as follows [7]: the Federal Aviation Administration (FAA), the Portland Cement Association (PCA), the Waterways Experiment Station (S-77-1), and the British Air Ministry (LCN). The majority of these methods use the California bearing ratio (CBR) method of design for flexible pavements and Westergaard stress analysis for the rigid pavements [7].

7.2.1. Flexible Pavements

For flexible pavements, CBR is the standard measurement used to classify the bearing strength of the subgrade. It is essentially the ratio of the bearing strength of a given soil sample to that of crushed limestone gravel. It is expressed as a percentage of the limestone figure, i.e., a CBR of ten means that the subgrade has a bearing strength of ten percent to that of crushed aggregate. The original design method, which was developed by the California Division of Highways in 1928, evaluates the pavement thickness requirements for a given load condition and soil strength, assuming that the load is carried on a single wheel with a circular footprint area.

Until the middle of the 1950s, the analysis developed for the B-29, which features a dual wheel configuration, was extended to develop thickness design relationships for new aircraft with twin-tandem configurations. However, it appears that the analysis tends to produce slightly unconservative thickness estimates. Subsequent reevaluation of the theoretical work, which is based on Boussinesq’s theory [5], and test data showed that the slopes of pavement deflection versus wheel offset for the single wheel were equal to or steeper than for dual wheels at equal depths, as shown here in Fig. 7.2. A direct result of this study is the introduction of the concept of the equivalent single-wheel load (ESWL), which eventually became the foundation of the S-77-1 design method [34 and 35]. ESWL is essentially a fictitious load on a isolated wheel, having the same inflation pressure, and causing the same stresses in the runway material as those due to a group of wheels. This fictitious wheel load accounts for the fact that a given loading, spread over a number of contact areas, causes lower stresses in the runway material than would be the case when the same load is concentrated on a single wheel.
Probable locations where maximum pavement bearing stress might occur, e.g., directly under and between the tire contact areas, are shown in Fig. 7.3. The offset distance between these points and the center of individual tire contact area, as well as the depths below the surface at which the ESWL is computed, which is treated as the thickness of the pavement in the analysis, are subsequently represented in terms of the radius of the footprint area ($r$) [7, p. 429]

$$r = \frac{\sqrt{A}}{\pi}$$  \hspace{1cm} (7.1)

and the tire-ground contact area ($A$) is defined as

$$A = \frac{F}{P}$$  \hspace{1cm} (7.2)

where $F$ is the vertical main assembly load (per strut) and $P$ is the tire inflation pressure.

Given the offset distances and depths, curves such as the ones shown in Fig. 7.4 are used to determine the corresponding deflection factors. The principle of superposition is then used in calculating the multiple-wheel deflection factor ($j$), which is equal to the summation of the deflection factors produced by each tire in the multiple-wheel assembly at the point of analysis.
The ratio of load intensity of the single-wheel configuration to a single wheel of the multiple-wheel configuration is defined as the inverse of the ratio of the maximum deflection factors at a given depth, \(i.e.,\) the pavement thickness, \(\text{[7, p. 430]}\)

\[
\frac{F_s}{F_m} = \frac{f_m}{f_s}
\]  
(7.3)

where subscripts \(s\) and \(m\) denote single- and multiple-wheel configurations, respectively. Once the ratio of load intensity is determined, the \(ESWL\) is calculated using the expression

\[
ESWL = \frac{F_s F}{F_m N_w}
\]

(7.4)

where \(N_w\) is the number of wheels per strut. To account for the loading effect caused by the number of annual aircraft operations, the design thickness \((t)\) corresponding to a given CBR value is estimated using the expression \(\text{[7, p. 433]}\)

\[
t = \alpha_i \sqrt{\frac{ESWL}{8.1 CBR}} - \frac{A}{\pi}
\]

(7.5)

where \(\alpha_i\) is the load repetition factor as shown in Fig. 7.5. It is categorized by the number of tires used to calculate the \(ESWL\) and typically value corresponding to 10,000 passes are used in the calculation \(\text{[33]}\).
Figure 7.4  Deflection factor curves for Poisson’s ratio of 0.5 [34]

Figure 7.5  Aircraft load repetition factor [7]
7.2.2. Rigid Pavements

Stress in a concrete pavement is induced in four ways: tire loads, change of shape of slab due to differential in temperature and moisture between the top and the bottom of the slab, and the friction developed between slab and foundation when the slab expands/contracts. Since the primary consideration in the design of any pavement is the load which it is to carry, only the stresses induced by tire loads will be addressed.

The Westergaard stress analysis [36] assumes that the slab is a homogeneous, isotropic, and elastic solid in equilibrium. The reactions of the subgrade are assumed to be in the vertical direction only, and is proportional to the deflections of the slab. Additionally, the wheel load is assumed to be distributed over an elliptical footprint area. The stiffness of the slab relative to that of the subgrade is represented by the radius of relative stiffness of the concrete \( l \) [37, p. 56]

\[
l = \frac{E d^3}{\sqrt{12(1 - \mu^2)k}} 
\]  

(7.6)

where \( E \) is the modulus of elasticity for the concrete, \( d \) is the thickness of the slab, \( \mu \) is the Poisson’s ratio for the concrete, and \( k \) is the modulus of subgrade reaction. Typically, \( E \) is taken as 4,000,000 psi and \( \mu \) as 0.15 [7].

Critical bearing stresses for the interior and edge loading cases are examined. For the interior loading case, the load is applied at the interior of the slab at a considerable distance from any edge or joint. The maximum tensile stress \( (\sigma) \) at the bottom of the slab is given as [7, p. 441]

\[
\sigma_{int} = \frac{F_s}{d^2} \left\{ 0.275(1 + \mu)\log_{10} \frac{E d^3}{k(\frac{a + b}{2})} + 0.293(1 - \mu)\frac{a - b}{a + b} \right\} 
\]  

(7.7)

where \( F_s \) is the single wheel load, \( d \) is the design thickness, and \( a \) and \( b \) are the semi-axes of the footprint area ellipse. Considering the edge loading case next, the load is applied adjacent to an edge that has no capacity for load transfer. The maximum tensile stress is given as [7, p. 442]
Although the edge loading case produces a maximum stress that is the more critical of the two cases, in reality the probability of occurrence of this type of loading is relatively small, i.e., the traffic tends to be channelized with the highest concentration in the vicinity of the runway and taxiway centerlines [7]. In addition, rigid pavement design charts as provided by PCA, which are used as reference data in the following section, are based on the interior loading case. Therefore, the interior loading condition is selected as the basis of the rigid pavement analysis.

### 7.3. Pavement Thickness Estimates

Design pavement thickness and corresponding ACNs for the Boeing Models 737, 747, 767, and McDonnell Douglas DC10 were determined for four subgrade strength categories: ultra-low, low, medium, and high [33]. Each category is assigned a CBR value for the flexible pavements and a $k$ value for the rigid pavements; numerical values of each category are listed in Table 7.1.

<table>
<thead>
<tr>
<th>Category</th>
<th>CBR</th>
<th>$k$, lb/in$^3$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ultra-low</td>
<td>3.0</td>
<td>75.0</td>
</tr>
<tr>
<td>Low</td>
<td>6.0</td>
<td>150.0</td>
</tr>
<tr>
<td>Medium</td>
<td>10.0</td>
<td>300.0</td>
</tr>
<tr>
<td>High</td>
<td>15.0</td>
<td>550.0</td>
</tr>
</tbody>
</table>

For flexible pavements, ESWLs were computed using Eq. (7.4) from the surface down in multiples of footprint area radius. At each analysis depth, a CBR value was calculated using Eq. (7.5) and the repetition factor corresponding to 10,000 aircraft passes [33]. The result of this calculation is a set of design thickness and CBRs. Linear interpolation is then used to determine the final design thickness corresponding to the subgrade strength CBR values.

\[
\sigma_{ext} = \frac{2.2(1 + \mu)F^2}{(3 + \mu)d^2} \log_{10} \frac{Ed^3}{100k[(a+b)/2]^4} + \\
\frac{3(1 + \mu)F}{\pi(3 + \mu)d^2} \left[ 1.84 - \frac{4}{3} \mu + (1 + \mu) \frac{a - b}{a + b} + 2(1 - \mu) \frac{ab}{(a+b)^2} + 1.18(1 + 2\mu) \frac{b}{l} \right] 
\]

(7.8)
For rigid pavements, $ls$ were computed using Eq (7.6) from the surface down in predetermined increments, *i.e.*, the design thickness, for each of four subgrade categories. At each design thickness and $k$ value, a maximum tensile stress was calculated using Eq. (7.7). The result of this calculation is four sets of design thickness and the corresponding stresses. Linear interpolation is then used to determine the final design thickness corresponding to a concrete working stress of 400 psi [2].

Actual [7, 22, 38, and 39] and estimated pavement thickness are compared to determine the reliability of both analyses. As shown in Fig. 7.6a, the S-77-1 method tends to underestimate the required pavement thickness at the lower end of the CBR range, while it tends to overestimate the required pavement thickness at the upper end of the CBR range. Yet, the trend is consistent with the results obtained from a number of full-scale test tracks, *i.e.*, for heavy wheel loads, the theoretical thickness appeared to be too low for lower CBR values, and too high for higher CBR values. An interesting trend is observed upon closer examination of the actual pavement thickness data. As the subgrade strength increases, the required pavement thickness for aircraft with dual-twin truck assembly configurations, *i.e.*, B747, B767, and DC10, approach, if not fall below, the one required by aircraft with twin-wheel configuration, *i.e.*, B737. This can be attributed to the fact that the load on the pavement is better distributed as the number of wheels per assembly increases.

A vastly different trend, as shown in Fig. 7.6b, is exhibited by the Westergaard stress analysis: it tends to underestimate the required pavement thickness by roughly 30 percent across the entire $k$ range. The discrepancy can be attributed to the simplicity of the analysis itself. Primarily, the analysis did not consider the variations in the location and direction of maximum moment and stress in the concrete slab [37]. Essentially, the position of the maximum stress can be shifted and rotated depending on the magnitude of $l$ and the configuration and dimension of the truck assembly. In addition, the analysis did not include detailed design parameters such as fatigue of concrete due to repeated loading and interactions between layers of materials.
Figure 7.6 Actual and estimated pavement thickness comparison

a) Flexible pavements

b) Rigid pavements
Linear regression analysis was used to calibrate the estimated pavement thickness \( (t_{est}) \) against actual data. At each subgrade strength category, an aircraft weight-based correction factor is calculated using the expression

\[
f_c = c_1 W + c_2
\]

where \( c_1 \) and \( c_2 \) are constant coefficients as listed in Table 7.2. The estimated value and correction factor are then combined to arrive at the calibrated pavement thickness \( (t_{cal}) \), that is,

\[
t_{cal} = t_{est} + f_c
\]

The objective of this effort is to ensure that the discrepancy between the actual and estimated values will remain within a tolerable range. This is important when both analyses are used to examine the flotation characteristics of aircraft that are outside the existing pavement thickness database, namely, the next-generation high capacity commercial transports. As shown in Fig. 7.7, the calibrated thickness compared reasonably with the actual data.

<table>
<thead>
<tr>
<th></th>
<th>( c_1 )</th>
<th>( c_2 )</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Flexible</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>0.0000017</td>
<td>3.726</td>
</tr>
<tr>
<td>Low</td>
<td>0.000002</td>
<td>0.198</td>
</tr>
<tr>
<td>Medium</td>
<td>-0.000002</td>
<td>-1.630</td>
</tr>
<tr>
<td>High</td>
<td>-0.000007</td>
<td>-0.008</td>
</tr>
<tr>
<td><strong>Rigid</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>0.000003</td>
<td>4.002</td>
</tr>
<tr>
<td>Low</td>
<td>0.000003</td>
<td>3.420</td>
</tr>
<tr>
<td>Medium</td>
<td>0.000001</td>
<td>3.407</td>
</tr>
<tr>
<td>High</td>
<td>0.000000</td>
<td>3.325</td>
</tr>
</tbody>
</table>
Figure 7.7 Actual and calibrated pavement thickness comparison
7.4. ACN-PCN Conversion

In an effort to resolve the difference among various pavement design and evaluation methods, the International Civil Aviation Organization (ICAO) recommended universal adoption of the Aircraft-Pavement Classification Number (ACN-PCN) system [39] in 1983. The ACN-PCN system is not intended for the design or evaluation of pavements. It is, instead, a convenient and simple way of categorizing and reporting the pavement's capability to support aircraft on an unrestricted basis. The major appeal of the system is that it allows aircraft manufacturers to use any design/evaluation method of choice to determine the pavement thickness requirements of a particular aircraft. The design thickness is then converted to ACN and compared to PCNs of the airports to be served. If the ACN is equal to or less than the PCNs, the aircraft is cleared to operate out of the given airports subject to any limitation on the tire pressure.

The flexible pavement ACN is calculated using the expression [33, p. 3-11]

\[
ACN = \frac{(t^2 / 1000)}{(0.878/CBR - 0.01249)}
\]

(7.10)

where the design thickness \( t \) is expressed in terms of centimeters. As for the rigid pavements, ACN is obtained using the conversion chart as shown in Fig. 7.8.

![Figure 7.8 Rigid pavement ACN conversion chart [33]](image)

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7.4.1. ACN Estimates

Flexible and rigid pavement thickness requirements obtained earlier were converted to ACNs for conversion validation purposes. As shown in Fig. 7.9a, the estimated flexible pavement ACNs exhibit a trend similar to that of the thickness estimates, i.e., too low for lower CBR values and too high for higher CBR values. Apparently, the thickness calibration process did not eliminate the discrepancy introduced in the pavement thickness calculation entirely, and that the trend is carried over into the ACN conversion process. On the other hand, it appears that the calibration process for the rigid pavement has removed most of discrepancy that was introduced in the pavement thickness calculation. As shown in Fig. 7.9b, the conversion, in fact, overestimated the ACN for all aircraft across the entire k range.

![Graph showing actual and estimated ACN comparison](image)

Figure 7.9 Actual and estimated ACN comparison
Linear regression analysis was again used to calibrate the estimated ACN ($ACN_{est}$) against actual data. At each subgrade strength category, an aircraft weight-based correction factor is calculated using Eq. (7.9), except in this case the constant coefficients are $c_3$ and $c_4$ as listed in Table 7.3. The estimated value and correction factor are then combined to arrive at the calibrated ACN ($ACN_{cal}$), that is,

$$ACN_{cal} = ACN_{est} + f_c$$

(7.11)

As shown in Fig. 7.10, the calibration process has successfully brought the estimated ACNs closer to the actual data and thus improved the reliability of the flotation analysis.
Table 7.3 ACN correction constants

<table>
<thead>
<tr>
<th></th>
<th>$c_3$</th>
<th>$c_4$</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Flexible</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>0.000008</td>
<td>0.5178</td>
</tr>
<tr>
<td>Low</td>
<td>0.000010</td>
<td>-6.326</td>
</tr>
<tr>
<td>Medium</td>
<td>0.000009</td>
<td>-6.769</td>
</tr>
<tr>
<td>High</td>
<td>0.000022</td>
<td>-16.182</td>
</tr>
<tr>
<td><strong>Rigid</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>0.000006</td>
<td>-8.245</td>
</tr>
<tr>
<td>Low</td>
<td>0.000002</td>
<td>-4.940</td>
</tr>
<tr>
<td>Medium</td>
<td>0.000009</td>
<td>-7.628</td>
</tr>
<tr>
<td>High</td>
<td>0.000008</td>
<td>-6.519</td>
</tr>
</tbody>
</table>

Figure 7.9 Actual and calibrated ACN comparison

a) Flexible pavements
b) Rigid pavements

Figure 7.10 Actual and calibrated ACN comparison (concluded)
Chapter 8  Weight Estimation

8.1. Introduction

Statistical weight equations, although capable of producing landing gear group weights quickly and generally accurately, do not respond to all the variations in landing gear design parameters. In addition, the equations are largely dependent on the database of existing aircraft. For future large aircraft, such weight data is virtually non-existent. Thus, it is desirable that an analytical weight estimation method which is more sensitive than statistical methods to variations in the design of the landing gear should be adopted. The objectives are to allow for parametric studies involving key design considerations that drive landing gear weight, and to establish crucial weight gradients to be used in the optimization process.

Based on the procedures described in this chapter, algorithms were developed to size and estimate the weight of the structural members of the landing gear. The weight of non-structural members were estimated using statistical weight equations. The two were then combined to arrive at the final group weight.

8.2. Current Capabilities

The primary shortcoming of statistical methods is that only a limited number of weight-affecting parameters are considered, e.g., length of the strut, material ultimate strength, vertical load, and number of tires. As a result, it is extremely difficult to distinguish landing gears with different geometric arrangements using these parameters alone. Statistical weight equations are also constrained by what has been designed in the past, i.e., if an unconventional design or a new class of aircraft such as the proposed ultra-high-capacity transport is involved, there might not be sufficient data to develop a statistical base for the type of landing gear required.

The majority of existing equations calculate the landing gear weight purely as a function of aircraft takeoff gross weight. It is the simplest method for use in sizing analysis, and is adopted in ACSYNT as well as by Torenbeek [5] and General Dynamics, as given by Roskam [3]. The Douglas equation used in the blended-spanload concept [41] also falls into this category. Other weight equations, e.g., Raymer [42] and FLOPS (Flight Optimization System) [13], include the length of the landing gear in the calculation and
thus are able to produce estimates which reflect the effect of varying design parameters to some extent.

Actual and estimated landing gear weight fractions are presented in Fig. 8.1. Figure 8.1a provides comparisons for estimates which only use MTOW. Figure 8.1b provides comparisons with methods which take into account more details, specifically the gear length. As shown in Fig. 8.1a, for an MTOW up to around 200,000 lb, the estimated values from ACSYNT and Torenbeek are nearly equal. However, as the MTOW increases, completely different trends are observed for the two equations: an increasing and then a decreasing landing gear weight fraction is predicted by ACSYNT, whereas a continual increasing weight fraction is predicted by Torenbeek. As for the Douglas equation, an increasing weight fraction is observed throughout the entire MTOW range. Upon closer examination of the data presented, it was found that only a small number of actual landing gear weight cases are available to establish trends for aircraft takeoff weight above 500,000 pounds. In addition, even within the range where significant previous experience is available, the data scatter between actual and estimated values is too large to draw conclusions on the accuracy of existing weight equations. Evidently a systematic procedure is needed to validate the reliability of the statistical equations, and provide another level of estimation.

8.3. Analytical Structural Weight Estimation

Analytical weight estimation methods are capable of handling varying configurations and geometry, in addition to design parameters used in the statistical methods. As typified by Kraus [43] and Wille [44], the procedure consists of five basic steps: definition of gear geometry, calculation of applied loads, resolution of the loads into each structural member, sizing of required member cross-sectional areas, and calculation of component and total structural weight. Although these studies provided an excellent guideline toward the development of an MDO-compatible analysis algorithm, detailed discussions in the area of load calculations and structural design criteria were not included in the papers. To fill the gap, simplified loading conditions were determined from Torenbeek and the FAA [20], and structural analyses were developed as part of this work. Loading conditions are presented in Section 8.3.2., and the structural analyses are presented in Sections 8.3.3. and 8.3.4. and Appendix B.
Figure 8.1 Landing gear weights comparison
8.3.1 Generic Landing Gear Model

A generic model consisting of axles, truck beam, piston, cylinder, drag and side struts, and trunnion is developed based on existing transport-type landing gears. Since most, if not all, of the above items can be found in both the nose and main gear, the model can easily be modified to accommodate both types of assembly without difficulty. Although the torsion links are presented for completeness, they are ignored in the analysis since their contributions to the final weight are minor.

The model shown in Fig. 8.2 represents a dual-twin-tandem configuration. The model can be modified to represent a triple-dual-tandem or a dual-twin configuration with relative ease, *i.e.*, by including a center axle on the truck beam, or replacing the bogie with a single axle, respectively. The model assumes that all structural components are of circular tube construction except in the case of the drag and side struts, where an I-section can be used depending on the configuration. When used as a model for the nose gear, an additional side strut arranged symmetrically about the plane of symmetry is included.

![Figure 8.2 Generic landing gear model](image-url)
For added flexibility in terms of modeling different structural arrangements, the landing gear geometry is represented by three-dimensional position vectors relative to the aircraft reference frame. Throughout the analysis, the \(xz\)-plane is chosen as the plane of symmetry with the \(x\)-axis directed aft and the \(z\)-axis upward. The locations of structural components are established by means of known length and/or point locations, and each point-to-point component is then defined as a space vector in the \(x\), \(y\), and \(z\) directions. Based on this approach, a mathematical representation of the landing gear model is created and is shown in Fig. 8.3.

![Diagram of landing gear model]

**Figure 8.3 Mathematical representation of the landing gear model**

### 8.3.2. Applied Loads

External loads applied to the gear assemblies can be divided into dynamic and static loads: the former occurs under landing conditions while the latter occurs during ground operations. As listed in Table 8.1, seven basic loading conditions have been selected for analysis with the applied loads calculated as specified in FAR Part 25 [20]. These conditions are also illustrated in Fig. 8.4.
Table 8.1 Basic landing gear loading conditions [20]

<table>
<thead>
<tr>
<th>Dynamic</th>
<th>Static</th>
</tr>
</thead>
<tbody>
<tr>
<td>Three-point level landing</td>
<td>Turning</td>
</tr>
<tr>
<td>One-wheel landing</td>
<td>Pivoting</td>
</tr>
<tr>
<td>Tail-down landing</td>
<td></td>
</tr>
<tr>
<td>Lateral drift landing</td>
<td></td>
</tr>
<tr>
<td>Braked roll</td>
<td></td>
</tr>
</tbody>
</table>

The corresponding aircraft attitudes are shown in Fig. 8.4, where symbols $D$, $S$ and $V$ are the drag, side and vertical forces, respectively, $n$ is the aircraft load factor, $W$ is aircraft maximum takeoff or landing weight, $T$ is the forward component of inertia force, and $I$ is the inertial moment in pitch and roll conditions necessary for equilibrium. The subscripts $m$ and $n$ denote the main and nose gear, respectively.

![Diagram of aircraft attitudes](image)

a) Three-point level landing

![Diagram of aircraft attitudes](image)

b) One-wheel landing

Figure 8.4 Aircraft attitudes under dynamic and static loading conditions [20]
c) Tail-down landing

d) Lateral drift landing

e) Braked roll

f) Turning

Figure 8.4 Aircraft attitudes under dynamic and static loading conditions [20] (continued)
For the dynamic landing conditions listed in Table 8.1, the total vertical ground reaction \( F \) at the main assembly is obtained from the expression \([43]\)

\[
F = \frac{cW}{\eta S \cos \alpha} \left( \frac{V_s^2}{g} + S \cos \alpha \right)
\]

(8.1)

where \( c \) is the aircraft weight distribution factor, \( \eta \) is the gear efficiency factor, \( S \) is the total stroke length, \( \alpha \) is the angle of attack at touchdown, \( V_s \) is the sink speed, and \( g \) is the gravitational acceleration. Although the vertical force generated in the gear is a direct function of the internal mechanics of the oleo, in the absence of more detailed information Eq. (8.1) provides a sufficiently accurate approximation.

The maximum vertical ground reaction at the nose gear, which occurs during low-speed constant deceleration, is calculated using the expression \([5, p. 359]\)

\[
F_n = \frac{l_m + a_x / g}{l_m + l_n} h_c g W
\]

(8.2)

For a description of variables and the corresponding values involved in Eq. (8.2), refer to Chapter Four, Section Two.

The ground loads are initially applied to the axle-wheel centerline intersection except for the side force. As illustrated in Fig. 8.5, the side force is placed at the tire-ground contact point and replaced by a statically equivalent lateral force in the \( y \) direction and a couple whose magnitude is the side force times the tire rolling radius.
To determine the forces and moments at the selected structural nodes listed in Table 8.2, the resisting force vector \( \mathbf{F}_{\text{res}} \) is set equal and opposite to the applied force vector \( \mathbf{F}_{\text{app}} \)

\[
\mathbf{F}_{\text{res}} = -\mathbf{F}_{\text{app}}
\]  
(8.3)

whereas the resisting moment vector \( \mathbf{M}_{\text{res}} \) is set equal and opposite to the sum of the applied moment vector \( \mathbf{M}_{\text{app}} \) and the cross product of the space vector \( \mathbf{r} \) with \( \mathbf{F}_{\text{app}} \)

\[
\mathbf{M}_{\text{res}} = -\left( \mathbf{M}_{\text{app}} + \mathbf{r} \times \mathbf{F}_{\text{app}} \right)
\]  
(8.4)

Table 8.2 Selected structural nodes description

<table>
<thead>
<tr>
<th>Node</th>
<th>Description</th>
<th>Location (Figure 8.3)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Axle-beam centerline intersection</td>
<td>G/J</td>
</tr>
<tr>
<td>2</td>
<td>Beam-piston centerline intersection</td>
<td>F</td>
</tr>
<tr>
<td>3</td>
<td>Drag/side/shock strut connection</td>
<td>E</td>
</tr>
<tr>
<td>4</td>
<td>Cylinder-trunnion centerline intersection</td>
<td>B</td>
</tr>
</tbody>
</table>

8.3.3. Forces and Moment Resolution

Three-dimensional equilibrium equations are used to calculate member end reactions. Internal forces and moments are then determined from equilibrium by taking various cross-sectional cuts normal to the longitudinal axis of the member. To ensure that the information is presented in a concise manner, the methods used in the analysis are discussed only in general terms, while detailed derivations are compiled and presented in Appendix B.
8.3.3.1. Coordinate Transformation

Given that the mathematical landing gear model and the external loads are represented in the aircraft reference frame, transformation of nodal force and moment vectors from the aircraft to body reference frames are required prior to the determination of member internal reactions and stresses. The body reference frames are defined such that the $x_3$-axis is aligned with the component’s axial centerline, and $xz$-plane is a plane of symmetry if there is one. The transformation is accomplished by multiplying the force and moment vectors represented in the aircraft reference frame by the transformation matrix $L_{BA}$ [45, p. 117]

$$
F_B = L_{BA} F_A \quad (8.5)
$$

$$
M_B = L_{BA} M_A \quad (8.6)
$$

where subscripts $A$ and $B$ denote the aircraft and landing gear body reference frames, respectively. By inspection of the angles in Fig. 8.7, where subscripts 1, 2, and 3 denote the rotation sequence from the aircraft ($x$, $y$, and $z$) to the body ($x_3$, $y_3$, and $z_3$) reference frame, the three localized transformation matrices are [45, p. 117]

$$
L_1(\varphi_1) = \begin{bmatrix}
1 & 0 & 0 \\
0 & \cos \varphi_1 & \sin \varphi_1 \\
0 & -\sin \varphi_1 & \cos \varphi_1
\end{bmatrix} \quad (8.7a)
$$

$$
L_2(\varphi_2) = \begin{bmatrix}
\cos \varphi_2 & 0 & -\sin \varphi_2 \\
0 & 1 & 0 \\
\sin \varphi_2 & 0 & \cos \varphi_2
\end{bmatrix} \quad (8.7b)
$$

$$
L_3(\varphi_3) = \begin{bmatrix}
\cos \varphi_3 & \sin \varphi_3 & 0 \\
-\sin \varphi_3 & \cos \varphi_3 & 0 \\
0 & 0 & 1
\end{bmatrix} \quad (8.7c)
$$

Thus, the matrix $L_{BA}$ is given as [45, p. 117]

$$
L_{BA} = L_3(\varphi_3) L_2(\varphi_2) L_1(\varphi_1) \quad (8.8)
$$

or

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\[ L_{BA} = \begin{bmatrix}
    \cos \varphi_2 \cos \varphi_3 & \sin \varphi_1 \sin \varphi_2 \cos \varphi_3 & -\cos \varphi_1 \sin \varphi_2 \cos \varphi_3 \\
    -\cos \varphi_1 \sin \varphi_3 & -\sin \varphi_1 \sin \varphi_2 \sin \varphi_3 & \cos \varphi_1 \sin \varphi_2 \sin \varphi_3 \\
    \sin \varphi_2 & -\sin \varphi_1 \cos \varphi_2 & \cos \varphi_1 \cos \varphi_2
\end{bmatrix} \] (8.9)

a) About the \( x, x_1 \)-axis

b) About the \( y_1, y_2 \)-axis

c) About the \( z_2, z_3 \)-axis

Figure 8.6 Orientation of the axes and the corresponding rotation angles

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8.3.3.2. The Main Assembly

The main assembly drag strut and side strut structure is modeled as a space truss consisting of ball-and-socket joints and two-force members. As shown in Fig. 8.7 the loads applied to the cylinder consist of the side strut forces \( F_{\text{side}} \), drag strut force \( F_{\text{drag}} \), an applied force with components \( F_x \), \( F_y \), and \( F_z \), and an applied couple with moment components \( C_x \), \( C_y \), and \( C_z \). Internal axial actions are obtained using the method of sections. Equilibrium equations are then used to determine the magnitude of the internal axial forces in the isolated portion of the truss.

The shock strut cylinder, in addition to supporting the vertical load, also resists a moment due to asymmetric ground loads about the \( z \)-axis. This moment is transmitted from the truck beam assembly to the cylinder through the torsion links. Note that in the tandem configurations, the moment about the \( y \)-axis at the piston-beam centerline is ignored because of the pin-connection between the two. However, this moment must be considered in the dual-twin configuration, where the moment is resisted by the integrated axle/piston structure.

Figure 8.7 Idealized main assembly cylinder/drag/side struts arrangement
8.3.3.3. The Nose Assembly

As mentioned in the geometric definition section, an additional side strut, arranged symmetrically about the \(xz\)-plane, is modeled for the nose assembly. The addition of the second side strut results in a structure that is statically indeterminate to the first degree as shown in Fig. 8.8. The reactions at the supports of the truss, and consequently the internal reactions, can be determined by Castigliano's theorem [46, p. 611]

\[
\frac{\partial u_j}{\partial P_j} = \sum_{i=1}^{n} \frac{F_i}{E} \frac{\partial F_i}{\partial P_j}
\]

(8.10)

where \(u_j\) is the deflection at the point of application of the load \(P_j\), \(E\) is the modulus of elasticity, and \(l, F, \text{ and } A\) are the length, internal force, and cross-sectional area of each member, respectively. The theorem gives the generalized displacement corresponding to the redundant, \(P_j\), which is set equal to a value compatible with the support condition. This permits the solution of the redundant, and consequently all remaining internal actions, via equilibrium. As detailed in Appendix B, Section Two, the procedure is to first designate one of the reactions as redundant, and then determine a statically admissible set of internal actions in terms of the applied loads and the redundant load. By assuming a rigid support which allows no deflection, Eq. (8.10) is set to zero and solved for \(P_j\).

![Figure 8.8 Idealized nose gear cylinder/drag/side struts arrangement](image-url)

Figure 8.8 Idealized nose gear cylinder/drag/side struts arrangement
8.3.3.4. The Trunnion

When the gear is in the down-and-locked position, the trunnion is modeled as a prismatic bar of length $L$ with clamped ends. As shown in Fig. 8.9, the trunnion is subjected to a force with components $F_x$, $F_y$, and $F_z$, and a couple with components $C_x$ and $C_y$, at axial position $x = l_j$, where $0 < l_j < L$ and $0 \leq x \leq L$. Clamped end-conditions at $x = 0$ and $x = L$ yield ten homogeneous conditions, five at each end. At the load point $x = l_j$, there are five continuity conditions, i.e., $u$, $v$, $w$, $v'$, and $w'$, and five jump conditions corresponding to point-wise equilibrium of the internal actions and the external loads.

The linear elastic response of the trunnion is statically indeterminate, but can be readily solved by the superposition of an extension problem for the $x$-direction displacement component $u(x)$, a bending problem in the $xy$-plane for the $y$-direction displacement $v(x)$, and a bending problem in the $xz$-plane for the $z$-direction displacement $w(x)$. Using classical bar theory, the governing ordinary differential equation (ODE) for $u(x)$ is second order, while the governing ODEs for $v(x)$ and $w(x)$ are each fourth order. The governing equations are solved in the open intervals $0 < x < l_i$ and $l_i < x < L$, where the 20 constants of integration ($c_i$) resulting from integration of the ODEs with respect to $x$ are determined using the boundary and transition conditions as given above. Details of the solution are given in Appendix B, Section Three.

![Figure 8.9 Trunnion modeled as a clamped-clamped bar](image-url)
8.3.4. Member Cross-sectional Area Sizing

With the resolution of various ground loads, each structural member is subjected to a number of sets of internal actions that are due to combinations of extension, general bending, and torsion of the member. To ensure that the landing gear will not fail under the design condition, each structural member is sized such that the maximum stresses at limit loads will not exceed the allowables of the material and that no permanent deformation is permitted.

A description of selected cuts near major component joints and supports is given in Table 8.3. Normal and shear stresses acting on the cross section due to the internal actions were calculated at these locations and used in the sizing of the required member cross-sectional area.

<table>
<thead>
<tr>
<th>Section</th>
<th>Description</th>
<th>Location (Figure 8.3)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Axle-beam centerline intersection</td>
<td>G/J</td>
</tr>
<tr>
<td>2</td>
<td>Beam-piston centerline intersection</td>
<td>F</td>
</tr>
<tr>
<td>3</td>
<td>Piston</td>
<td>E</td>
</tr>
<tr>
<td>4</td>
<td>Cylinder/struts connection</td>
<td>E</td>
</tr>
<tr>
<td>5</td>
<td>Cylinder/trunnion centerline intersection</td>
<td>B</td>
</tr>
<tr>
<td>6</td>
<td>Forward trunnion mounting</td>
<td>A</td>
</tr>
<tr>
<td>7</td>
<td>Aft trunnion mounting</td>
<td>C</td>
</tr>
<tr>
<td>8</td>
<td>Drag strut</td>
<td>A</td>
</tr>
<tr>
<td>9</td>
<td>Side strut</td>
<td>D</td>
</tr>
</tbody>
</table>

8.3.4.1. Normal and Shear Stresses In a Thin-walled Tube

The normal stresses induced on the structural members are determined by combining the effects of axial load and combined bending, while the shear stresses are determined by combining the effects of torsion and shear forces due to bending [47].

The normal stress \( \tau_{xx} \) due to combined axial force and bending moments is given as

\[
\tau_{xx} = \frac{N}{A} + \frac{M_y}{I_{yy}} y - \frac{M_z}{I_{zz}} z
\]  

(8.11)

where \( N \) is the maximum axial force, \( A \) is the cross-sectional area of the member, \( M_y \) and \( M_z \) are the internal moment components, and \( I_{yy} \) and \( I_{zz} \) are the second area moments about the \( y \)- and \( z \)-axis, respectively. As shown in Appendix B, Section Four, the extremum values of the normal stress on a circular-tube cross section under combined axial and bending actions are
where \( r \) is the mean radius of the tube and \( t \) is the wall thickness. In the case of drag and side struts, the last two terms in Eq. (8.11) are zero since both members are modeled as pin-ended two-force members, thus,

\[
\tau_{xx} = \frac{N}{A} \quad \text{(8.13)}
\]

The shear stress \( \tau_{xs} \) due to combined transverse shear forces and torque is given as

\[
\tau_{xs} = \frac{q(s)}{t} + (\tau_{xs})_{torque} \quad \text{(8.14)}
\]

where \( q \) is the shear flow due to bending of a thin-walled tube, see Fig. 8.10. Given that

\[
\tan \theta_{max} = -\frac{V_z}{V_y} \quad \text{(8.15)}
\]

where \( \theta_{max} \) is the polar angle where the bending shear flow attains an extremum value and \( V_y \) and \( V_z \) are the shear forces components, Eq. (8.14) then becomes

\[
\tau_{xs_{max}} = \frac{1}{\pi rt} \left( \frac{T}{2r} \pm \sqrt{V_y^2 + V_z^2} \right) \quad \text{(8.16)}
\]

where \( T \) is the applied torque. Details of the solution are given in Appendix B, Section Four.
8.3.4.2. Design Criteria

Although aircraft structural design calls for multiple load paths to be provided to give fail-safe capability, the concept cannot be applied in the design of the landing gear structures. Accordingly, the gear must be designed such that the fatigue life of the gear parts can be safely predicted or that the growth of cracks is slow enough to permit detection at normal inspection intervals [4].

Von Mises yield criterion for ductile materials combined with a factor of safety is used to determine the stress limit state. The Mises equivalent stress is given as [46, p. 368]

$$\sigma_{Mises} = \sqrt{\tau_{xx}^2 + 3\tau_{xy}^2} \quad (8.17)$$

and the factor of safety is defined as the ratio of the yield stress of the material to the Mises equivalent stress, that is,

$$F.S. = \frac{\sigma_{yield}}{\sigma_{Mises}} \quad (8.18)$$

If this value is less than the specified factor of safety, the cross-sectional area of the component is increased until the desired value is attained.

In addition to material limit state, the critical loads for column buckling of the drag and side struts are considered because of the large slenderness ratio associated with these members. The slenderness ratio is defined as the length of the member \(L\) divided by the minimum radius of gyration \(\rho_{min}\). Assuming a perfectly aligned axial load, the critical buckling load for a pin-ended two-force member can be calculated using Euler's formula [46, p. 635]

$$N_{cr} = \frac{\pi^2EI}{L^2} \quad (8.19)$$

where \(E\) is the modulus of elasticity. In the case of a member with circular cross section, the moment of inertia \(I\) of the cross section is the same about any centroidal axis, and the member is as likely to buckle in one plane as another. For other shapes of the cross section, the critical load is computed by replacing \(I\) in Eq. (8.19) with \(I_{min}\), the minimum second moment of the cross section (bending about the weak axis). Note that the Euler’s formula only accounts for buckling in the long column mode and is valid for large
slenderness ratio, e.g., $L/p_{min} > 80$ for 6061-T6 Aluminum alloy. For slenderness ratio below this range, intermediate column buckling should be considered [48].

8.3.4.3. Sizing of the Cross-sectional Area

For thin-walled circular tubes, the cross-sectional area of the member is given as

$$A = \pi Dt$$

(8.20)

where the mean diameter ($D$) and design thickness ($t$) are both design variables. Instead of using these two variables in the analysis directly, the machinability factor ($k$), which is defined as the mean diameter divided by the wall thickness, is introduced to account for tooling constraints [49]. The factor is defined as

$$k = \frac{D}{t}$$

(8.21)

and has an upper limit of 40. For the thin-wall approximation to be valid in the structural analysis $k > 20$. Thus, the machinability factor is limited to

$$20 \leq k \leq 40$$

(8.22)

By replacing $t$ in Eq. (8.20) with Eq. (8.21) and using $D$ as a limiting design variable, the desired cross-sectional area can then be determined by iterating on $k$. Note that the lower limit of $k$ given in Eq. (8.21) may be violated in some instances. For structural members such as the axles, the truck beam, and piston, which typically feature $k$ values in the mid-teens, St. Venant's theory for torsion and flexure of thick-walled bars [50] should be used to calculate shear stresses. Essentially, the problem is broken down into torsion and bending problems and the shear stresses are calculated separately based on the linear theory of elasticity.

In general, the diameter of each cylindrical component is a function of either the piston or wheel dimension. In the case of shock strut, it is assumed that the internal pressure is evenly distributed across the entire cross-sectional area of the piston. That is, the piston area is a function of the internal oleo pressure ($P_2$) and the maximum axial force, that is,

$$A = \frac{N}{P_2} = \frac{\pi D_p^2}{4}$$

(8.23)

where $D_p$ is the outer diameter of the piston. Rearrangement of Eq. (8.23) gives
Assuming a perfect fit between the piston lining and the inner cylinder wall, the minimum allowable mean diameter of the cylinder is obtained by adding the wall thickness of the cylinder to the piston outer diameter. To reduce the level of complexity, the minimum allowable mean diameter of the trunnion is assumed to be identical to that of the cylinder. Similar assumptions are made concerning the axle and truck beam, except that the outer diameter of the above members is treated as a function of the diameter of the wheel hub. In the case of the axle, the maximum allowable mean diameter is obtained by subtracting the axle wall thickness from the hub diameter.

For the thin-walled I-section bar shown in Fig. 8.11, the cross-sectional area and principal centroidal second area moments are

\[ A = t(2b + h) \]  
\[ (8.25) \]

\[ I_{yy} = t \left[ \frac{h^3}{12} + 2b \left( \frac{h}{2} \right)^2 \right] \]  
\[ (8.26) \]

and

\[ I_{zz} = \frac{b^3 t}{6} \]  
\[ (8.27) \]

where \( h \) is the web height and \( b \) is the width of the two flanges. Assume that \( I_{yy} > I_z \), algebraic manipulations then result in

\[ \frac{h}{b} > \sqrt{2} \]  
\[ (8.28) \]

and the \( z \)-axis is the weak axis in bending. The cross-sectional area is related to the second area moment by the minimum radius of gyration, that is,

\[ A = \frac{I_{zz}}{\rho_{\text{min}}} \]  
\[ (8.29) \]

or for the I-section
Since only the cross-sectional area is used in the weight computation, it is not necessary to determine the actual dimensions of the sectional height and width. Instead, one of the dimensions, usually the height, is treated as a function of the piston diameter and the other is then calculated with a predetermined \( h/b \) ratio.

8.3.5. Structural Weight Calculation

The final step of the analytical procedure is to calculate the weight of each member based on its cross-sectional area, length, and the material density. Recall that seven different loading conditions were examined in the analysis, which results in seven sets of cross-sectional areas for each member. To ensure that the component will not fail under any of the seven loading conditions, the maximum cross-sectional area from the sets is selected as the final design value. Component weights are then calculated by multiplying each of the cross-sectional areas by the corresponding length and material density. The summation of these calculations then becomes the structural weight of the idealized analytical model.

8.3.6. Validation of the Analysis

For analysis validation purposes, the landing gears for the Boeing Models 707, 727, 737 and 747 were modeled and analyzed. The estimated structural weight, which includes the axle/truck, piston, cylinder, drag and side struts, and trunnion, accounts for roughly 75 percent of the total structural weight that can be represented in the model [43]. The remaining 25 percent of the gear structural weight is made up of the torsion links, fittings, miscellaneous hardware, and the internal oleo mechanism, \( e.g. \), the metering tube, seals,
oil, pins, and bearings. Note that actual and estimated structural weights presented in Tables 8.4 and 8.5 only account for the components that were modeled in the analysis.

Table 8.4 Main assembly structural weight comparison

<table>
<thead>
<tr>
<th>Aircraft</th>
<th>Estimated, lb</th>
<th>Actual, lb</th>
<th>Est/Act</th>
</tr>
</thead>
<tbody>
<tr>
<td>B737</td>
<td>784</td>
<td>768</td>
<td>1.02</td>
</tr>
<tr>
<td>B727</td>
<td>1396</td>
<td>1656</td>
<td>0.84</td>
</tr>
<tr>
<td>B707</td>
<td>2322</td>
<td>2538</td>
<td>0.91</td>
</tr>
<tr>
<td>B747</td>
<td>9788</td>
<td>11323</td>
<td>0.86</td>
</tr>
</tbody>
</table>

Table 8.5 Nose assembly structural weight comparison

<table>
<thead>
<tr>
<th>Aircraft</th>
<th>Estimated, lb</th>
<th>Actual, lb</th>
<th>Est/Act</th>
</tr>
</thead>
<tbody>
<tr>
<td>B737</td>
<td>107</td>
<td>145</td>
<td>0.74</td>
</tr>
<tr>
<td>B727</td>
<td>171</td>
<td>327</td>
<td>0.52</td>
</tr>
<tr>
<td>B707</td>
<td>159</td>
<td>222</td>
<td>0.72</td>
</tr>
<tr>
<td>B747</td>
<td>1010</td>
<td>1439</td>
<td>0.70</td>
</tr>
</tbody>
</table>

Differences between the actual and estimated structural weights can be attributed to several factors. First, the models analyzed are extremely simple, i.e., structural members were represented with simple geometric shapes and no considerations have been given to fillet radii, local structural reinforcement, bearing surfaces, etc. As for the analysis itself, simplistic equations were used to calculate the applied static and dynamic loads, and idealized structural arrangements were used to determine the member internal reactions. However, it should be noted that the results are consistent with Kraus’ original analysis; where an average of 13 percent deviation was cited [43].

8.4. Landing Gear Group Weight Estimation

Although proven to be far more responsive to variations in design parameters, it is unlikely that an analytical tool will replace statistical methods. In fact, both methods should be used as complements to one another. This is particularly true in the calculation of the landing gear group weight, where the analytical and statistical methods can be used to determine the structural and non-structural component weights, respectively.

For large transports, landing gear structural weight accounts for roughly 57 percent of the landing gear group weight. The remaining weight is made up by the rolling stock and controls; the former accounts for roughly 34 percent of the total weight, while the latter accounts for the last nine percent. Note that the weights of the tires, wheels and brakes that make up the rolling stock have already been determined in previous chapters and no
additional calculations are required. As for the controls, \textit{i.e.}, actuation and steering mechanisms, the items can be estimated statistically with sufficient accuracy and thus eliminates the need to resort to an analytical method [App. A]. A detailed weight breakdown is provided in Table 8.6; the values are presented in terms of percent total landing gear weight.

<table>
<thead>
<tr>
<th>Component</th>
<th>Main assembly</th>
<th>Nose assembly</th>
</tr>
</thead>
<tbody>
<tr>
<td>Rolling stock</td>
<td>32.0</td>
<td>2.0</td>
</tr>
<tr>
<td>Wheels</td>
<td>6.0</td>
<td>1.0</td>
</tr>
<tr>
<td>Tires</td>
<td>10.0</td>
<td>1.0</td>
</tr>
<tr>
<td>Brakes</td>
<td>16.0</td>
<td>0.0</td>
</tr>
<tr>
<td>Miscellaneous</td>
<td>0.0</td>
<td>0.0</td>
</tr>
<tr>
<td>Structure</td>
<td>50.0</td>
<td>7.0</td>
</tr>
<tr>
<td>Shock strut</td>
<td>32.0</td>
<td>4.0</td>
</tr>
<tr>
<td>Braces</td>
<td>12.0</td>
<td>1.0</td>
</tr>
<tr>
<td>Fittings</td>
<td>5.0</td>
<td>1.0</td>
</tr>
<tr>
<td>Miscellaneous</td>
<td>1.0</td>
<td>1.0</td>
</tr>
<tr>
<td>Controls</td>
<td>7.0</td>
<td>2.0</td>
</tr>
<tr>
<td>Total</td>
<td>89.0</td>
<td>11.0</td>
</tr>
</tbody>
</table>

Using the combined analytical and statistical approach presented here, the landing gear group weight for the Boeing Models 707, 727, 737, and 747 were calculated and compared with actual values. As presented in Table 8.7a, the analysis tends to underestimate the group weight as the aircraft takeoff weight increases. Linear regression analysis was used to calibrate the estimated group weights \(W_{\text{est}}\) so they agree with the actual values. Correction factors were calculated using the expression

\[
f_c = 0.005W - 525 \tag{8.31}
\]

where \(W\) is the aircraft weight. The correction factor is then combined with \(W_{\text{est}}\) to arrive at the calibrated landing gear group weight \(W_{\text{cal}}\), that is,

\[
W_{\text{cal}} = W_{\text{est}} + f_c \tag{8.32}
\]
The objective of this effort is to ensure that the discrepancy between the actual and estimated values will remain within a tolerable range. This is important when the analysis is used to examine the weight of landing gear for aircraft that are outside the existing pavement thickness database. The calibrated results are shown in Table 8.7b.

Table 8.7 Landing gear group weight comparison

a) Estimated group weight

<table>
<thead>
<tr>
<th>Aircraft</th>
<th>Estimated, lb</th>
<th>Actual, lb</th>
<th>Est/Act</th>
</tr>
</thead>
<tbody>
<tr>
<td>B737</td>
<td>4479</td>
<td>4382</td>
<td>1.02</td>
</tr>
<tr>
<td>B727</td>
<td>5976</td>
<td>6133</td>
<td>0.97</td>
</tr>
<tr>
<td>B707</td>
<td>9510</td>
<td>11216</td>
<td>0.85</td>
</tr>
<tr>
<td>B747</td>
<td>27973</td>
<td>31108</td>
<td>0.90</td>
</tr>
</tbody>
</table>

b) Calibrated group weight

<table>
<thead>
<tr>
<th>Aircraft</th>
<th>Calibrated, lb</th>
<th>Actual, lb</th>
<th>Cal/Act</th>
</tr>
</thead>
<tbody>
<tr>
<td>B737</td>
<td>4499</td>
<td>4382</td>
<td>1.03</td>
</tr>
<tr>
<td>B727</td>
<td>6301</td>
<td>6133</td>
<td>1.03</td>
</tr>
<tr>
<td>B707</td>
<td>10545</td>
<td>11216</td>
<td>0.94</td>
</tr>
<tr>
<td>B747</td>
<td>31138</td>
<td>31108</td>
<td>1.00</td>
</tr>
</tbody>
</table>
Chapter 9 Analysis Package

9.1. Introduction

Four FORTRAN programs and a spreadsheet based on the analyses as outlined in previous chapters were developed for eventual incorporation into existing MDO codes. Programs CONFIG, LIMIT, PAVE, and GEARWEI can be used together in an iterative fashion to study the global effects of variations in the landing gear design parameters on configuration, system integration, airfield compatibility, and weight. In addition, the programs can be used individually to analyze a particular aspect of a given concept. In both cases, aircraft configuration characteristics have to be imported either from existing aircraft sizing codes or disciplinary analyses, while landing gear-related parameters must be specified by the user or set up as defaults. Within an optimization framework, these parameters would be treated as design variables whose optimum values would be computed by the optimizer to achieve a desired objective. However, the goal here is to demonstrate the algorithms, which can be used to help automate the landing gear design process.

In addition to the four programs as mentioned above, a simple Microsoft Excel-based spreadsheet was created to establish the maximum permissible cg range of a particular aircraft concept. The spreadsheet requires estimated component weights be imported from existing aircraft sizing code, while the corresponding component cg ranges can be specified by the user or set up as defaults.

9.2. Description of Programs

A simple spreadsheet software is used to establish the forward and aft limits of the permissible aircraft cg range. Given the aircraft configuration characteristics and component weights, the spreadsheet uses the specified component cg range as detailed in Chapter Two to calculate the maximum permissible aircraft cg range.

The primary task for program CONFIG is to develop a landing gear model that can be used as the baseline configuration. Given the aircraft weight, configuration characteristics, and the number of struts and tires, the program determines the loads on the tires and the
total braking energy to be absorbed by the brakes. Suitable tires, wheels, and brakes are either selected from manufacturers' catalogs or sized statistically as detailed in Chapters Four and Five. The length of the structural components, e.g., axles, truck beam, piston, cylinder, and trunnion, are determined based on the attachment scheme and clearance requirements. As for the linkages, a generic attachment scheme derived from existing commercial transports is used to determine the arrangement and required length of the drag and side struts. Based on this information, the program establishes a mathematical model of the notional landing gear in three-dimensional space, which is to be used by the remaining programs for detailed analysis.

Program LIMIT is used to examine the design and kinematic characteristics of the landing gear. Given the configuration characteristics of the aircraft and the model of the notional landing gear, turnover angle, pitch and roll angles during takeoff and landing, ground clearance, and turning radii are calculated using procedures as detailed in Chapter Three. The calculated values are then compared with a list of specified requirements to identify possible constraint violations. From the dimension and arrangement of the landing gear and the allocated stowage space, pivot axis and retraction angle are determined using mathematical kinematic analysis as detailed in Chapter Six. In addition, retraction path, swept volume, and stowed position are established and compared with stowage boundaries for possible structural interference.

The flotation characteristics of the aircraft are determined by program PAVE. Flexible and rigid pavement bearing stresses associated with specified loading conditions are calculated using pavement design procedures as detailed in Chapter Seven. The required pavement thickness is converted to the standard pavement bearing strength reporting system and tabulated for comparison purposes.

The component and group weights of the landing gear are calculated by program GEARWEI. As detailed in Chapter Eight, the structural weight of the landing gear is determined analytically from the notional landing model, while the weight of the non-structural components is determined from a statistical database. These weights are combined to arrive at the landing gear group weight.
9.3. *Organization of Analyses*

The programs are organized as shown in Figure 9.1 for use in an iterative fashion to study the global effects of variations in the landing gear design parameters. Aircraft weight and configuration characteristics, as well as a limited number of landing gear-related design parameters, enter the package through program CONFIG. The former set of data is obtained either from existing aircraft sizing codes or disciplinary analyses, *e.g.*, ACSYNT and FLOPS, whereas the latter is user-specified or is set up as defaults. Using this information as a starting point, program CONFIG generates a notional landing gear model, as well as data sets to be used as inputs for programs PAVE, LIMIT, and GEARWEI. The first two programs then assess flotation, operational stability, maneuverability, and stowage aspects of the aircraft/landing gear concept are examined. If all the design constraints are satisfied, landing gear weight is then estimated in program GEARWEI. Note that if any of the design constraints cannot be satisfied by the current configuration, user-specified modifications to the model or design parameters will be needed to resolve the violations through an iterative process. The execution of all the programs is essentially instantaneous.

The current state of the analysis package is a compilation of a number of separate analysis codes. The package does not have the capability to generate the required landing gear-related parameters, *e.g.*, the number of tires and struts, attachment location, and stowage space, based on imported aircraft configuration characteristics. Thus, starting values, or "guesstimates", must be provided for these design parameters. The parameters can then easily be varied by the user, or an optimizer, for parametric study purposes and the information used to select the optimum design.
9.3.1. Input/Output Data

Data required by the analysis package are listed in Table 9.1. The majority of this information consists of geometric and weight characteristics associated with the aircraft: wing area and span, quarter chord sweep, fuselage length and width, maximum takeoff/landing weight, aircraft cg location, etc. These design parameters are readily available from existing aircraft sizing codes and can easily be rearranged into the "card-style" inputs used by the analyses. The remaining information consists of landing-gear related parameters, and as mentioned in the previous section, must be provided by the user or selected from defaults.
Table 9.1 Required input data

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Type</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>Imported</td>
<td>Geometric characteristics; location</td>
</tr>
<tr>
<td>Fuselage</td>
<td>Imported</td>
<td>Geometric characteristics</td>
</tr>
<tr>
<td>Engine/Nacelle</td>
<td>Imported</td>
<td>Geometric characteristics; location</td>
</tr>
<tr>
<td>Weight</td>
<td>Imported</td>
<td>Takeoff/landing weights; weight distribution; aircraft cg location</td>
</tr>
<tr>
<td>Landing gear</td>
<td>User-specified or default</td>
<td>Design/selection criteria; number of tires/struts; location; clearance; stowage space</td>
</tr>
</tbody>
</table>

A description of the results generated by individual analysis is given in Table 9.2. It should be pointed out that these data only represent part of information that is produced by the analyses. Intermediate results, e.g., constraint boundaries, landing gear loads and induced stresses, that might be of interest or importance to a particular discipline, are currently internal to the analyses. To access this information would require modification of the output section of the program(s) to extract these data. Sample input/output files for the four programs can be found in Appendix E.

Table 9.2 Analysis-generated output data

<table>
<thead>
<tr>
<th>Program</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>CONFIG</td>
<td>Selected tires/wheels data; strokes; load-stroke curve; mathematical landing gear model</td>
</tr>
<tr>
<td>LIMIT</td>
<td>Trunnion alignment; retracted landing gear position; stability/operational characteristics; constraint violations</td>
</tr>
<tr>
<td>PAVE</td>
<td>ESWLs; concrete bearing stresses; pavement thickness; ACNs</td>
</tr>
<tr>
<td>GEARWEI</td>
<td>Structural member dimensions; landing gear component/group weight</td>
</tr>
</tbody>
</table>
9.4. Aircraft CG Estimation Spreadsheet

In addition to the four programs that made up the analysis package, a simple Microsoft Excel-based spreadsheet was created to establish the maximum permissible aircraft cg range for any given aircraft concept. Aircraft component identification was provided in the first column, while estimated component weights as obtained from existing sizing codes are entered into the second column. Given the aircraft geometric characteristics, the forward and aft component cg limits determined based on the generic aircraft layout developed in Chapter Three are entered into column three and four, respectively. The spreadsheet calculates the moments corresponding to the forward and aft component cg limits, and then divides the sums of the moments by the total component weight to arrive at the maximum forward and aft aircraft cg limits.
Chapter 10  Parametric Studies

10.1. Introduction

The emergence of the next-generation high-capacity commercial transports [51 and 52] provides an excellent opportunity to demonstrate the capability of the landing gear analysis package as detailed in the previous chapter. Landing gear design variables were varied parametrically to show their effects on the weight, flotation, and stability characteristics. Dependencies between the variables and characteristics established from the parametric analysis, as well as the magnitude of the effect, can be used as a guideline in selecting the most effective means to alter a particular aircraft-landing gear configuration so that the desired characteristics can be obtained.

10.2. The Ultra-High-Capacity Transports

A conceptual ultra-high-capacity transport (UHCT) was established based on a study by Arcara et al. [53] and industry forecasts [54, 55 and 56]. Configuration characteristics of the aircraft are presented in Table 10.1. Note that the aircraft is classified as a Design Group VI aircraft according to its wingspan, which is slightly over the specified 262-foot upper limit [7]. To match the geometric model of the aircraft as found in ACSYNT, the wing is modeled as a simple trapezoid without an inboard trailing-edge extension, i.e., the Yehudi. As a result, the location of the wing mac and hence the aircraft cg location and the attachment position of the main assembly are slightly forward of where they would be in the actual design.

Twenty-four main assembly tires arranged in a triple-dual-tandem configuration, i.e., six tires per strut, are used as an initial design. Tire selection is based on the minimum weight criterion. Forged aluminum and carbon are selected as the construction materials for the wheels and brakes, respectively. For the landing gear structure, 300M high-strength steel is used. The attachment scheme calls for two main gear units mounted on the wing and two units on the fuselage: the wing-mounted units retract inboard, while the fuselage-mounted units retract forward into the fuselage. The ensuing wheelbase and track dimensions are approximately 102 and 39 feet, respectively. Given this information, the analysis package as described in Chapter Nine is used to determine the design characteristics associated with this particular aircraft-landing gear combination. As shown in Table 10.2, all design constraints are satisfied. The landing gear weighs about 56,900
pounds and accounts for roughly 17.4 percent of the aircraft structural weight, or 4.6 percent of the MTOW.

Table 10.1 Configuration characteristics of a conceptual UHCT

<table>
<thead>
<tr>
<th></th>
<th>Baseline</th>
</tr>
</thead>
<tbody>
<tr>
<td>Passenger capacity</td>
<td>800</td>
</tr>
<tr>
<td>Range, nmi</td>
<td>7,500</td>
</tr>
<tr>
<td>Fuselage length, ft</td>
<td>250.0</td>
</tr>
<tr>
<td>Fuselage width, ft</td>
<td>24.0</td>
</tr>
<tr>
<td>Wingspan, ft</td>
<td>264.0</td>
</tr>
<tr>
<td>Wing area, ft²</td>
<td>8,324</td>
</tr>
<tr>
<td>Aspect ratio</td>
<td>8.4</td>
</tr>
<tr>
<td>MTOW, lb</td>
<td>1,230,000</td>
</tr>
<tr>
<td>Fuel, lb</td>
<td>550,000</td>
</tr>
</tbody>
</table>

Table 10.2 Baseline aircraft design characteristics

<table>
<thead>
<tr>
<th></th>
<th>Calculated</th>
<th>Constraint</th>
</tr>
</thead>
<tbody>
<tr>
<td>Sideways turnover angle, deg</td>
<td>40.7</td>
<td>&lt; 63.0</td>
</tr>
<tr>
<td>Roll angle, deg</td>
<td>7.2</td>
<td>&lt; 8.0</td>
</tr>
<tr>
<td>Available touchdown angle, deg</td>
<td>16.7</td>
<td>~ 15.0</td>
</tr>
<tr>
<td>Available takeoff rotation angle, deg</td>
<td>15.4</td>
<td>~ 15.0</td>
</tr>
<tr>
<td>Nacelle-to-ground clearance, in</td>
<td>10.0</td>
<td>&gt; 7.0</td>
</tr>
<tr>
<td>Castor angle, deg</td>
<td>37.0</td>
<td>&lt; 60.0</td>
</tr>
<tr>
<td>Turning radius, ft</td>
<td>78.4</td>
<td>&lt; 100.0</td>
</tr>
<tr>
<td>Gear weight, lb</td>
<td>56,885</td>
<td>-</td>
</tr>
<tr>
<td>Weight fraction, %MTOW</td>
<td>4.63</td>
<td>-</td>
</tr>
</tbody>
</table>

The flotation characteristics are given in Table 10.3 along with actual data for the McDonnell Douglas DC10, which are highest among existing aircraft. As shown in Table 10.3, major runway reinforcements will be needed at airports with a combination of flexible pavements and a low bearing strength subgrade. Costs associated with such an upgrade could be in the $100 million range [6], an investment that might not be acceptable to airport authorities. Consequently, some major international airports with flexible pavements might not be able to handle the UHCT unless design changes are made to the aircraft. Results in Table 10.3 indicate that airports with rigid pavements are better suited in handling this class of aircraft. Note that as the subgrade strength approaches its upper limit, the required flexible and rigid pavement thickness for the new aircraft are actually lower than the ones required by the DC10. This is consistent with the trend observed in Chapter.
Seven, i.e., as the number of wheels per strut increases, the required pavement thickness decreases with the increase in the subgrade strength.

Table 10.3 Baseline aircraft flotation characteristics

<table>
<thead>
<tr>
<th>Subgrade strength</th>
<th>Thickness, in (UHCT/DC10)</th>
<th>ACN (UHCT/DC10)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flexible</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>73.5/63.9</td>
<td>134/97</td>
</tr>
<tr>
<td>Low</td>
<td>39.1/37.8</td>
<td>80/70</td>
</tr>
<tr>
<td>Medium</td>
<td>25.5/26.9</td>
<td>60/59</td>
</tr>
<tr>
<td>High</td>
<td>16.0/20.2</td>
<td>47/53</td>
</tr>
<tr>
<td>Rigid</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>18.6/17.0</td>
<td>96/75</td>
</tr>
<tr>
<td>Low</td>
<td>16.4/15.2</td>
<td>79/64</td>
</tr>
<tr>
<td>Medium</td>
<td>13.3/13.0</td>
<td>62/53</td>
</tr>
<tr>
<td>High</td>
<td>11.5/11.8</td>
<td>50/44</td>
</tr>
</tbody>
</table>

10.3. Parametric Studies

Given the baseline aircraft-landing gear combination as characterized in the previous section, landing gear design variables were varied parametrically to show their effects on the weight, flotation, and stability characteristics. Dependencies between the various control variables and resulting aircraft characteristics established from this study, as shown here in Fig. 10.1, can be used as a guideline in selecting the most effective means to alter a particular aircraft-landing gear configuration so that the desired characteristics may be obtained. Note that there are instances where flotation and stability characteristics remain unchanged despite variations in the design parameters. Thus, only the characteristics being affected will be discussed.

In order for the UHCT to be able to operate from current airports without extensive runway reinforcement, additional tires are required to redistribute the weight of the aircraft over a larger tire-ground contact area. Provided the number of main assembly struts remains unchanged at four, the number of tires were varied both above and below the baseline (24). As shown in Fig. 10.1a, landing gear weight fraction increases with the increase in the number of tires. Evidently, weight penalties associated with the dimension of the truck assembly as well as the increased part-count, easily outstrip weight savings obtained from lighter tire and wheel designs that come with reduced load-carrying requirements. As shown in Table 10.4, the increased tire-ground contact area leads to
reductions in required pavement thickness and the corresponding ACN when compared to
the baseline figures.

![Graphs showing changes in landing gear weight fraction due to design parameter variations]

Figure 10.1 Changes in landing gear weight fraction due to design parameter variations
Figure 10.1 Changes in landing gear weight fraction due to design parameter variations (concluded)

Varying the number of main assembly struts is another option to be considered in producing the desired flotation characteristics. As shown in Fig. 10.1b, provided the number of tires remains unchanged at 24, a reduction in the landing gear weight fraction is realized with an increase in the number of main assembly struts. The reduction can be attributed to the decrease in the number of tires found on each strut, which effectively lowers the combined load on the structural members and therefore leads to a lighter structure. As shown in Table 10.5, a reduction in the required flexible pavement thickness is evident as the number of the struts increases. Recall that in multiple-wheel assemblies, the flexible pavement bearing stresses are directly proportional to the number of tires per strut involved in the calculation and hence the required pavement thickness. The rigid
pavement thickness requirements remain unchanged since the stresses obtained from Westergaard’s analysis are independent of the number of main assembly struts.

Table 10.4 Number of main assembly tires, four-strut configuration

<table>
<thead>
<tr>
<th>Subgrade strength</th>
<th>Thickness, in ACN</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>20 tires (Des./Base)</td>
</tr>
<tr>
<td>Flexible</td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>71.1/73.5</td>
</tr>
<tr>
<td>Low</td>
<td>39.0/29.1</td>
</tr>
<tr>
<td>Medium</td>
<td>24.6/25.5</td>
</tr>
<tr>
<td>High</td>
<td>15.6/16.0</td>
</tr>
<tr>
<td>Rigid</td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>19.6/18.6</td>
</tr>
<tr>
<td>Low</td>
<td>17.3/16.4</td>
</tr>
<tr>
<td>Medium</td>
<td>14.1/13.3</td>
</tr>
<tr>
<td>High</td>
<td>12.2/11.5</td>
</tr>
</tbody>
</table>

Table 10.5 Number of main struts, 24-tire configuration

<table>
<thead>
<tr>
<th>Subgrade strength</th>
<th>Thickness, in ACN</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>five struts (Des./Base)</td>
</tr>
<tr>
<td>Flexible</td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>73.5/73.5</td>
</tr>
<tr>
<td>Medium</td>
<td>25.5/25.5</td>
</tr>
<tr>
<td>High</td>
<td>16.0/16.0</td>
</tr>
<tr>
<td>Rigid</td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>18.6/18.6</td>
</tr>
<tr>
<td>Low</td>
<td>16.4/16.4</td>
</tr>
<tr>
<td>High</td>
<td>11.5/11.5</td>
</tr>
</tbody>
</table>

Besides increasing the number of main assembly tires and struts to bring about the desired reduction in the required pavement thickness, another option is to select a tire with a lower inflation pressure. As shown in Fig. 10.1c, the minimum inflation pressure candidate offers the lowest landing gear weight fraction of the three selection criteria. A reduced inflation pressure also means an increased tire-ground contact area, hence reduced
pavement loads and pavement thickness requirements as shown in Table 10.6. It should be noted that all but a select few of large tires available are capable of meeting the performance requirements imposed by the UHCT. That is, the inflation pressure, size, and weight of the candidate tires are nearly identical. As a result, the effects due to such variations might not be as apparent as they would be for other types of aircraft, where the selection is based on a larger pool of candidate tires.

Variations in MTOW have an obvious impact on the configuration of the landing gear and the pavement thickness. As a minimum, the structural dimensions of the landing gear and hence the structural weight would vary as the design weight of the aircraft changes between different configurations. As shown in Fig 10.1d, the landing gear weight fraction decreases even though the actual landing gear weight increases with the MTOW. This can be attributed to the fact that the landing gear weight does not increase with the MTOW in a pound-for-pound manner, and therefore a decreasing weight fraction is observed. Similarly, the landing gear weight decreases at a slower rate than the MTOW, yielding a higher weight fraction. The magnitude of the landing gear weight variation is similar to that provided by industry, where a 40-pound increase in the landing gear weight per 1,000 pounds increase in the MTOW is anticipated [App. A]. As reaffirmed in Table 10.7, an increase in the MTOW would require a thicker pavement to support the aircraft, and vice versa.

Table 10.6 Tire selection criteria, 24-tire configuration

<table>
<thead>
<tr>
<th>Subgrade strength</th>
<th>Thickness, in Min. press (Des./Base)</th>
<th>ACN Min. press (Des./Base)</th>
<th>Min. size (Des./Base)</th>
<th>Min. size (Des./Base)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flexible</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>73.1/73.5</td>
<td>133/134</td>
<td>135/134</td>
<td></td>
</tr>
<tr>
<td>Low</td>
<td>39.4/39.1</td>
<td>81/80</td>
<td>80/80</td>
<td></td>
</tr>
<tr>
<td>Medium</td>
<td>24.3/25.5</td>
<td>55/60</td>
<td>60/60</td>
<td></td>
</tr>
<tr>
<td>High</td>
<td>15.3/16.0</td>
<td>44/47</td>
<td>47/47</td>
<td></td>
</tr>
<tr>
<td>Rigid</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>18.3/18.6</td>
<td>92/96</td>
<td>96/96</td>
<td></td>
</tr>
<tr>
<td>Low</td>
<td>16.1/16.4</td>
<td>75/79</td>
<td>78/79</td>
<td></td>
</tr>
<tr>
<td>Medium</td>
<td>12.9/13.3</td>
<td>58/62</td>
<td>62/62</td>
<td></td>
</tr>
<tr>
<td>High</td>
<td>10.9/11.5</td>
<td>45/50</td>
<td>50/20</td>
<td></td>
</tr>
</tbody>
</table>
Although the location of aircraft cg has always played a decisive role in the positioning of the landing gear, instances are possible where design considerations become conclusive in deciding the mounting location, i.e., the landing gear has to be located at a specific location so that desired stability and maneuverability characteristics can be obtained. As shown in Fig. 10.1e, for this particular aircraft-landing gear combination, provided that the location of the main assembly group is fixed, an optimum aircraft cg location exists at a short distance aft of the current position where the weight fraction of the landing gear is at its minimum. In such cases, the location of the aircraft cg must be maintained at a particular position during takeoff and landing conditions through a controlled loading scheme. Once airborne, the constraints can be relaxed by redistributing the fuel among various fuel tanks.

As shown in Fig. 10.1f, the repositioning of the main assembly group in the aft direction results in a landing gear weight fraction that is lower than the one corresponding to a shift in the forward direction. This trend can be attributed to the reduced load that follows directly from an increased offset between the main assembly group and the location of the aircraft cg, i.e., a longer moment arm to counteract the applied ground loads. Note that when a highly-swept, high-aspect ratio wing is considered, a rearward movement of the main assembly group might be extremely difficult. Moving the gear aft could effect takeoff rotation speed and takeoff distance, which has to be checked. Also, brake weight may increase if the rotation speed increases, increasing the deceleration demands for the balanced field length requirement. Finally, the shift may not be feasible due to wing planform constraints, such as the size of the inboard trailing-edge extension (the Yehudi), required to provide suitable attachment location, as well as sufficient space to house the trailing-edge control surfaces and the associated actuation systems. The Yehudi also incurs drag and weight penalties that need to be considered.

The repositioning of the wing-mounted assemblies in the lateral direction affects primarily the stability and maneuverability characteristics of the aircraft. As shown in Table 10.8, an outboard movement of the wing-mounted assemblies produces a desired reduction in the sideways turnover angle; however, such a movement shifts the minimum 180-degree turn radius closer to the Class VI 100-foot upper limit [5]. As shown in Fig. 10.1g, the increasing landing gear weight fraction can be associated with the outboard movement of the assemblies. This leads to an increase in the length of the side strut, as well as an increase in the drag and shock struts due to wing dihedral, and hence the
structural weight of the landing gear. Conversely, an inboard movement of the assemblies exhibits a higher sideways turnover angle, a smaller turning radius, and a decreasing landing gear weight fraction.

Table 10.7 MTOW variations

<table>
<thead>
<tr>
<th>Subgrade strength</th>
<th>Thickness, in</th>
<th>ACN</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>-10,000 lb</td>
<td>+ 10,000 lb</td>
</tr>
<tr>
<td></td>
<td>(Des./Base)</td>
<td>(Des./Base)</td>
</tr>
<tr>
<td>Flexible</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>73.2/73.5</td>
<td>73.8/73.5</td>
</tr>
<tr>
<td>Low</td>
<td>39.0/39.1</td>
<td>39.3/39.1</td>
</tr>
<tr>
<td>Medium</td>
<td>25.4/25.5</td>
<td>25.6/25.5</td>
</tr>
<tr>
<td>High</td>
<td>16.0/16.0</td>
<td>16.0/16.0</td>
</tr>
<tr>
<td>Rigid</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>18.5/18.6</td>
<td>18.7/18.6</td>
</tr>
<tr>
<td>Low</td>
<td>16.3/16.4</td>
<td>16.5/16.4</td>
</tr>
<tr>
<td>High</td>
<td>11.5/11.5</td>
<td>11.5/11.5</td>
</tr>
</tbody>
</table>

Table 10.8 Wing-mounted assemblies location variations, lateral

<table>
<thead>
<tr>
<th>Design characteristics</th>
<th>20.0 in outboard</th>
<th>20.0 in inboard</th>
</tr>
</thead>
<tbody>
<tr>
<td>Sideways turnover angle, deg</td>
<td>38.4</td>
<td>43.2</td>
</tr>
<tr>
<td>Available touchdown angle, deg</td>
<td>16.9</td>
<td>16.5</td>
</tr>
<tr>
<td>Available takeoff rotation angle, deg</td>
<td>15.3</td>
<td>15.5</td>
</tr>
<tr>
<td>Turning radius, ft</td>
<td>80.1</td>
<td>76.7</td>
</tr>
</tbody>
</table>

Changes in the stability characteristics and ground clearance due to variations in landing gear strut length are of primary interest when a growth version of the aircraft is considered. Features typically associated with the growth options are a stretched fuselage obtained from the addition of plugs forward and aft of the wing, and upgraded power plants that come with a larger fan diameter. Both of the above features would require an extension of the strut length to maintain the desired operation angles and nacelle-to-ground clearance. As shown in Table 10.9, the growth-related modifications can result in an increased sideways turnover angle and a reduced permissible pitch angle during takeoff/landing operations. As can be expected and reaffirmed in Fig. 10.1h, an increase in strut length leads to an increase in structural weight, and therefore an increase in the landing gear weight fraction, as well as vice versa. The magnitude of the landing gear weight variation is again similar to the one
provided by industry, where a 60-pound increase in weight per strut is anticipated for every inch increase in strut length [App. A].

Changes in the size of the tires, wheels, and brakes due to varying design parameters, e.g., loading conditions and braking energy requirements, can alter the dimensions of the truck beam and axles. As can expected and reaffirmed by Figs 10.1i and 10.1j, an increase in the component length leads to a higher landing gear weight fraction, and vice versa. Data presented in Tables 10.10 and 10.11 show that an increase in either truck beam or axle length will result in a thicker pavement.

Table 10.9 Strut length variations

<table>
<thead>
<tr>
<th>Design characteristics</th>
<th>-3.0 in</th>
<th>+3.0 in</th>
</tr>
</thead>
<tbody>
<tr>
<td>Sideways turnover angle, deg</td>
<td>40.2</td>
<td>41.1</td>
</tr>
<tr>
<td>Available touchdown angle, deg</td>
<td>16.9</td>
<td>16.5</td>
</tr>
<tr>
<td>Available takeoff rotation angle, deg</td>
<td>15.3</td>
<td>15.5</td>
</tr>
</tbody>
</table>

Table 10.10 Truck beam length variations

<table>
<thead>
<tr>
<th>Subgrade strength</th>
<th>Thickness, in</th>
<th>ACN</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>-3.0 in</td>
<td>+3.0 in</td>
</tr>
<tr>
<td></td>
<td>(Des./Base)</td>
<td>(Des./Base)</td>
</tr>
<tr>
<td>Flexible</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>73.1/73.5</td>
<td>73.7/73.5</td>
</tr>
<tr>
<td>Low</td>
<td>39.1/39.1</td>
<td>39.2/39.1</td>
</tr>
<tr>
<td>Medium</td>
<td>25.5/25.5</td>
<td>25.5/25.5</td>
</tr>
<tr>
<td>High</td>
<td>16.0/16.0</td>
<td>16.0/16.0</td>
</tr>
<tr>
<td>Rigid</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>18.6/18.6</td>
<td>18.6/18.6</td>
</tr>
<tr>
<td>Low</td>
<td>16.4/16.4</td>
<td>16.4/16.4</td>
</tr>
<tr>
<td>Medium</td>
<td>13.3/13.3</td>
<td>13.3/13.3</td>
</tr>
<tr>
<td>High</td>
<td>11.5/11.5</td>
<td>11.5/11.5</td>
</tr>
</tbody>
</table>

10.4. Derivatives of the Baseline Aircraft

In today’s highly competitive environment, flexibility in being able to meet the vastly different requirements from various airline customers, e.g., a longer range and an extended payload capacity, has become one of the primary considerations in the design and marketing of a new aircraft. To ensure that a customer will have a list of options to select
from when it comes time to place an order, derivatives are considered early on in the conceptual design phase, and more than likely, pursued in parallel with the baseline aircraft.

Table 10.11 Axle length variations

<table>
<thead>
<tr>
<th>Subgrade strength</th>
<th>Thickness, in</th>
<th>ACN</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>-3.0 in (Des./Base)</td>
<td>+3.0 in (Des./Base)</td>
</tr>
<tr>
<td>Flexible</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>73.4/73.5</td>
<td>73.6/73.5</td>
</tr>
<tr>
<td>Low</td>
<td>38.7/39.1</td>
<td>39.5/39.1</td>
</tr>
<tr>
<td>Medium</td>
<td>25.1/25.5</td>
<td>25.8/25.5</td>
</tr>
<tr>
<td>High</td>
<td>15.7/16.0</td>
<td>16.3/16.0</td>
</tr>
<tr>
<td>Rigid</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>18.6/18.6</td>
<td>18.6/18.6</td>
</tr>
<tr>
<td>Low</td>
<td>16.4/16.4</td>
<td>16.4/16.4</td>
</tr>
<tr>
<td>High</td>
<td>11.5/11.5</td>
<td>11.5/11.5</td>
</tr>
</tbody>
</table>

Two derivatives were envisioned for the baseline UHCT: advanced (high aspect ratio) wing and extended range (8,000 nmi); corresponding configuration characteristics are shown in Table 10.12. Although the wing planform of the advanced wing derivative is slightly different from the baseline and the extended range version, it is assumed that the configuration of the landing gear on all three aircraft are identical, i.e., 24 main assembly tires on four struts. Note that this assumption does not imply that the weights of all three landing gear are identical.

Table 10.12 Derivative configuration characteristics

<table>
<thead>
<tr>
<th></th>
<th>Extended range</th>
<th>Advanced wing</th>
</tr>
</thead>
<tbody>
<tr>
<td>Passenger capacity</td>
<td>800</td>
<td>800</td>
</tr>
<tr>
<td>Range, nmi</td>
<td>8,000</td>
<td>7,500</td>
</tr>
<tr>
<td>Fuselage length, ft</td>
<td>250.0</td>
<td>250.0</td>
</tr>
<tr>
<td>Fuselage width, ft</td>
<td>24.0</td>
<td>24.0</td>
</tr>
<tr>
<td>Wing span, ft</td>
<td>264.0</td>
<td>261.0</td>
</tr>
<tr>
<td>Wing area, ft²</td>
<td>8,324</td>
<td>7,423</td>
</tr>
<tr>
<td>Aspect ratio</td>
<td>8.4</td>
<td>9.2</td>
</tr>
<tr>
<td>MTOW, lb</td>
<td>1,350,000</td>
<td>1,140,000</td>
</tr>
<tr>
<td>Fuel, lb</td>
<td>640,000</td>
<td>460,000</td>
</tr>
</tbody>
</table>

As shown in Figure 10.2, the advanced wing derivative has the highest landing gear weight fraction of the three configurations, whereas the extended range derivative has the
lowest of the three. For identical mission requirements between the baseline and the advanced wing derivative, the baseline aircraft will be the preferred choice if the deciding factor is based on landing gear weight fraction, its lower landing gear weight fraction implies that a greater fraction of the total aircraft weight is made up by revenue-generating payloads. However, if the deciding factor is something other than the landing gear weight faction, e.g., operating cost or runway upgrade cost, the advanced wing configuration will be the preferred choice due to its lower mission fuel requirements and lighter MTOW, respectively. As for the extended range derivative, although the landing gear weight fraction is lower than the other two aircraft, the required pavement thickness as shown in Table 10.13 can result in a prohibitive runway upgrade cost. However, the desired flotation characteristics can be obtained by replacing the conventional wing design with the one found on the advanced wing derivative. The reduction in mission fuel weight associated with higher performance due to the advanced wing design would then lower the MTOW of the extended range derivative and hence the required pavement thickness.

![Figure 10.2 Changes in landing gear weight fraction due to aircraft configuration variations](image)

10.5. *Landing Gear Weight Trend for Large Aircraft*

The baseline aircraft along with its derivatives are used to provide some analytically-based landing gear weight estimates that can be used to help calibrate existing statistical weight equations. Although statistical weight equations are capable of producing quick and fairly accurate group weights within the range where significant previous experience is available, their reliability is questionable at best for aircraft with takeoff weight beyond one million pounds, i.e., they are constrained by what has been designed in the past. The uncertainty is
made evident by the two possible weight trends available: a decreasing trend as predicted by ACSYNT and an increasing trend as predicted by Douglas and Torenbeek. As shown in Fig. 10.3, landing gear weight fractions corresponding to the baseline aircraft and its derivatives suggest that the weight equation used by ACSYNT is likely to produce a more accurate trend than the ones used by Douglas and Torenbeek. In addition, an increase in the number of main assembly struts from four to six did not result in a step increase in the weight fraction as expected. Again, this can be attributed to the decrease in the number of tires found on each strut, which effectively lowered the combined load on the structural members and therefore led to a lighter structure. Note that additional aircraft within the UHCT class must be modeled to extend the database so that the weight trends as observed here may be confirmed.

Table 10.13 Aircraft configuration variations

<table>
<thead>
<tr>
<th>Subgrade strength</th>
<th>Thickness, in</th>
<th>ACN</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Ext. range (Des./Base)</td>
<td>Adv. wing (Des./Base)</td>
</tr>
<tr>
<td>Flexible</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ultra-low</td>
<td>77.1/73.5</td>
<td>70.0/73.5</td>
</tr>
<tr>
<td>Low</td>
<td>40.8/39.1</td>
<td>37.9/39.1</td>
</tr>
<tr>
<td>Medium</td>
<td>25.5/25.5</td>
<td>24.6/25.5</td>
</tr>
<tr>
<td>High</td>
<td>15.6/16.0</td>
<td>15.6/16.0</td>
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<tr>
<td>Rigid</td>
<td></td>
<td></td>
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<tr>
<td>Ultra-low</td>
<td>19.3/18.6</td>
<td>18.2/18.6</td>
</tr>
<tr>
<td>Low</td>
<td>16.9/16.4</td>
<td>16.1/16.4</td>
</tr>
<tr>
<td>Medium</td>
<td>13.6/13.3</td>
<td>13.2/13.3</td>
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<tr>
<td>High</td>
<td>11.6/11.5</td>
<td>11.6/11.5</td>
</tr>
</tbody>
</table>
Figure 10.3 Landing gear weight fraction beyond one million pounds MTOW
Chapter 11  Costs

11.1. Introduction

The manufacturing cost of the landing gear cannot be treated simply as a function of weight or strut length. Instead, cost estimation must take into account the costs of development, certification, marketing, life-cycle, spares, etc. A typical program cost is roughly in the range of $10 to $12 million dollars, based on industry survey [App. A]. However, detailed information is considered proprietary and is difficult to obtain from the manufacturers. Thus, the cost issue will only be discussed in qualitative terms, while actual unit costs will be provided whenever available.

11.2. Maintenance and Overhaul

The maintenance costs associated with the landing gear represent a considerable item in the total maintenance bill [3]. The cost of the tires, wheels and brakes will remain relatively unchanged for new programs. The limiting factor is the size of the tire that can be constructed and tested without a major new investment in manufacturing and testing facilities. Current hardware limits the maximum diameter to 56 inches for the bias-ply tire and 58 inches for the radial-ply tire [App. A]. Dimensions and costs of several tires found on existing large aircraft are listed in Table 11.1. For the aluminum wheel and carbon-carbon heat sink found on the Boeing Model 747-400, the unit price is valued at $70,000.

Table 11.1  Description of selected aircraft tires [App. A]

<table>
<thead>
<tr>
<th>Tire</th>
<th>Type</th>
<th>Aircraft</th>
<th>Application</th>
<th>Cost, $</th>
</tr>
</thead>
<tbody>
<tr>
<td>H49x19.0-22, 32-ply</td>
<td>Bias</td>
<td>Boeing Model 747</td>
<td>Main/Nose</td>
<td>2,100</td>
</tr>
<tr>
<td>42x17.0-18, 28-ply</td>
<td>Radial</td>
<td>Boeing Model 777</td>
<td>Nose</td>
<td>2,100</td>
</tr>
<tr>
<td>50x20.0-22, 32-ply</td>
<td>Radial</td>
<td>Boeing Model 777</td>
<td>Main</td>
<td>2,900</td>
</tr>
</tbody>
</table>

The landing gear overhaul interval varies between 33,000 to 42,000 flight hours, or roughly within six years [App. A]. Generally, the parts of a landing gear are given an ultimate ‘safe life’ beyond which they would, if still in service, be scrapped [57]. A justification of this approach is that deterioration in service can go unseen since corrosion
and other process can occur in concealed areas which are only revealed when the assembly is completely stripped down.

The preferred method is to overhaul the entire set at the same time to minimize the down-time; however, it might be necessary to overhaul the set separately due to schedule, parts and facility constraints. Components may require extensive rework in the shops and thus it is difficult to quote a total throughput time. Given a supply of serviceable components to replace those sent shop-to-shop, it is possible to turn around a B747 assembly within five weeks [57]. Due to the length of time required to rectify each constituent part of a particular assembly, a unit nearly always loses its identity as such, and the end product may contain only a few parts of the original assembly. However, it is noted that when refurbished, the assembly may be better than a new one since it embodies modifications designed to increase the subsequent overhaul life [57]. For the B747 type landing gear, the overhaul cost is estimated at $400,000 [App. A]. Replacement of the carbon heat sink occurs every 1,200 to 1,500 landings, while only 300 landings are allowed for the wheel before replacement. The overhaul cost for the wheel and heat sink is pre-negotiated with the contractors and is known as cost-per-landing. Quoting the B747 figures, the cost for the wheel, including tire, is estimated at $5 per landing, while the cost for the carbon-carbon heat sink is estimated at $10 per landing [App. A].

11.3. Cost Reduction

With the financial challenges arising from the deregulation of the air-travel industry, the airlines are faced with the challenge of reducing operating costs to remain competitive. As a result, the airlines have demanded that the aircraft manufacturers produce new designs with high reliability and low maintenance requirements. In basic design, costs associated with the landing gear may be reduced by aiming at simplicity, compactness, and minimum weight and maintenance requirements. Simplified design and improved manufacturing techniques, e.g., die-forging and three-dimensional machining [9], are being used to reduce the part-count associated with the landing gear system. In addition, recent technologies, e.g., carbon-carbon heat sinks, radial tires, and high-strength steel, are being introduced. Potential savings associated with the application of these technologies have already been mentioned in Chapter Four.
Chapter 12  Future Considerations

Although an initial validation of the methodology was done, further validation and is required. Since all readily available data was used in this study, the methods should be checked against more thorough analysis done by industry on actual new products, such as the new Boeing 747 derivative, as they appear. In addition, several areas where refinements could be made have been identified as a result of the experience to date. However, we are not able to quantify the benefits that these extensions in the methodology would produce.

Refinement of the landing gear analysis package should include the improvement of the pavement thickness and landing gear weight predictions. A method to calculate the rigid pavement bearing stress that includes location and direction of maximum moment considerations [36] would improve the reliability of the estimated rigid pavement thickness and the corresponding ACN. The experimental test program being conducted by the FAA and Boeing to determine the exact flotation requirements for the B777 may provide useful information for this extension. The accuracy of the landing gear structural weight can be improved by extending the analysis to include intermediate column buckling analysis [48] for structural members with large slenderness ratio, e.g., drag and side struts, and St. Venant’s theory for torsion and flexure of thick-walled bars [50] for structural members with low machinability factors, e.g., axle and truck beam.

Finally, the full potential of the analysis package would emerge if a graphical front-end and the Dynamic Integration System (DIS)-based wrapping technique [58] were incorporated. The former would enable the user to interactively prepare input for the analysis and interpret the output, while the latter would provide a common interface such that coordinated execution of disciplinary analyses as found in ACSYNT can be achieved.
Chapter 13  Conclusions

The design of the landing gear is one of the more fundamental aspects of aircraft design. The design and integration process encompasses numerous engineering disciplines, e.g., structure, weights, runway design, and economics, and has become extremely sophisticated in the last few decades. These considerations were incorporated in an MDO procedure for use in the conceptual design of large transport aircraft. Accomplishments include:

- Aircraft cg estimation methods were studied and a new approach to cg estimation in conceptual design was demonstrated.
- An automated landing gear modeling algorithm for large transport aircraft was developed, and conformance with typical FAR requirements was assessed automatically.
- Airfield compatibility considerations associated with pavement thickness and runway and taxiway dimensions were automated.
- An analytical structural weight estimation procedure was developed to complement existing statistical landing gear weight estimation methods.
- A multidisciplinary analyses computer program package for landing gear design and was created for use in large MDO aircraft design programs.
- Results obtained from the analysis package were presented, illustrating the trade-off studies and parametric results available for incorporation into a complete MDO design procedure.
References


Appendix A  Industry Survey

To try to ensure that our work was current and relevant, we made many contacts with industrial and government engineers that work with landing gears. The approach was to conduct a survey. In this section we present the material given to them, and report on what we found. In general, we got the best information in telephone interviews. The questions initiated discussions that were often broader and less focused than the questions themselves. Thus, the discussion of results presented in the following sections follows the broader areas, and does not explicitly summarize individual answers to the questions. In general, the company contacts were not able to give us detailed written material because they considered their expertise proprietary.

The issues we identified that needed to be addressed were: runway compatibility, landing gear integration, landing gear configuration, landing gear weight, advanced technologies, and cost. A list of questions was developed covering these considerations to ask engineers associated with landing gear systems. Using a few suggestions from contacts in industry and government, we started making calls. In some cases, we sent a fax of our questions. Often, we were directed to contact someone else in the organization, or, someone at another company. Eventually, the survey included major airframers, landing gear manufacturers, airlines, and government agencies and technical societies. The list of questions was circulated among the manufacturers for comments and suggestions, while airlines were contacted to obtain operating and maintenance cost information.

A.1. General script for our phone interviews

The landing gear integration issue for advanced aircraft is being investigated under a NASA Ames research grant to Virginia Tech. The project objective is the
formulation of a methodology to include landing gear considerations explicitly in the conceptual design stage. In particular, the project addresses the special design considerations associated with the next-generation high-capacity transport with a TOGW exceeding one million pounds. Our landing gear design and integration related issues were defined during the initial background research with heavy reliance on N. S. Currey's *Aircraft Landing Gear Design: Principles and Practices*. We have questions concerning landing gear configuration, aircraft-landing gear integration, runway compatibility, advanced technologies, weight, maintenance, and cost.

A.2 The questions

• What are the design parameters given to the landing gear designer? What is the design envelope you have to work with? Which is the primary design goal, minimum weight, stowage space, or complexity?

• What are the major problems encountered concerning the integration of the landing gear for the ultra high capacity type aircraft currently under study? What kind of special design considerations are required?

• What are some advanced technologies that will change the landing gear configuration of the ultra high capacity type aircraft dramatically in the next decade or two? How will they change the configuration? What kind of weight reduction can be expected with these technologies?

• What method is used to calculate the landing gear ground and landing loads? Which specification is used? Is there a set of equations that can be readily used?

• What method is used to calculate the aircraft flotation requirements? How do you account for multiple main strut configurations? What kind of constraint in gear configuration is imposed by the flotation requirements?

• For a takeoff gross weight outside the experience base are there some “first principles” that can be followed for landing gear weight estimation?

• What will be the most likely landing gear configuration for the ultra high capacity type aircraft? How many main struts can be expected for a takeoff weight exceeding one million pounds? How would you arrange the main assembly if you have six main struts? What is the major advantage/disadvantage of increasing the number of main struts?

• What method is used to produce the initial landing gear weight estimation? What would be the scaling factor if we are to estimate the weight by scaling up current configurations to meet the demand? Can we obtain geometry and weight information on existing landing gears to be used as a design database?
• What method is used for the initial landing gear cost estimate? What are the major cost drivers and the corresponding sensitivities?

A.3. The Contacts

A list of survey participants and their telephone numbers are presented in Table A.1.

A.4. Findings

A.4.1 Runway Compatibility

Due to economic considerations, the ultra high capacity transport, UHCT, must be able to operate out of Class V airports, e.g., the Boeing Model 747 class airports, without requiring extensive runway reinforcement and modification. Flotation requirements can be obtained using the PCA methods for rigid pavement and the CBR method for flexible pavements. Effects of multiple-strut/multiple-wheel landing gear configurations on the pavement bearing strength have yet to be addressed fully by industry. However, preliminary finite element analyses suggest interaction among wheels can be neglected outside a radius of ten footprint radii from the point where the flotation analysis is performed. Based on this information, the number of wheels, i.e., the equivalent number of wheels per strut (ENWS), used to select the proper repetition factor curve (this is the factor that accounts for the number of landings per year on the pavement) becomes the number of wheels found within the circle of ten foot-print radii centered at the strut-truck joint. With current tire inflation pressures, a 20-wheel main assembly is required for a TOGW between 1 and 1.2 million pounds, while a 24-wheel main assembly is required for a TOGW between 1.3 and 1.6 million pounds to produce the desired flotation characteristics. Both numbers include a 20 percent future growth factor.
<table>
<thead>
<tr>
<th>Company/Committee</th>
<th>Contact Name</th>
<th>Phone</th>
<th>Fax</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Federal Aviation Administration</strong></td>
<td>John Rice</td>
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<td>(202) 267-5383</td>
</tr>
<tr>
<td></td>
<td>Niel Schalekanp</td>
<td>(206) 227-2112</td>
<td></td>
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<tr>
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<td>(206) 227-1320</td>
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<tr>
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<td></td>
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<tr>
<td></td>
<td>Henry Pollack</td>
<td>(513) 255-4158</td>
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<td><strong>SAE A-5 Committee</strong></td>
<td>Richard Vandame</td>
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<td>(412) 776-0002</td>
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<tr>
<td><strong>Waterways Experiment Station</strong></td>
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<tr>
<td><strong>The Tire &amp; Rim Association, Inc.</strong></td>
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<td></td>
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<td>(310) 496-9244</td>
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<td>Larry McBee</td>
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<td></td>
<td>Richard Luu</td>
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<td><strong>Michelin</strong></td>
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<td>Ron Olds</td>
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<td><strong>U.S Air</strong></td>
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<td>(412) 747-3975</td>
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<td></td>
<td>Ed Pozzi</td>
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<tr>
<td><strong>Northwest</strong></td>
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<tr>
<td></td>
<td>Steve Lydon</td>
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<td>(612) 726-6844</td>
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A.4.2 Integration

Aircraft-landing gear integration will be the primary concern for the next-generation high-capacity transports. The location dependency of the wing and the main gear assembly to the aircraft cg will play a major role in the integration issue. With the introduction of multiple-strut configurations, the envelope within which the landing gear has to be located to produce the ideal loading and stability characteristics may no longer be large enough to accommodate the increased number of main assembly struts. This phenomenon is known as location stagnation by the landing gear community. Modification in design and flotation requirements must be made, if necessary, to accommodate kinematic and stowage constraints such that the landing gear can be deployed and stowed without interference with surrounding structures. A forward-retracting scheme for the fuselage struts is preferred, which allows the gears to be deployed using the slip-stream in case of a hydraulic failure. However, stowage limitations could result in an aft-retracting scheme for the center-line strut located between the wing-mounted struts in a five-plus struts configuration.

A.4.3 Configuration

The number of wheels imposed by the flotation requirement can be accommodated with either a four-, five- or six-strut configuration. One of the centerline struts will be located abreast of the wing-mounted struts for the five-plus main gear struts configurations. With the introduction of the centerline strut(s), a double-keel layout is required, i.e., the stowage space is divided into three compartments with two identical keels placed parallel to each other. The centerline strut(s) will then be mounted and stowed between the keels. The fuselage width of the new aircraft, which will be 20 to 30 inches wider than that of the B747, should be able to accommodate the double-keel layout with relative ease. However, one of the drawbacks is that the structural weight associated with the keels will be doubled, since both keels have to withstand the same
buckling load and thus have to be similar in dimension to the one found in the single-keel layout. Another drawback is that a complex deploying/retracting scheme for the landing gear doors must be developed to prevent interference between the doors and the gear itself.

The length of the strut will be dictated by the condition on aircraft ground clearance requirements during cross-wind landings imposed by the large nacelle diameter of the advanced engines. The vertical spacing between the nacelle and the wing, i.e., the gully, will be reduced to a minimum, provided that desirable flow characteristics are maintained, before any extension in strut length is made. A main gear steering system will be needed to meet the ground operation requirements, with the most demanding maneuver being the 180-degree turn on existing runways. Options include the fuselage strut steering system found on the B747 and the forward-aft wheel steering system found on the B777.

The wheel truck dimensions of the dual-twin-tandem and triple-dual-tandem configurations will be similar to those of the B747 and B777, respectively. The longitudinal spacing between tires will be maintained at roughly six inches for ease of removal of the wheel plugs, while lateral spacing will be slightly wider in both configurations due to the increased brake size required for the new aircraft. Due to the limited stowage volume, the truck assembly might have to be rotated prior to retraction to minimize the stowage space required.

A.4.4 Loads

The dynamic and ground loads are determined in accordance with FAR Part 25. It is unlikely that the new aircraft will be subjected to rough field operating requirements, and thus a single-acting shock absorber will be sufficient to handle the kinetic energy experienced during landing and taxiing. Based on preliminary analysis from industry, the
new aircraft will require a shock strut with a 24-inch stroke at the minimum, a piston
diameter of 15 inches, and internal oleo pressures between 1,500 and 1,800 psi. Canting
of the strut should be avoided, if possible, due to the load path considerations. Active
struts will likely be used to provide improved and acceptable ground ride quality.
Improvements will probably be internal, e.g., bearings, finishes, and rebound damping,
but little difference will be seen in the external configuration.

A.4.5 Weight

The design of the new landing gear must be as simple as possible, since
complexity drives up the cost faster than weight. However, weight also appears to be
inversely proportional to the level of complexity. With the reduction in the complexity
level, e.g., the number of supports, structural members are forced to withstand a higher
load, which in turn increases the structural weight due to an increase in cross-sectional
area. Therefore, a balance must be reached between simplicity and weight, and this can
only be accomplished through parametric studies of different landing gear configurations.
Note that a step increase in total landing gear weight occurs with each additional strut.
Therefore, the number of struts must be kept at a minimum while at the same time
meeting the flotation and simplicity requirements. Existing data indicates that fuselage
strut weight is roughly 25 to 40 percent less than that of the wing strut, and overall, total
gear weight will remain at roughly five percent of the maximum take-off weight.

Structural weight estimation should be obtained using an analytical approach,
while the following “rules of thumb” for sensitivities were provided by the industrial
contacts. Weight scaling taken to a 1.1 power will give a reasonable estimation for sub-
components, i.e., the steering system, up locks, down locks, fittings and miscellaneous
items. A landing gear gross weight variation of 5 pounds per 1,000 pounds increase in
TOGW for the nose gear was suggested, while a 40-pound variation per 1,000 pounds
increase in TOGW for the main gear should be used. Weight variation of 40 pounds per
inch increase in strut length per strut was also suggested. The wheel and tire weights will
be similar to that of the B747, i.e., 190 pounds and 290 pounds, respectively, while the heat sink weight will be heavier, again due to the increased braking energy requirements. A step increase in the landing gear group weight will occur with each additional strut; therefore, the number of struts should be kept at a minimum.

A.4.6 Advanced Technologies

Advanced technologies will play a major role in reducing the weight of the UHCT type landing gears. A five to seven percent weight reduction can be obtained with the use of high strength steel for the landing gear strut and carbon for the brake. Radial-ply tires, although having a higher initial cost, offer a 20 percent weight reduction over bias-ply tires, while at the same time allowing more landings per life-cycle. Further weight reduction can be achieved by the use of a steer-by-wire concept in place of the conventional cable-and-pulley system. Electrical actuation units will be introduced as a way to reduce weight in secondary mechanisms, but the primary actuation method will remain hydraulic.

A 4.7 Cost

The manufacturing cost of the landing gear cannot be treated simply as a function of weight or strut length. Instead, cost estimation must take into account the costs of development, material and processes, certification, marketing, overhaul, refurbishment, and spares. Typical program cost is roughly in the range of $10 to $12 million dollars. The cost of the tire, wheel and brake will remain relatively unchanged. The limiting factor is the size of the tire that can be constructed and tested without a major new investment in the manufacturing and testing facilities. Current hardware limits the maximum diameter to 56 inches for the bias-ply tire and 58 inches for the radial-ply tire. The H49x19.0-22, a 32-bias-ply tire found on both the nose and main gear of the B747, is valued at $2,100. This can be compared to the radial, the 50x20.0-22, which is found on the main gear of the B777 with a 32-ply rating, which is valued at $2,900, and the
42x17.0-18, which is found on the nose gear of the B777 with a 28-ply rating, and is valued at $2,100. Due to its light weight and the increased number of landings allowed per life-cycle, the radial tire has become the preferred choice by airlines even though it costs more. As for the aluminum wheel and carbon-carbon heat sink found on the B747, the unit price is valued at $70,000.

The landing gear overhaul interval varies between 33,000 to 42,000 flight hours. The preferred method is to overhaul the entire set at the same time to minimize the downtime. However, it might be necessary to overhaul the set separately due to schedule, parts and facility constraints. For the B747 type landing gear, the overhaul cost is estimated at $400,000. Replacement of the carbon heat sink occurs every 1,200 to 1,500 landings, while only 300 landings are allowed for the wheel before replacement. The overhaul cost for the wheel and brake is pre-negotiated with the contractors and is known as cost-per-landing. Quoting the B747 figures, the cost for the carbon-carbon brake is estimated at $10 per landing, while the cost for the wheel, including tire, is estimated at $5 per landing.

To conclude, due to the competition among the airframe and landing gear manufacturers, landing gear design procedures, and weight and cost data are considered to be company-proprietary. As a result, the majority of the survey participants were only willing to address the issues in general terms. However, the survey results did provide some useful insights to the design of the landing gear, and reaffirmed design and analysis procedures as previously documented.
Appendix B  Structural Analysis Derivations

B.1. Introduction

Detailed derivations of selected structure analyses as introduced in Chapter Eight are compiled and presented in the following sections. The sections outline the procedures used to determine the internal actions for the nose assembly and the trunnion, both of which involve statically-indeterminate structures, as well as stress calculation for thin-walled circular tubes. These items involve derivations of the basic equations that cannot be presented in a concise manner within the main text.

B.2. The Nose Assembly

The reactions at the supports of the truss that represents the nose gear cylinder/drag/side struts structure, and consequently the internal actions, can be determined by Castigliano’s theorem [46, p. 611], that is,

\[ u_j = \frac{\partial U}{\partial P_j} = \sum_{i=1}^{n} \frac{F_i}{A_i E} \frac{\partial F_i}{\partial P_j} \]  \hspace{1cm} (B.1)

where \( u_j \) is the deflection at the point of application of the load \( P_j \), \( E \) is the modulus of elasticity, and \( l, F, \) and \( A \) are the length, internal force, and cross-sectional area of each member, respectively. The above theorem gives the generalized displacement corresponding to the redundant, \( P_j \), which is set equal to a value compatible with the support condition. This permits the solution of the redundant, and consequently all remaining internal actions, via equilibrium.
As shown in Figure B.1, the port side strut* is designated as redundant and released from its support at point \( K \). Using Eq. (B.1) the deflection at point \( K \) can be written as

\[
y_K = \frac{F_{10}l_{10}}{A_{10}E} \frac{\partial F_{10}}{\partial R_K} + \frac{F_{10}l_{10}}{A_{10}E} \frac{\partial F_{10}}{\partial R_K} + \frac{F_{K0}l_{K0}}{A_{K0}E} \frac{\partial F_{K0}}{\partial R_K}
\]  

(B.2)

![Free-body diagram of the nose gear structure in the yz-plane](image)

* This is the strut on the right. You are looking aft in this figure.
where $S$ and $V$ are the applied side and vertical force, respectively, and $\theta_{\text{drag}}$ and $\theta_{\text{side}}$ are the angles between the axial centerlines of the drag and side struts and the $xy$-plane, respectively. Differentiating Eqs (B.3a, b, and c) with respect to $R_K$ results in

$$\frac{\partial F_{JO}}{\partial R_K} = 1$$  \hspace{1cm} (B.4a)

$$\frac{\partial F_{JO}}{\partial R_K} = -\sin \theta_{\text{side}}$$  \hspace{1cm} (B.4b)

$$\frac{\partial F_{KO}}{\partial R_K} = 1$$  \hspace{1cm} (B.4c)

To determine the reaction at point $K$ and subsequently the internal force in each structural component, substitute the relationships of Eqs (B.3a, b, and c) and (B.4a, b, and c) back into Eq. (B.2), apply the no-deflection condition and then solve for $R_K$.

**B.3. The Trunnion**

The trunnion model shown in Figure 8.9 is repeated here as Figure B.2. As shown in Figure B.2, the trunnion is subjected to a force with components $F_x$, $F_y$, and $F_z$, and a couple with moment components $C_y$ and $C_z$, at axial position $x = l_1$, where $0 < l_1 < L$ and $0 \leq x \leq L$. Clamped end-conditions at $x = 0$ and $x = L$ yield ten homogeneous conditions, five at each end. At the load point $x = l_1$, there are five continuity conditions, i.e., $u$, $v$, $w$, $v'$, and $w'$, and five jump conditions corresponding point-wise equilibrium of the internal actions and the external loads. These twenty conditions are

$$u_1(0) = u_2(L) = 0$$  \hspace{1cm} (B.5a)

$$v_1(0) = v_1'(0) = v_2(L) = v_2'(L) = 0$$  \hspace{1cm} (B.5b)

---

**A slight elaboration: the $\theta_{\text{drag}}$ is the angle between the drag strut and the $x$-$y$ plane. It, along with the drag strut, is not shown in Fig. B-1 because the attachment location of the drag strut is below the side strut attachment and the figure only represents the structural arrangement at a distance above that point.**
\[ w_1(0) = w_1'(0) = w_2(L) = w_2'(L) = 0 \] (B.5c)

\[ u_1(l_1) = u_2(l_1) \] (B.6a)

\[ v_1(l_1) = v_2(l_1) \] (B.6b)

\[ w_1(l_1) = w_2(l_1) \] (B.6c)

\[ \frac{dv_1(l_1)}{dx} = \frac{dv_2(l_1)}{dx} \] (B.6d)

\[ \frac{dw_1(l_1)}{dx} = \frac{dw_2(l_1)}{dx} \] (B.6e)

\[ -N_{x1}(l_1) + N_{x2}(l_1) + F_x = 0 \] (B.7a)

\[ -V_{y1}(l_1) + V_{y2}(l_1) + F_y = 0 \] (B.7b)

\[ -V_{z1}(l_1) + V_{z2}(l_1) + F_z = 0 \] (B.7c)

\[ -M_{z1}(l_1) + M_{z2}(l_1) + C_z = 0 \] (B.7d)

\[ -M_{y1}(l_1) + M_{y2}(l_1) + C_y = 0 \] (B.7e)

Figure B.2 Trunnion modeled as a clamped-clamped end bar
In the \(xz\)-plane, equilibrium gives

\[
\frac{dM_y}{dx} - V_z = 0 \tag{B.8}
\]

and

\[
\frac{dV_z}{dx} = 0 \tag{B.9}
\]

where \(M_y\) and \(V_z\) are the internal moment and shear components, respectively. Given the equation of elastic curve as

\[
M_y = EI_{yy} \left( -\frac{d^2 w}{dx^2} \right) \tag{B.10}
\]

where \(E\) is the modulus of elasticity and \(I_{yy}\) is the second area moment about the \(y\)-axis, Eqs (B.8) and (B.9) become

\[
V_z = EI_{yy} \left( -\frac{d^3 w}{dx^3} \right) \tag{B.11}
\]

and

\[
EI_{yy} \left( -\frac{d^4 w}{dx^4} \right) = 0 \tag{B.12}
\]

Integrating Eq. (B.12) four times with respect to \(x\) results in

\[
w_1 = \frac{c_1}{6} x^3 + \frac{c_2}{2} x^2 + c_3 x + c_4 \quad 0 \leq x \leq l_1 \tag{B.13a}
\]

\[
w_2 = \frac{c_5}{6} x^3 + \frac{c_6}{2} x^2 + c_7 x + c_8 \quad l_1 \leq x \leq L \tag{B.13b}
\]

To determine \(w_1\) and \(w_2\) at either end of the trunnion, boundary conditions as given in Eqs (B.5b and c) and (B.6b, c, d, and e) were used to solve for the eight integration constants \((c_i)\) in Eqs (B.13a and b). Finally, substitute \(w_1\) and \(w_2\) back into Eqs (B.10) and (B.11) and use the static boundary conditions as given in Eqs (B.7a, b, c, and d) to obtain the
internal shear force and bending moment, respectively. The same procedure is used to determine \( v(x) \) and the internal actions in the \( xy \)-plane.

In the longitudinal direction, equilibrium gives

\[
\frac{dN}{dx} = 0 \quad (B.14)
\]

where \( N \) is the axial force. In addition, the material law gives

\[
N = EA \frac{du}{dx} \quad (B.15)
\]

where \( A \) is the cross-sectional area. Since the axial force is spatially uniform, or piecewise constant, integrating Eq. (B.15) once with respect to \( x \) results in

\[
u_1 = \frac{N_1}{EA} x + c_9 \quad 0 \leq x \leq l_1 \quad (B.16a)
\]

\[
u_2 = \frac{N_2}{EA} x + c_{10} \quad l_1 \leq x \leq L \quad (B.16b)
\]

where the two integration constants \( c_9 \) and \( c_{10} \) are determined using the boundary conditions as given in Eq. (B.6a). Finally, substitute \( u_1 \) and \( u_2 \) back into Eq. (B.15) and sum the forces in the \( x \) direction at \( x = l_1 \) to obtain the internal axial force.

**B.4. Normal and Shear Stresses In a Thin-walled Tube**

The normal stresses induced on the structural members are determined by combining the effects of axial load and combined bending, while the shear stresses are determined by combining the effects of torsion and shear forces due to bending [47].

The normal stress \( (\tau_{xx}) \) due to combined axial force and bending moments is given as

\[
\tau_{xx} = \frac{N}{A} + \frac{M_y}{I_{yy}} z - \frac{M_z}{I_{zz}} y \quad (B.17)
\]
For a thin-walled circular tube referred to polar coordinates as shown in Figure B.3, the principal centroidal second area moments about the $y$- and $z$-axes are

$$I_{yy} = I_{zz} = \frac{1}{2} (r \sin \theta)^2 trd\theta = \pi r^3 t$$  \hspace{1cm} (B.18)

where $r$ is the mean radius and $t$ is the wall thickness. Given the relationship of Eq. (B.18), Eq. (B.17) becomes

$$\tau_{xx} = \frac{N}{A} + \frac{1}{\pi r^2 t} (M_y \sin \theta - M_z \cos \theta)$$  \hspace{1cm} (B.19)

Differentiate Eq. (B.19) with respect to $\theta$ to get

$$\frac{d\tau_{xx}}{d\theta} = \frac{1}{\pi r^2 t} (M_y \cos \theta + M_z \sin \theta)$$  \hspace{1cm} (B.20)

and at the extremum, i.e., $d\tau_{xx}/d\theta = 0$, so that

$$\tan \theta_{max} = -\frac{M_y}{M_z}$$  \hspace{1cm} (B.21a)

$$\sin \theta_{max} = \frac{M_z}{\sqrt{M_y^2 + M_z^2}}$$  \hspace{1cm} (B.21b)

$$\cos \theta_{max} = -\frac{M_y}{\sqrt{M_y^2 + M_z^2}}$$  \hspace{1cm} (B.21c)

Given the relationships of Eqs (B.21b and c), the extremum values of the bending normal stresses are determined using the expressions

$$\tau_{xx, \text{bending}}(\theta_{max}) = -\frac{1}{\pi r^2 t} \sqrt{M_y^2 + M_z^2}$$  \hspace{1cm} (B.22a)

and

$$\tau_{xx, \text{bending}}(\theta_{max} + \pi) = \frac{1}{\pi r^2 t} \sqrt{M_y^2 + M_z^2}$$  \hspace{1cm} (B.22b)

Thus, the extremum values of the normal stress on a circular-tube cross section under combined axial and bending actions are
\[ \tau_{xx_{\text{max}}} = \frac{N}{A} \pm \frac{1}{\pi r^2} \sqrt{M_y^2 + M_z^2} \]  

(B.23)

Figure B.3 Annular section showing positive shear forces and bending moments

The shear stress \( \tau_{xs} \) due to combined transverse shear force and torque is given as

\[ \tau_{xs} = \frac{q(s)}{t} + (\tau_{xs})_{\text{torque}} \]  

(B.24)

where \( q \) is the shear flow due to bending. As indicated by in Figure B.4, the shear flow from some arbitrary origin to any point round the cross-section of a circular tube for axial equilibrium is

\[ q = q_0 - \frac{dF}{dx} \]  

(B.25)

where

\[ \frac{dF}{dx} = \frac{d}{dx} \int_0^s \tau_{xx} \, ds = \int_0^\theta \frac{d\tau_{xx}}{dx} \, trd\theta \]  

(B.26)
Given that

\[
\frac{dN}{dx} = 0 \tag{B.27}
\]

and

\[
\frac{dM_y}{dx} = V_z \tag{B.28}
\]

and

\[
\frac{dM_z}{dx} = -V_y \tag{B.29}
\]

rearrangement of Eq. (B.26) results in

\[
\frac{dF}{dx} = r^2 \left[ \frac{V_z}{I_{yy}} (1 - \cos \theta) + \frac{V_y}{I_{zz}} \sin \theta \right] \tag{B.30}
\]

From the relationships of Eqs (B.25) and (B.30), the integral of \( q \) round the cross section is

\[
\oint \! q r \, ds = \oint \! q_0 r \, ds - \int \frac{dF}{dx} \, r \, ds \tag{B.31}
\]

Since only bending is considered in this case, the left-hand side of Eq. (B.31) is zero, that is,
and the integration results in
\[ q_0 = r^2 \frac{V_z}{I_{yy}} \] (B.33)

Given the relationships of Eqs (B.18), (B.30) and (B.33), the magnitude of the shear flow is determined using the expression
\[ q = \frac{1}{\pi r} \left( V_z \cos \theta - V_y \sin \theta \right) \] (B.34)

Differentiating Eq. (B.34) with respect to \( \theta \) gives
\[ \frac{dq}{d\theta} = -\frac{1}{\pi r} \left( V_y \cos \theta + V_z \sin \theta \right) \] (B.35)

and at the extremum, i.e., \( dq/d\theta = 0 \), so that
\[ \tan \theta_{\text{max}} = -\frac{V_y}{V_z} \] (B.36a)
\[ \sin \theta_{\text{max}} = \frac{V_z}{\sqrt{V_y^2 + V_z^2}} \] (B.36b)
\[ \cos \theta_{\text{max}} = -\frac{V_y}{\sqrt{V_y^2 + V_z^2}} \] (B.36c)

Given the relationships of Eqs (B.36b and c), the minimum and maximum values of the shear flow are determined using the expressions
\[ q(\theta_{\text{max}}) = -\frac{1}{\pi r t} \sqrt{V_y^2 + V_z^2} \] (B.37a)

and
\[ q(\theta_{\text{max}} + \pi) = \frac{1}{\pi r t} \sqrt{V_y^2 + V_z^2} \] (B.37b)
The shear stress due to torque is given as

$$(\tau_{xs})_{\text{torque}} = \frac{T r}{J}$$  \hspace{1cm} (B.38)

where $J$ is the polar area moment

$$J = \int r^3 t d\theta = 2\pi r^3 t$$  \hspace{1cm} (B.39)

So, for the thin-wall approximation the maximum stresses will occur on the contour of the circular tube, consistent with bending analysis. Thus, given the relationships of Eqs (B.37a and b) and (B.39), Eq. (B.24) becomes

$$\tau_{xs} = \frac{1}{\pi r t} \left( \frac{T}{2 r} \pm \sqrt{V_y^2 + V_z^2} \right)$$  \hspace{1cm} (B.40)
Appendix C  Bibliography

This appendix provides a summary of the material used in doing the work. Not all of the citations included here were explicitly cited in the report, but contain material of interest to people studying the landing gear problem.

C.1. Textbooks


C.2. AGARD Reports


AGARD CP 484: *Landing Gear Design Loads*, October 1990. (N91-28150), see in particular:

C.3. Government/Industry Standards


The following aerospace recommended practices have been developed by the Society of Automotive Engineers (SAE) A-5 Aerospace Landing Gear System Committee:

- ARP 597 Wheels and Brakes, Supplementary Criteria for Design Endurance, Civil Transport Aircraft, April 1991.
- ARP 1494 Verification of Landing Gear Design Strength, February 1978.
- ARP 1821 Aircraft Flootation Analysis Methods, May 1988.
- ARP 4243 Landing Area/Landing Gear Compatibility, April 1993.

C.4. Technical Papers/Reports

C.4.1 Landing Gear Design

O'Massey, R. C., "Introduction to Landing Gear Design," ASM Paper W70-18.1, March 1970. (extensive data on shock absorber struts, somewhat lesser data on tire scrubbing and materials)


Veaux, J., "New Design Procedures Applied to Landing Gear Development," *Journal of Aircraft*, Vol. 25, No. 10, October 1988, pp. 904-910 (This paper describes the use of CAD tools in the landing gear design process, and is not directly germane to the current project.)


**C.4.2 Weight Estimation**


**C.4.3 Center of Gravity and Load Balancing**


Packing Problem References


This paper has several algorithms for aircraft cargo loading and attainment of a certain balance range.


C.4.4 Pavement Flotation


C.4.5 Cost


C.4.6 Vehicle Integration


C.5. Related Articles (arranged in chronological order)


C.6. Aircraft Data


**C.7 Simple Landing Gear Dynamics Models for Insight.**

Flügge, W., “Landing-Gear Impact,” NACA TN 2743, October 1952. This is a classic for insight. It provides a simple dynamic model of the landing gear as a spring-mass-damper. I actually think the equations are a tad oversimplified since today they would be solved using computational methods. An example analysis of the metering pin is included. The report also includes the analysis of the landing gear hitting a step in the pavement. Curiously, the design community does not use this first principals analysis to do the design. Probably the analysis is not complete enough for actual design.


**C.8 Historical (pre 1970) and Miscellaneous**


Appendix D    Aircraft Tire Database

This appendix contains the information on the tires and wheels required in the landing gear analysis. Table D.1 contains the tire information, and Table D.2 contains the wheel information.

Table D.1, Aircraft tire data, contains the following for 100 tires: (from Ref. 23.)

- Item number
- Size (Outside diameter x section width — rim diameter)
- Ply. This is an index of the tire strength, and does not necessarily represent the number of cord plies in the tire.
- Speed in mph. The maximum speed to which the tire is qualified
- Load in pounds. The maximum load for the ply rating of the tire.
- Max braking in pounds. The maximum steady braking which may be applied to a tire during landing.
- Inflation pressure in psf. This is the inflation pressure required to support the rated load.
- Tire weight in pounds. This is the calculated weight of approved construction, not the maximum weight.
- Maximum inflated outside diameter in inches
- Maximum inflated width in inches
- Aspect ratio, the ratio of the tire section height to the tire section width

Table D.2, Aircraft Wheel data, contains the following for 100 wheels:

- Item number
- Size
- Width in inches
- Diameter in inches
Table D.1 Aircraft tire data (1 of 4) [24]

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</table>
Appendix E  Analysis Package User’s Manual

E.1. Introduction

The package is intended to provide aircraft conceptual designers with tools to help automate the landing gear design process.

E.2. Package Organization

The package consists of four executable files, config.for, limit.for, pave.for, and gearwei.for. An input file with extension .inp is required for each program. Program config.for currently acts as the front-end of the package and accepts all the data that is input, even though some of the data may not be used by the program itself. Program config.for then creates input files for the other three programs. Data files tire.dat and pavecoef.dat are required to provide database for programs config.for and pave.for, respectively.

The first line in each input file is a blank card, to be used as a case title card. All the rest of the input is formatted. We suggest that the sample input files be used as templates. Typically, the character data is read in as alphanumeric format, the integer data is read in as 3(10x, i10), and real data is read in as 3(10x, f10.2). The fields that are skipped are intended for variable labels. Note that if the given aircraft does not exhibit a fuselage-mounted landing gear, zeros should be entered in place of those input variables that are related to the fuselage-mounted gear.

The codes produce minimal screen output, and do not write out anything until they enter subroutine output, at the end of the computation. Config does provide some write statements, to provide an indication of the progress during the calculation. If there are problems with input data sets it will be at least slightly difficult to troubleshoot. The input and output files names are hardwired, but could easily be changed to prompt the user for file names. The most painful input appears to be the stowage volume definition. This input
in `config` can be fictitious and the program will still execute. Note that three sets of stowage volumes are read in, whether a fuselage mounted main assembly is used or not. As mentioned above, zeros should be entered when a fuselage mounted gear is not present. If you don’t input all three, the code will fail, giving an out of data error.

**Process**

Required aircraft/landing gear characteristics are arranged into card-style input file “config.inp” to be read in by program “config.for”. Selected tire/wheel characteristics and landing gear model, as well as initial data are arranged to form “limit.inp”, “pave.inp”, and “gearwei.inp” to be read in by “limit.for”, “pave.for”, and “gearwei.for”, respectively. Examination of the list of constraint violations as generated by “limit.for”, e.g., sideways turnover angle, takeoff rotation angle, turning radius, and stowage characteristics, and pavement thickness requirement and ACN as generated by “pave.for” will provide insight to what should be done to resolve these constraint violations. Some possible options include relocating the landing gear, extending the strut length, modifying the aircraft cg height off the ground, and an increase/decrease of clearance requirements. After all of the design constraints are satisfied through an iterative modification process, the finalized landing gear model is passed to “gearwei.for” for component/group weight estimation.

The following sections define the subroutines and calling tree for each program. The program input and output is also summarized. Details are contained in Chapter 9.
config.for

Subroutines

datain
brsize - brake sizing
select - tire/wheel selection
attach - landing gear attachment scheme
pivaxi - pivot axis alignment
cropro - cross-product
output

Subroutine calling sequence:

datain
brsize
select
attach
  pivaxi
  cropro
output

Outputs:

Brake dimension and weight
Tire/wheel design characteristics: dimensions and weight
Stroke length
Load-stroke curve
Mathematical landing gear model: axle, truck beam, piston, cylinder, trunnion, drag
and side struts
limit.for

Subroutines

datain
layout - landing gear positioning constraints
runway - ground operation characteristics
stowag - stowage constraints
skewed - skewed pivot axis alignment
output

Subroutine calling sequence

datain
layout
stowag
  pivaxi
    skewed
  cropro
retrac
violat
output

Outputs

Takeoff/landing stability characteristics: pitch and roll angles
Ground stability characteristics: sideways turnover and tail-tipping
Ground clearance: nacelle-to-ground and wingtip-to-ground
Maneuverability characteristics: centerline-guidance tracking
  and minimum turning radius
Kinematics: pivot axis alignment, retraction angle, landing gear retracted position
pave.for

Subroutines
datain
offset - offset distance, analysis node to tire contact area
aceswl - equivalent single wheel load
rigith - rigid pavement thickness and ACN
flexth - flexible pavement thickness and ACN
output

Subroutine calling sequence

datain
offset
aceswl
rigith
flexth
output

Outputs

Flexible and rigid pavement thickness and corresponding ACN
gearwei.for

This program computes an estimate of the landing gear weight.

**Subroutines**
- datain
- exload - applied load
- noreac - structural nodal actions
- crosec - cross-sectional area sizing
- weiest - weight estimation
- cropro - cross-product
- cotran - coordinate transformation
- matinv - matrix inverse
- rowpiv - row pivoting
- ccross - cylindrical cross section sizing
- cirstr - circular tube stresses
- icross - i-bar cross section sizing
- select - design cross section selection
- output

**Subroutine calling sequence**
- datain
- exload
  - cropro
- noreac
  - cropro
- crosec
  - cotran
    - matinv
      - rowpiv
    - ccross
    - cirstr
  - icross
  - select
  - weiest
  - output

**Outputs**
- Component dimensions
- Component/group weight estimation
### E.3. Program Input Variables

<table>
<thead>
<tr>
<th>Variable</th>
<th>Description</th>
<th>Options</th>
</tr>
</thead>
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<td>aircraft</td>
<td>Aircraft identification</td>
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<tr>
<td>brake</td>
<td>Brake material</td>
<td>1 steel, 2 carbon</td>
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<tr>
<td>wheel</td>
<td>Wheel material</td>
<td>1 forged aluminum, 2 cast aluminum, 3 titanium, 4 steel</td>
</tr>
<tr>
<td>objec</td>
<td>Wheel selection criterion</td>
<td>1 minimum pressure, 2 minimum weight, 3 minimum size</td>
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<td>metal</td>
<td>Landing gear structure material</td>
<td>1 4340 steel, 2 300M steel</td>
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<tr>
<td>mtow</td>
<td>MTOW, lb</td>
<td></td>
</tr>
<tr>
<td>mldw</td>
<td>Maximum landing weight, lb</td>
<td></td>
</tr>
<tr>
<td>fuel</td>
<td>Fuel weight, lb</td>
<td></td>
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<tr>
<td>cmax</td>
<td>Maximum main assembly load, percent MTOW</td>
<td></td>
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<tr>
<td>cmin</td>
<td>Minimum main assembly load, percent MTOW</td>
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</tr>
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<td>warena</td>
<td>Wing area, ft$^2$</td>
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<td>wspan</td>
<td>Wing span, in</td>
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<tr>
<td>qswep</td>
<td>Quarter chord sweep, deg</td>
<td></td>
</tr>
<tr>
<td>dihed</td>
<td>Dihedral, deg</td>
<td></td>
</tr>
</tbody>
</table>
croot Root chord, in

taper Taper ratio

clmax Clmax, landing

nms Number of main struts

nmw Number of main wheels

nnw Number of nose wheels

wpsm Number of wheels per strut, main assembly

wpsn Number of wheels per strut, nose assembly

dyna Landing gear load factor

alpha Angle of attack, touchdown, deg

wbeta Truck beam rotation angle, wing-mounted assembly, deg

fbeta Truck beam rotation angle, fuselage-mounted assembly, deg

incl Axle incline from the vertical, deg

scrap Tail scrape angle, deg

dnace Nacelle diameter, in

clear Nacelle-to-ground clearance, in

cg(i) Aircraft cg location, aircraft reference frame, in

wing(i) Wing root leading edge location, aircraft reference frame, in

i = 1, .., 3 (x, y, and z coordinate, airframe)

engi(i) Inboard engine location, aircraft reference frame, in

i = 1, .., 3 (x, y, and z coordinate, airframe)

tcon(i) Tail bumper location, aircraft reference frame, in

i = 1, .., 3 (x, y, and z coordinate, airframe)

gear(i,j) Landing gear assembly location, aircraft reference frame, in

in the order: main, nose, body

well(i,j,k) Landing gear stowage volume, aircraft reference frame, in
in the order main, nose, body

The number at the end of the variable denotes the corners of the rectangular-shaped stowage volume:

1  upper starboard corner, forward
2  upper port corner, forward
3  lower starboard corner, forward
4  upper port corner, forward
5  upper starboard corner, aft
6  upper port corner, aft
7  lower starboard corner, aft
8  upper port corner, aft
E.4. Sample Input Files

747conf.inp

c landing gear layout/configuration input file

aircraft: b747

brake = 1, wheel = 1, objc = 2
metal = 1
mtow = 738000.00, mldw = 564000.00, fuel = 316307.00

cmax = 0.96, cmin = 0.88, warea = 5500.00
wspan = 2348.00, qswee = 37.70, dihed = 7.00

croot = 642.00, taper = 0.25, clmax = 2.55
nms = 4.00, nmw = 16.00, nmw = 2.00
wpsm = 4.00, wdsn = 2.00, dyna = 1.20
alpha = 4.00, wbeta = 60.00, fbeta = 0.00
incl = 10.00, scrap = 12.00, dnace = 110.00
clear = 12.00

c component location

xcg = 1260.00, ycg = 0.00, zcg = -24.00
xwing = 870.00, ywing = 0.00, zwing = -88.00
xengi = 1050.00, yengi = -115.00, zengi = -95.00
xtcon = 2375.00, ytcon = 0.00, ztcon = 0.00
xmain = 1254.00, ymain = -216.00, zmain = -62.00
xnose = 290.00, ynose = 0.00, znose = -106.00
xbody = 1375.00, ybody = -75.00, zbody = -118.00

c wing-mounted main assembly stowage

xm1 = 1164.00, ym1 = -17.00, zm1 = -38.00
xm2 = 1164.00, ym2 = -115.00, zm2 = -38.00
xm3 = 1164.00, ym3 = -115.00, zm3 = -136.00
xm4 = 1164.00, ym4 = -17.00, zm4 = -136.00
xm5 = 1260.00, ym5 = -17.00, zm5 = -38.00
xm6 = 1260.00, ym6 = -115.00, zm6 = -38.00
xm7 = 1260.00, ym7 = -115.00, zm7 = -136.00
xm8 = 1260.00, ym8 = -17.00, zm8 = -136.00

c nose assembly stowage

xn1 = 150.00, yn1 = 32.00, zn1 = -38.00
xn2 = 150.00, yn2 = -32.00, zn2 = -38.00
xn3 = 150.00, yn3 = -32.00, zn3 = -88.00
xn4 = 150.00, yn4 = 32.00, zn4 = -88.00
xn5 = 290.00, yn5 = 32.00, zn5 = -38.00
xn6 = 290.00, yn6 = -32.00, zn6 = -38.00
xn7 = 290.00, yn7 = -32.00, zn7 = -112.00
xn8 = 290.00, yn8 = 32.00, zn8 = -112.00

c fuselage-mounted main assembly stowage

xb1 = 1260.00, yb1 = -8.00, zb1 = -38.00
xb2 = 1260.00, yb2 = -115.00, zb2 = -38.00
xb3 = 1200.00, yb3 = -115.00, zb3 = -136.00
xb4 = 1200.00, yb4 = -8.00, zb4 = -136.00
xb5 = 1390.00, yb5 = -8.00, zb5 = -38.00
xb6 = 1390.00, yb6 = -115.00, zb6 = -38.00
xb7 = 1390.00, yb7 = -115.00, zb7 = -136.00
xb8 = 1390.00, yb8 = -8.00, zb8 = -136.00

168
747limi.inp

landing gear layout/stowage constraints input file

aircraft: b747

cmax = 0.96, cmin = 0.88, hcg = 181.00
wspan = 2348.00, qswep = 37.70, dihed = 7.00
croot = 642.00, taper = 0.25
rms = 4.00, wpsm = 4.00, wpsn = 2.00
scrap = 12.00, dnace = 110.00, clear = 12.00
wbeta = 60.00, fbeta = 0.00, incl = 6.00
smain = 27.65, snose = 27.97, sfuse = 27.65

component location

xcg = 1260.00, ycg = 0.00, zcg = -24.00
xwing = 870.00, ywing = 0.00, zwing = -88.00
xengi = 1050.00, yengi = -465.00, zengi = -95.00
xtcon = 2375.00, ytcon = 0.00, ztcon = 0.00
xmain = 1253.50, ymain = -215.00, zmain = -64.00
xnose = 290.00, ynose = 0.00, znose = -106.00
xfuse = 1375.00, yfuse = -75.00, zfuse = -118.00

wing-mounted main assembly stowage

xml = 1164.00, yml = -17.00, zm1 = -38.00
xm2 = 1164.00, ym2 = -132.00, zm2 = -38.00
xm3 = 1164.00, ym3 = -132.00, zm3 = -136.00
xm4 = 1164.00, ym4 = -17.00, zm4 = -136.00
xm5 = 1260.00, ym5 = -17.00, zm5 = -38.00
xm6 = 1260.00, ym6 = -132.00, zm6 = -38.00
xm7 = 1260.00, ym7 = -132.00, zm7 = -136.00
xm8 = 1260.00, ym8 = -17.00, zm8 = -136.00

nose assembly stowage

xn1 = 150.00, yn1 = 32.00, zn1 = -38.00
xn2 = 150.00, yn2 = -32.00, zn2 = -38.00
xn3 = 150.00, yn3 = -32.00, zn3 = -88.00
xn4 = 150.00, yn4 = 32.00, zn4 = -88.00
xn5 = 290.00, yn5 = 32.00, zn5 = -38.00
xn6 = 290.00, yn6 = -32.00, zn6 = -38.00
xn7 = 290.00, yn7 = -32.00, zn7 = -112.00
xn8 = 290.00, yn8 = 32.00, zn8 = -112.00

fuselage-mounted main assembly stowage

xf1 = 1260.00, yf1 = -17.00, zf1 = -30.00
xf2 = 1260.00, yf2 = -115.00, zf2 = -30.00
xf3 = 1200.00, yf3 = -115.00, zf3 = -136.00
xf4 = 1200.00, yf4 = -17.00, zf4 = -136.00
xf5 = 1390.00, yf5 = -17.00, zf5 = -30.00
xf6 = 1390.00, yf6 = -115.00, zf6 = -136.00
xf7 = 1390.00, yf7 = -115.00, zf7 = -136.00
xf8 = 1390.00, yf8 = -17.00, zf8 = -136.00

selected tire data

criterion: minimum weight

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<th>ply</th>
<th>speed (mph)</th>
<th>load (lb)</th>
<th>infl (psi)</th>
<th>brake (lb)</th>
<th>wei (in)</th>
<th>dia (in)</th>
<th>wid (in)</th>
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selected wheel data

material: aluminum, forging
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c mathematical model

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<table>
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c aircraft flotation input file
aircraft: b747
mtow = 738000.00, mlw = 564000.00
cremax = 0.96
nrmw = 16.00, wpsm = 4.00

c selected tire data

criterion: minimum weight

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<th>load (lb)</th>
<th>infl (psi)</th>
<th>brake (lb)</th>
<th>wei (lb)</th>
<th>dia (in)</th>
<th>wid (in)</th>
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<td>50400.0</td>
<td>210.0</td>
<td>75600.0</td>
<td>243.3</td>
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c selected wheel data

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<th>wid (in)</th>
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c mathematical model

wing

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<th>yl (in)</th>
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nose

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fuselage

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c landing gear weight estimation input file
aircraft: b747
metal = 1
mtow = 738000.00, mldw = 564000.00
cmax = 0.96, cmin = 0.88, hcg = 181.00
rms = 4.00, nw = 16.00, nw = 2.00
wpsm = 4.00, wpsn = 2.00, alpha = 4.00
dyna = 1.20, inpr = 1500.00, bwei = 262.11
smain = 26.65, snose = 26.97, sfuse = 26.65
c component location
xmain = 1254.00, ymain = -216.00, zmain = -62.00
xnose = 290.00, ynose = 0.00, znose = -106.00
xfuse = 1375.00, yfuse = -75.00, zfuse = -118.00
c selected tire data
criterion: minimum weight
type size ply speed load infl brake wei dia wid
(mph) (lb) (psi) (lb) (lb) (in) (in)
wing 49x17 32.0 235.0 50400.0 210.0 75600.0 243.3 48.8 17.3
nose 46x16 28.0 210.0 41800.0 210.0 62700.0 185.8 45.3 16.0
fuselage 49x17 32.0 235.0 50400.0 210.0 75600.0 243.3 48.8 17.3
c selected wheel data
material: aluminum, forging
type size dia wid hub wei
(in) (in) (in) (lb)
wing 49x17 13.3 20.0 10.0 86.2
nose 46x16 13.3 20.0 10.0 105.3
fuselage 49x17 13.3 20.0 10.0 86.2
c mathematical model
wing
component x0 y0 z0 xl yl zl
(in) (in) (in) (in) (in) (in)
tire 0.00 0.00 0.00 0.00 13.25 -20.20
axle 0.00 22.00 0.00 0.00 -22.00 0.00
truck beam -29.00 0.00 0.00 29.00 0.00 0.00
piston 0.00 0.00 0.00 0.00 0.00 -112.00
cylinder 0.00 0.00 0.00 0.00 42.00 4.00 -101.00
drag strut 0.00 0.00 0.00 0.00 0.00 -84.00 -86.00
side strut 0.00 0.00 0.00 0.00 0.00 -84.00 -86.00
forward trunnion 0.00 0.00 0.00 16.00 4.00 0.00
aft trunnion 16.00 4.00 0.00 56.00 18.00 0.00
nose
component x0 y0 z0 xl yl zl
(in) (in) (in) (in) (in) (in)
tire 0.00 0.00 0.00 0.00 13.25 -20.20
axle 0.00 17.00 0.00 0.00 -17.00 0.00
truck beam 0.00 0.00 0.00 0.00 0.00 0.00
piston 0.00 0.00 0.00 0.00 0.00 -34.00
cylinder 0.00 0.00 0.00 0.00 0.00 -78.00
drag strut 0.00 0.00 0.00 41.00 0.00 -82.00
side strut 0.00 0.00 0.00 0.00 19.00 38.00
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