SR-71 Blackbird Reverse Engineering Project

Signatures:

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Summary:

The SR-71 Blackbird, the fastest aircraft still stands as a testament to technological brilliance and the peak of aerospace engineering. In this project, one of the most recognizable airplanes from the Cold War era, The SR-71 Blackbird is reverse engineered in this effort by a group of aerospace engineering students. This study explores many aspects of the aircraft (such as aerodynamic, aerothermal, performance, propulsion, etc) and how modern materials, aerodynamics, and propulsion systems may be incorporated into a new SR-71 design in an effort to reimagine the aircraft. The report's conclusions deep understanding the iconic American aircraft and highlight the possibility for developing a new breed of supersonic aircraft that are more effective, ecologically friendly, and commercially feasible by utilizing lightweight composites, enhanced aerodynamics, and hybrid propulsion systems. In order to realize this vision of the SR-71 of the future, the paper ends with suggestions for more study and development.

Mission Details:

- ***Flight ceiling of 85,000 ft***
- ***Payload of 3,000 lbs***
- ***Single pilot with minimal equipment weight***
- ***Takeoff at Edwards Air Force base***

Distribution:

Institution:	***Dept.:***	***Name:***
The University of Texas at Arlington	*MAE*	*Dr. Bernd Chudoba*

Work Disclosure Statement

The work I performed to document the results presented in this report was performed by me, or it is otherwise acknowledged.

Date: 6/27/2024

Signature: Phat Nguyen

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Nomenclature

C = Coefficient
S = Reference Area / Distance
T = Thrust
W = Weight
V = Velocity
L = Lift
D = Drag
α = Angle of attack
ρ = Density
M = Mach number
μ = Viscosity

Subscripts
TOFL = Takeoff field length
BR = Braking
F = Flare
A = Approach
G = Ground roll
Do = Zero-lift drag
pln = Planform
d = 2D drag
l = 2D lift

I. Introduction

Fig. 1 Lockheed SR-71 Blackbird. [1]

A. Overview

Lockheed SR-71 Blackbird is most advanced member of the Blackbird family that also included the A-12, M-21, YF-12, D-21, was developed by a team at Lockheed under the leadership of Clarence "Kelly" Johnson for reconnaissance missions. Blackbird family's member are still the only production aircraft that can cruise at Mach 3 for an extended period and fly at heights more than 80,000 feet. The Blackbirds are designed to fly up to 85,000 feet in the air and at speeds of Mach 3.2, or little more than 2,200 mph.

The Blackbird was designed for the retirement of U-2 reconnaissance plane. Although the U-2 was a high-altitude reconnaissance plane with a ceiling of more than 70,000 feet, it was susceptible to enemy defenses. In 1960, the U-2 was shot down over Soviet territory, which resulted in the pilot's detention and a diplomatic incident. In response to these worries, the Blackbird family was developed with the intention of developing an aircraft that was faster, higher ceiling which was less vulnerable. Project OXCART was funded in August 1959 for development of the A-12. In early 1960s, prototype of A-12 was tested in NASA Ames Research Center High-speed Wind Tunnel. Project Kedlock also launched in early 1960 for the development of A-12. On 29 February 1964, President Lyndon Johnson announced the existence of the aircraft as "YF-12A". The aircraft was move to Edwards Air Force for flight test. YF-12A was designed for interceptor with advanced weapon and radar. However, YF-12A was never put into production due to the birth of nuclear weapon, high altitude interceptor was no longer necessary. Right after the YF-12A, two other members of the Blackbird's family, M-21 and SR-71 were debuted with two-seat and inherited technology from YF-12A. M-21 was designed to carry a supersonic reconnaissance drone D-21 as project Tagboard. However, during the flight test, a fatal accident occurred, the drone pitched down and hit the M-21 which made a crack in middle of the plane. Two pilot successfully ejected but one did not make it alive. M-21's project was cancelled by Kelly after the fatal accident, SR-71 was only member left in service. From 1966 through 1998, the SR-71 Blackbird was a functional aircraft in the US Air Force. The SR-71 was primarily employed for reconnaissance missions. The SR-71 was retired despite having excellent capabilities because of its expensive running expenses and the growing of satellite reconnaissance technology.

Aerodynamic, aerothermal and performance of the Blackbird are mainly focused on this report. The aircraft's swept-back wings and blended body which gives it stability and allows it to fly at high-speed exceeding Mach 3.

B. Mission Details

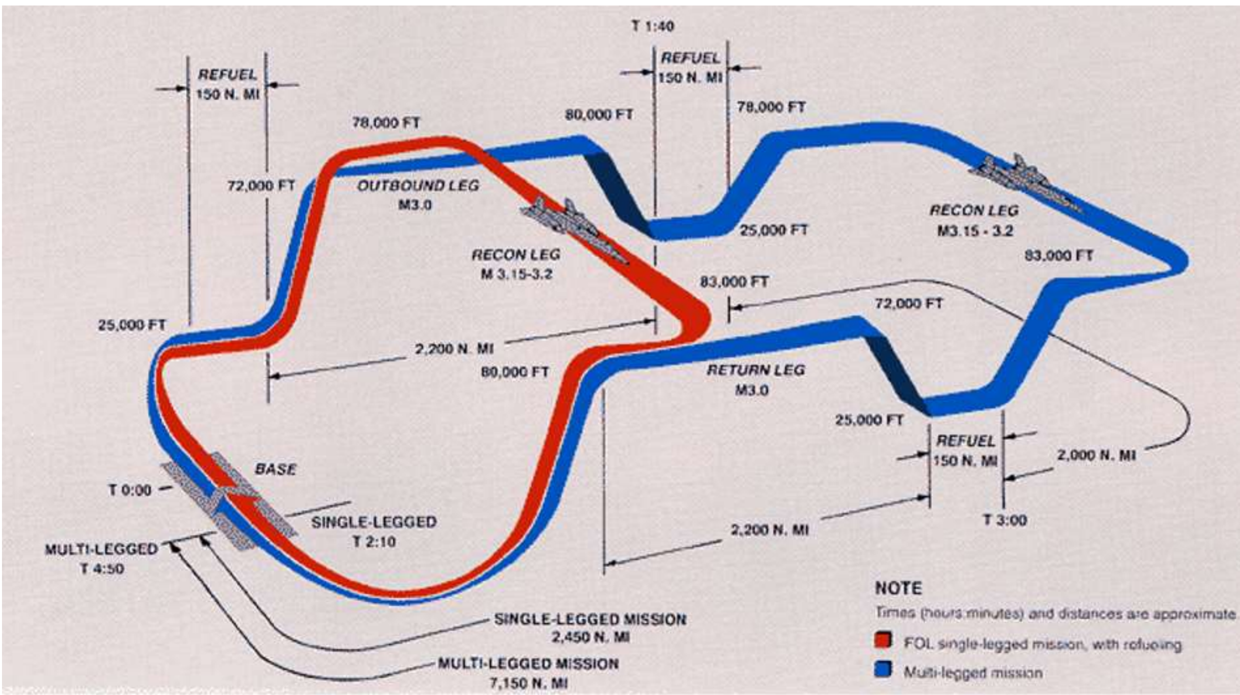

Fig. 2 SR- 71 Mission Profile. [7]

- •Flight ceiling of 85,000 ft
- Payload of 3,000 lbs
- Single pilot with minimal equipment weight
- Takeoff at Edwards Air Force base
- SR-71 Geometry remains unchanged, but the material selection and propulsion systems are updated to the current state of the art

Edwards Air Force Base is in California's Mojave Desert, roughly 100 miles northeast of Los Angeles. The two distinct natural resources contribute to the base's status as a leading flight test facility.
The main Edwards concrete runway is located next to Rogers Dry Lake and combining its 15,000-foot length with a 9,000-foot lakebed overrun affords pilots in an in-flight emergency one of the world's longest and safest runways. Rosamond Dry Lake is also utilized for routine flight testing and research, as well as emergency landings. Normal weather is around 14 Degree Celsius, 32% Humidity. (Source: NASA The Lake Beds)

C. General characteristics of SR-71 from its Manual (sr-71.org)

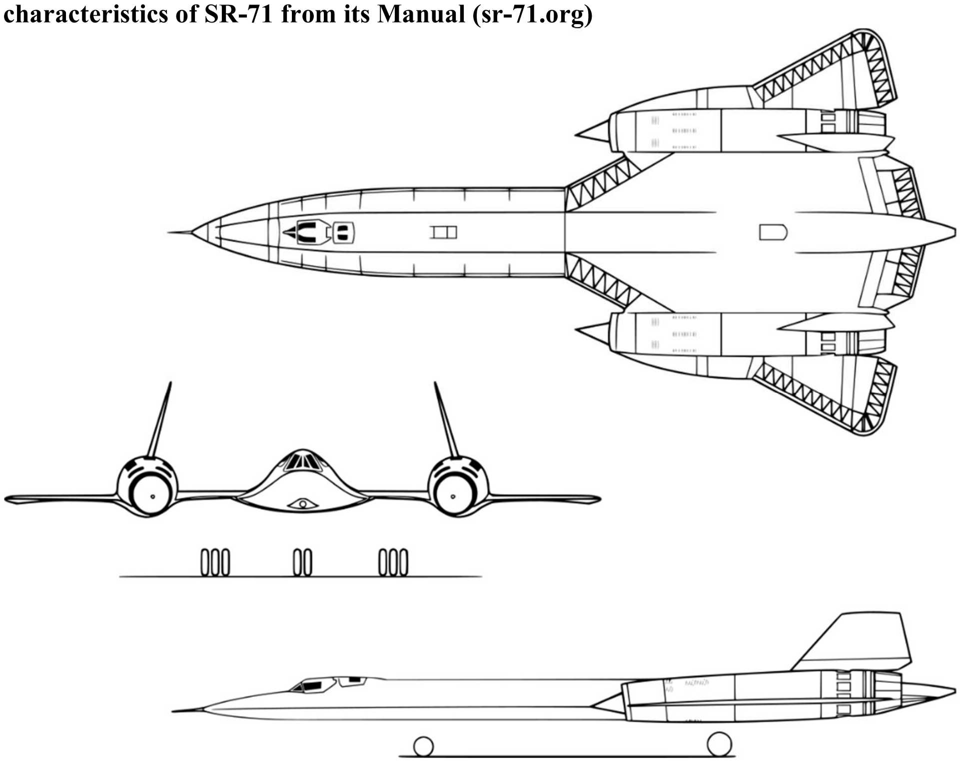

Fig. 3 SR-71 Drawing (Wikipedia)

SR-71 Specifications (sr-71.org Manual)	
Manufacturer:	Lockheed Aircraft Corporation
Length:	107' 5"
Length of Nose Probe:	4' 11"
Wing Span:	55' 7"
Wing Area:	1,795 ft. sq.
Wing Aspect Ratio:	1.939
Wing Root Chord:	60.533
Wing Dihedral Angle:	0 degrees
Wing Chord:	0.00
Wing Sweep:	52.629 degrees
Inboard Elevon Area:	39.00 ft. sq.
Outboard Elevon Area:	52.50 ft. sq.
Total Vertical Rudder Area:	150.76 ft. sq.

Moveable Rudder Area:	70.24 ft. sq.
Rudder Root Chord:	14.803 ft.
Rudder Tip Chord:	7.833 ft.
Height:	18' 6"
Empty Weight:	59,000 lbs.
Maximum Weight:	170,000 lbs.
Fuselage Diameter:	5.33 ft.
Service Ceiling:	85,000'+
Maximum Speed:	Mach 3.3+ (Limit CIT of 427 degrees C)
Cruising Speed:	Mach 3.2
Engines:	2 Pratt & Whittney J-58 (JT11D-20A) with 34,000 lbs. of thrust.
Range:	3,200 nautical miles (without refueling)

$W_{fuel} = 80000\ lb$ (Based on https://www.nasa.gov/centers/armstrong/news/FactSheets/FS-030-DFRC.html)

The SR-71's chinned fuselage is long and narrow, tapering to a point at the back to reduce drag. The cockpit is at the front of the fuselage, concave fairing that aids in the smooth passage of air over the fuselage.

The SR-71's wings were a highly swept, narrow delta wing with a leading-edge sweep angle of 60 degrees to minimize supersonic wave drag. The wing planform was substantially elliptical, resulting in a high aspect ratio and little induced drag. Winglets were added to the tips of the wing to minimize induced drag and increase the aircraft's overall aerodynamic efficiency. The SR-71's wing design was optimized for supersonic flight and was vital to the plane's ability to operate at high speeds and altitudes. The SR-71's high sweep angle, small wing, and winglets all contributed to lowering the wave drag associated with supersonic flight, allowing it to reach its supersonic speed.

The SR-71 is powered by two J58 engines located in pods behind the cockpit on the fuselage. These engines are capable of both propulsion and high-speed cruising. The engine inlets are positioned on the sidewalls of the fuselage, directly behind the cockpit. They are precisely engineered to supply the engines with the required air while retaining great efficiency at high speeds. The engine inlets are positioned on the sides of the fuselage, directly behind the cockpit. They are precisely engineered to supply the engines with the required air while retaining great efficiency at high speeds. The exhaust from the engines is released through two nozzles at the fuselage's rear. The nozzles are slightly tilted outward to enhance directional stability and to reduce the aircraft's heat.

SR-71 vertical tails are canted inward to reduce rolling moment by the side force. Tail tilting inward which reduces the moment arm of the side force.

II. Literature Review

1. Literature sources

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2. Design process.

Aircraft design is a long and iterative process. The process must begin with a good start, all detailed mission constraints, criteria and must be defined before starting. The constraints of the aircraft design are combined of many factors such as aircraft performance (such as clime rate, turn rate, endurance, range, takeoff and landing, speed, maneuverability, specific fuel consumption), airworthiness, cost. The convergence of the defined constraints is the point to start with the design. The more constraints, the closer to the high end aircraft design. However, over-constrained leads to dead end, no viable solution exists. A blindfolded start always ends up with a wasted effort and time. In the SR 71 design project, since it is military operation aircraft, cost is not the driving factor, the mission details are the paramount constraints.

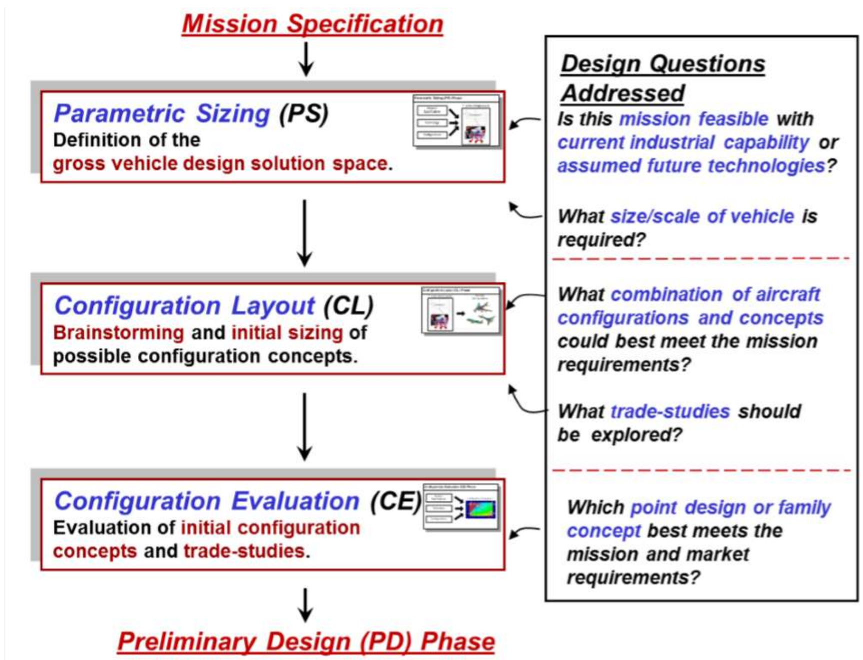


Fig. 4 Basic Phases in Design of Aerospace vehicle (Dr. Chudoba, Aerospace Vehicle Design I, 2023)[7]

According to certain mission requirements and constraints, Parametric sizing is an important first step in the creation of any new aircraft project since it gives a preliminary understanding of the design and potential trade-offs across many disciplines such as aerodynamics, structures, propulsion, and systems.

Parametric sizing helps engineers to examine and optimize designs based on numerous design criteria and limitations by using mathematical models and algorithms. This procedure aids in the identification of the optimal design options as well as the quantification of the potential advantages and hazards associated with each alternative. Furthermore, parametric sizing might help identify areas that require further research or technical advancements.

This information is crucial for making informed decisions on the feasibility and viability of the project, as well as developing a well-defined plan for the project's succeeding phases.

The sizing process is comprised of six key aspects, according to Gary Coleman's study:

1. Estimation of Operating Empty Weight (OEW) based on supplied or currently iterated geometry, payload weight, and TOGW. This shows the weight of the vehicle's structural components, systems, operational elements, and propulsion system.
2. Trajectory analysis (estimation of fuel weight) - The fuel weight is estimated based on the needed range and endurance. This level largely relies on aerodynamic and propulsion disciplinary.
3. Convergence logic - which is a technique used in aircraft design to identify a solution that meets the criteria and limitations of the design. Iteratively modifying design parameters such as takeoff gross weight (TOGW) until the implicit function established by the empty weight (OEW) estimation and trajectory analysis is fulfilled. The use of convergence logic is a method of balancing the trade-offs between aircraft geometry which influences aerodynamics and construction and takeoff gross weight which affects fuel weight and structural loading. The convergence logic process is often iterative, with the geometry held constant while the TOGW is modified until a required solution is met.
4. Constraint analysis - the method begins with determining the needed wing loading (W/S) and thrust loading (T/W) based on mission and operational requirements such as take-off field length, maximum cruising speed, approach speed, and one Engine Inoperative (OEI) climb performance. This data is utilized to define wing area and maximum sea-level thrust limits, which are two of the most essential design limitations. Aerodynamic and propulsion disciplinary and performance estimate methods are utilized to establish the needed wing loading and thrust loading. This necessitates knowledge of aerodynamics and propulsion systems, as well as a comprehension of the aircraft's performance objectives. The results of the constraint analysis serve as the foundation for further trade-off studies in the design process.
5. Sizing logic - a method for solving the implicit function formed by the OEW estimation, trajectory analysis, and constraint analysis. The implicit function generated by the OEW estimate and trajectory analysis illustrates the link between the aircraft geometry and its weight, performance, and flight characteristics. The function can be solved by iterative numerical, graphical methods.
6. Trade studies –it is the last element of aircraft design understanding which show the trade-offs between various design factors and identifying the best solutions to satisfy mission requirements and criteria. A trade study, for instance, to generate more lift the aircraft must have higher aspect ratio in trade off lower maneuverability.

Finding the best configuration that satisfies the design limitations and criteria, such as maximum weight, range, cargo capacity, stability, and maneuverability, is the goal of parametric sizing. Iterative in nature, the process may include several rounds of analysis and optimization while taking trade-offs between various design variables into consideration. To start with Parametric Sizing, there are some conventional tools and methods are used, such as Lofting Sizing, Roskam I Sizing, Breguet Range, Standard to Design, etc.

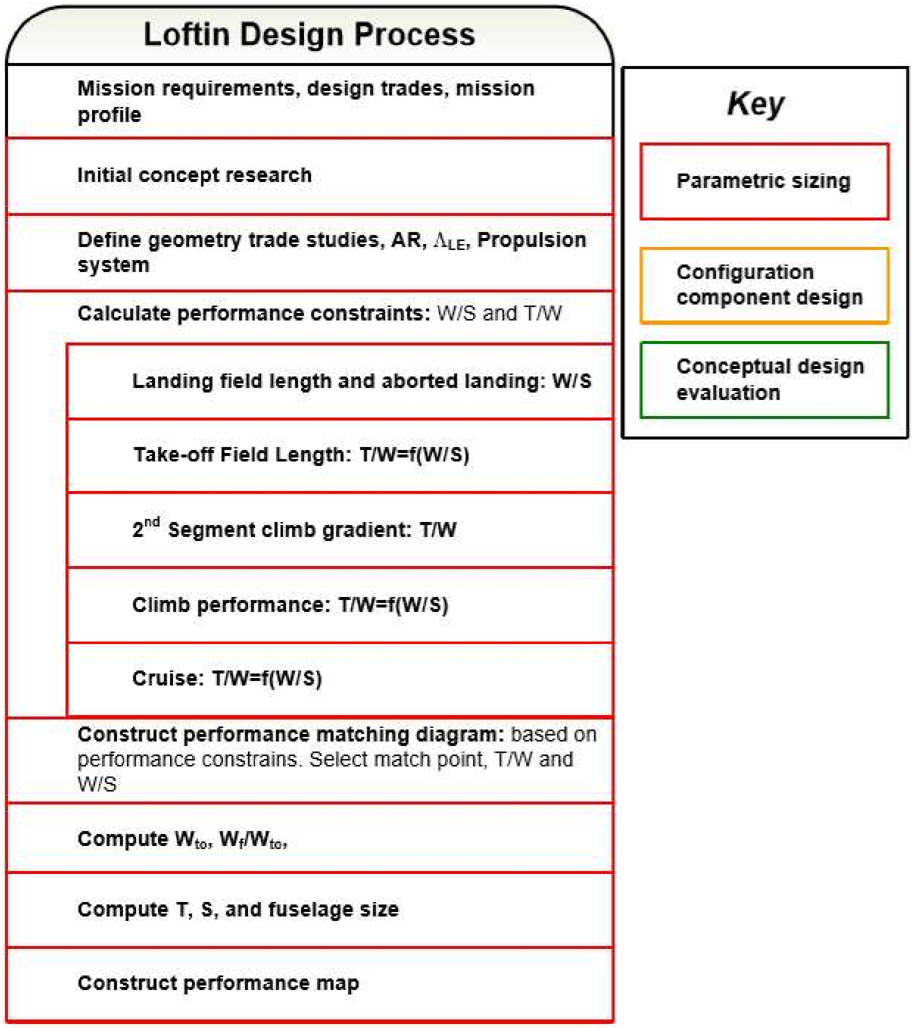


Fig. 5 Lofting Sizing Design Process (Dr. Chudoba, Aerospace Vehicle Design I, 2023)[7]

In the early stages of aircraft design, Lofting Sizing is used to visualize and aims to create a draft model of the aircraft which followed all the requirements and constraints. Following that, the model serves as the foundation for additional research and design work. Lofting starts with estimation of geometry of the aircraft based on mission requirement and initial concept research. The estimation of aircraft geometry is used as baseline to find the relationship between landing distances, takeoff filed length, climb gradient, and climb performance cruise in function of wing loading and thrust loading. Then all the functions are plotted on one graph to limit the aircraft design to a feasible area.

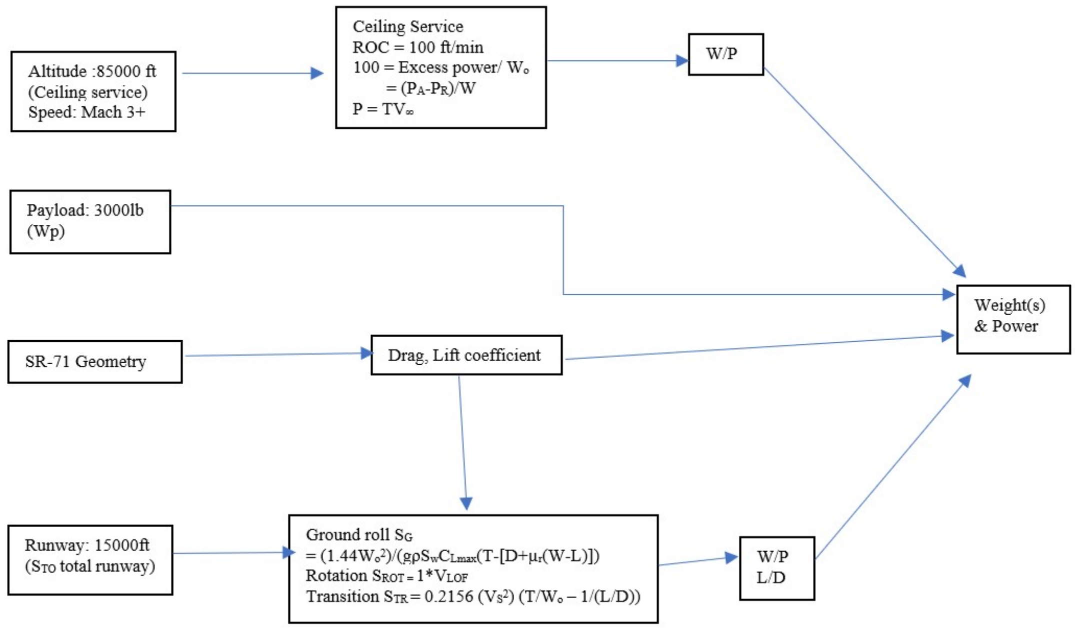


Fig 6. SR-71 Mission Requirement and Criteria Overview draft.

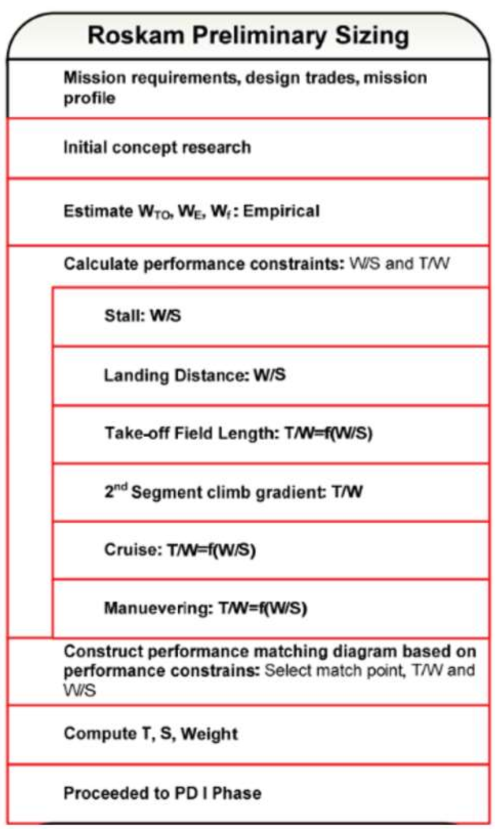


Fig. 7 Roskam Sizing Design Process (Dr. Chudoba, Aerospace Vehicle Design I, 2023)[7]

The Roskam Preliminary Sizing approach is designed to give a quick and easy way to assess the feasibility of a novel aircraft idea and identify its strengths and flaws. It is commonly employed in the early phases of the design process, prior to more extensive analysis and optimization. The Roskam's methodology is very similar to Loftin. Instead of starting with estimation of geometry of the aircraft based on mission requirement and initial concept research, Roskam's methodology starts with estimation of weight (takeoff weight, empty weight, fuel weight). The estimation of weight is used as baseline to create the relationship between stall speed, landing distances, takeoff filed length, climb gradient, cruise, maneuvering in function of wing loading and thrust loading.

The initial aircraft can also be sized by using Hypersonic convergence which was invented by Czysz. The Hypersonic Convergence sizing method centers around a pair of equations that work together to determine both the weight and volume of the aircraft are coupled and solved. Then the wing planform area and takeoff gross weight is iterated until their convergence based on the initial assumption of slenderness ratio based on mission profile.

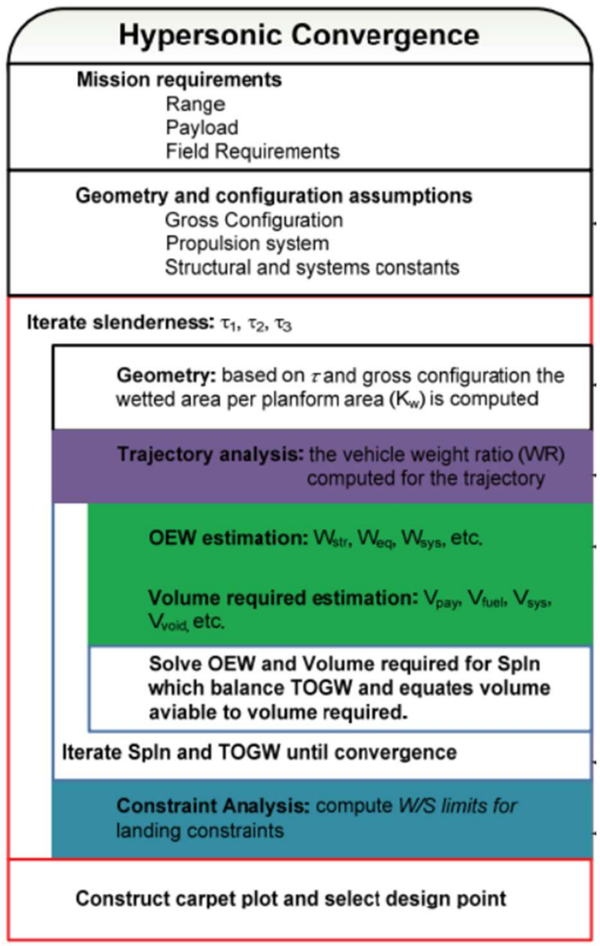


Fig 8. Hypersonic Convergence Process [7]

Weight budget equation:

$$W_{OEW} = \frac{I_{str} K_w S_{pln} + C_{sys} + W_{cprv} + \frac{T}{W} \cdot \frac{W_R}{E_{TW}} (Wpay + Wcrew) + W_{cpr}}{\frac{1}{1+\mu_a} - f_{sys} - \frac{\frac{T}{W} W_R}{E_{TW}}}$$

Volume budget equation:

$$W_{OWE} = \frac{\tau S_{pln}^{1.5} (1 - k_{vv} - k_{vs} - v_{fix} - N_{crw}(v_{crw} - k_{crw}) - Wpay/\rho pay}{\frac{W_R - 1}{\rho ppl} + \left(k_{ve} \frac{T}{W} \text{Max } W_R\right)}$$

$$W_{\mathrm{OWE}} = \mathrm{W}_{\mathrm{OEW}} + \mathrm{W}_{\mathrm{pay}} + \mathrm{W}_{\mathrm{crew}}$$

3. Supersonic aircrafts

1. Concord [14]

The Concorde aircraft had an ogival delta-wing shape and four Olympus engines that were originally designed for the Avro Vulcan strategic bomber. It had a maximum cruising altitude of 60,000 feet and an average cruising speed of Mach 2.02. It was the first civil airliner to incorporate an analog fly-by-wire flight control system, as well as retractable taxi and landing lights and a droop snoot lowering nose piece to improve visibility during approach. British Airways and Air France began commercial operations in 1976 and stopped in 2003. The Concorde used a cruise-climb approach to travel to higher altitudes as it burnt fuel and lost weight during regular flight, and dedicated oceanic airways were authorized across the Atlantic to allow it to gradually ascend from 50,000 to 60,000 feet.

Fig 9. Concord [14]

Performance: [14][28]
Maximum Speed: Mach 2.04 (1,350 mph, 2,170 km/h)
Range: 3,900 nm (4,500 mi, 7,250 km)
Service Ceiling: 60,000 ft (18,300 m)
Rate of Climb: 1,525 m (5,000 ft)/min (25,41 m/s)
Wing Loading: lb/ft² (kg/m²)
Thrust/Weight: .373
Fuel Consumption: 46.85 lb/mi (13.2 kg/km)
Maximum Nose Tip Temperature: 260 °F (127 °C)
Take-off speed: 250mph (400kph) >>Vstall = 208.33 mph = 305.50667ft/s
Cruising speed: 1,350mph (2,160kph/Mach Two) up to 60,000 ft
Landing speed: 187mph (300kph)
Length: 203ft 9ins (62.1m)
Wingspan: 83ft 8ins (25.5m)
Height: 37ft 1in (11.3m)
Fuselage width: 9ft 6ins (2.9m)
Fuel capacity: 26,286 Imperial gallons (119,500 litres)
Fuel consumption: 5,638 Imperial gallons (25,629 litres) per hour
Maximum take-off weight: 408,000lbs (185 tonnes)

Landing gear: 8 main wheels, 2 nose wheels.

Main problem during design of Concord

Neutral point shift in supersonic flight: When an aircraft reaches a supersonic speed, the center of pressure moves backward. This causes the aircraft to pitch downward as the center of gravity remains the same. To prevent this, the wings were designed to minimize the shift, but there was still some shift. Using trim controls could have corrected this, but it would have increased the drag on the aircraft at such high speeds. Instead, the fuel was redistributed along the aircraft during acceleration and deceleration to move the center of gravity and act as an additional trim control.

Heat problem during supersonic flight, the nose of any supersonic aircraft becomes extremely hot. Concord was built out of aluminum, but it could only endure a maximum temperature of 127°C, limiting the highest speed to Mach 2.02. During a flight, the aircraft went through two heating and cooling cycles. The heat generated by the pressure of the air would cause the fuselage to expand by nearly one foot. The fuel was used to drain heat from the air conditioning and cool the hydraulics to keep the cabin cold. Despite this, the windows in the cockpit became too hot to touch during supersonic flight.

Brake problem at takeoff, Concorde had to take off fast (around 250 miles per hour), so it needed strong brakes. Its brakes use anti-lock braking systems, which helped to keep the wheels from locking when fully engaged. This enabled the plane to slow down faster while maintaining control, especially in wet weather. If the take-off had to be canceled, the carbon brakes could bring Concorde (which weighed up to 185 tons / 188 tons at 190 mph) to a stop within one mile (1600 m). The brakes to become extremely hot (between 300°C and 500°C), require several hours of cooling time after use.

High angle of attack at takeoff and landing due to delta wing. Concord came up with droop nose, so the pilot could see during takeoff and landing.

Crash

On July 25, 2000, a metal strip caused the crash of Concorde by puncturing one of its tires. A piece of the disintegrated tire hit the fuel tank, causing it to leak and leading to the failure of one of the plane's four engines. The failure of the first engine caused the second engine to fail due to the inhalation of smoke and debris from the first engine. The failure of two engines created a side force on one side, causing the plane to skid and tilt, lose lift on the failure side, and ultimately turn upside down, resulting in the loss of all passengers and crew on board.

2. VALKYRIE XB-70 [15][16][17]

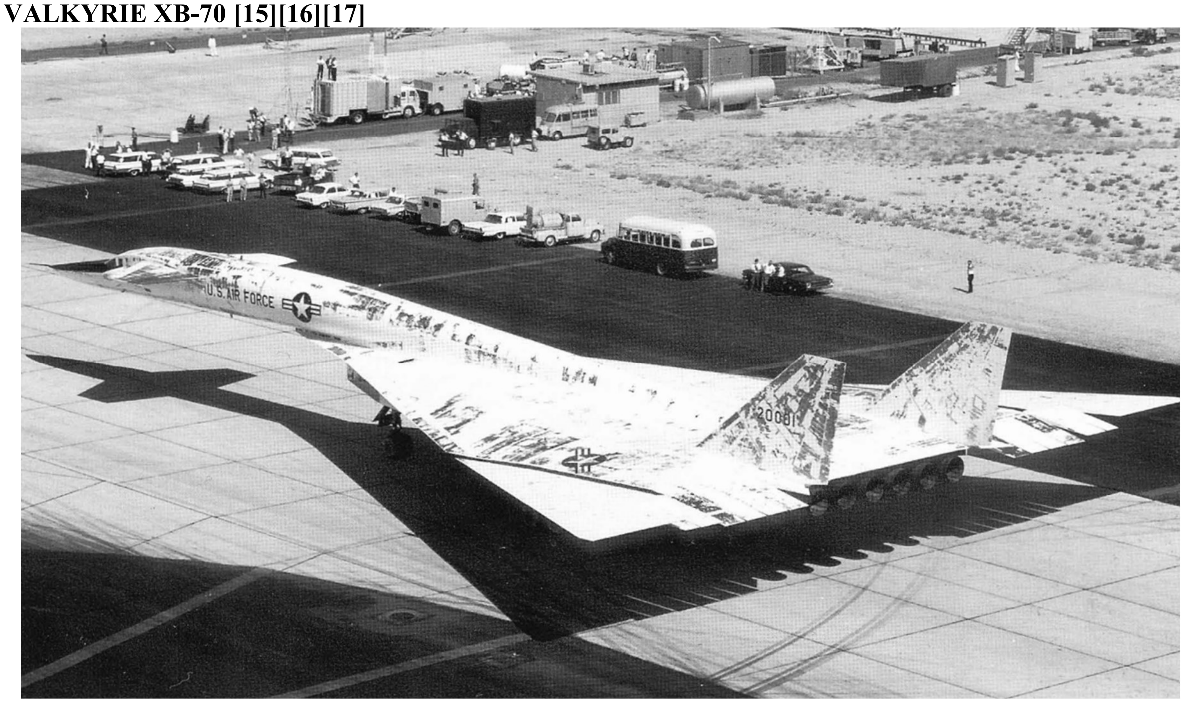


Fig. 10 XB-70 [15]

The XB-70 had exceptional performance capabilities, with a maximum design speed exceeding Mach 3 at altitudes up to 85,000 feet. It had a maximum taxi weight of 542,029 pounds and a maximum takeoff weight of 532,640 pounds, while the maximum landing weight was 524,209 pounds.

The XB-70 design includes canard and small canard's slap which to trim the aircraft, but it could also assist with pitch control in certain circumstances. The whole canard can deflect to six degrees, and the trailing edge could be lowered by up to 20 degrees to serve as a flap. The canard had a leading-edge sweep of 31.70 degrees and an area of 265.28 square feet, making the total area 415.59 square feet when the fuselage component was included. Wing of the XB-70 had aspect ratio of 1.751 with root chord of 117.75 ft and tip chord of 2.25 ft, with the swept angle of 65.56 degrees, planform area of 6297.17 ft. The wing was attached to the fuselage with a joint 80 feet long. The outer 40% of each wing could be folded downwards to improve directional stability at high speeds, and the wing tips could be positioned in three ways: UP, 1/2, and DOWN. The wing's elevons at the trailing edge were divided into six sections on each side to reduce the bending effects of air loads, and the XB-70As were equipped with a flight augmentation control system. If the manufacturer had not chosen to include the folding wingtips, the vertical stabilizers would have needed to be 467.92 square feet to provide the same level of directional stability. Vertical tails with a 51.76-degree angle and an area of 233.96 square feet. The rudders could move up to 12 degrees and had a hinge line at a 45-degree angle from the vertical tails. The folding wing tip helps increasing maneuverability in high speed by moving the neutral point forward. Folding wing tip also reduces drag in supersonic flight by limiting the area under the Mach cone.

Main problem design XB-70

XB-70 was designed to flight above Mach 3, but the titanium was scared at that time. XB-70 was built from PHI5-7-Mo stainless steel honeycomb sandwich for almost 69% of its airframe. At high speed, the aircraft suffered separation problems due to high temperature. Then the XB-70 was enhanced with 8% of titanium alloys.

The XB-70A's fuel tank sealing was also a severe issue, with leaks caused by pinhole faults where the tanks had been welded during manufacturing. North American chose to utilize Teflon as a sealant instead after several tries and errors.

General characteristics (Wikipedia)

Crew: 2
Length: 185 ft 0 in (56.39 m)
Wingspan: 105 ft 0 in (32.00 m)
Height: 30 ft 0 in (9.14 m)
Wing area: 6,297 sq ft (585.0 m2)
Airfoil: Hexagonal; 0.30 Hex modified root, 0.70 Hex modified tip
Empty weight: 253,600 lb (115,031 kg)
Gross weight: 534,700 lb (242,536 kg)
Max takeoff weight: 542,000 lb (245,847 kg)
Fuel capacity: 300,000 pounds (140,000 kg) / 46,745 US gal (38,923 imp gal; 176,950 l)
Powerplant: 6 × General Electric YJ93 afterburning turbojet, 19,900 lbf (89 kN) thrust each dry, 28,000 lbf (120 kN) with afterburner

Performance

Maximum speed: 1,787 kn (2,056 mph, 3,310 km/h)
Maximum speed: Mach 3.1
Cruise speed: 1,738 kn (2,000 mph, 3,219 km/h)
Take-off speed: 193mph = 283.067ft/s
Combat range: 3,725 nmi (4,287 mi, 6,899 km)
Service ceiling: 77,350 ft (23,580 m)
Lift-to-drag: about 6 at Mach 2
Wing loading: 84.93 lb/sq ft (414.7 kg/m2)
Thrust/weight: 0.314

Table 1. Comparison of similar supersonic aircraft and SR-71 [14][15][16]

	Concord	XB-70	SR-71
Length:	202 ft 4 in (61.66 m)		107.42 ft
Wing Span:	84 ft 0 in (25.6 m)	105 ft	55' 7"
Wing Area:	3,856 ft² (358.25 m²)	2482.34 ft²	1,795 ft. sq.
Wing Aspect Ratio:	1.85	1.751	1.939
Wing Root Chord:	90.75 ft (27.66m)	117.76 ft	60.533
Wing tip Chord		2.19 ft	0
Wing Dihedral Angle:	-	-	-
Wing Sweep:	55 degrees	65.57 degrees	52.629 degrees
Airfoil Section		0.30 to 0.70 HEX (MOD)	
Inboard Elevon Area:	172.2 sq. ft (16 sq. mtrs)	197.7 ft	39.00 ft. sq.
Outboard Elevon Area:	-		52.50 ft. sq.
Total Vertical Rudder Area:	365 ft. sq.	233.96 ft. sq.	150.76 ft. sq.
Moveable Rudder Area:	112 ft. sq.	191.11 ft. sq.	70.24 ft. sq.
Rudder Root Chord:	34' 8" (10.58m)	23.08 ft	14.803 ft.
Rudder Tip Chord:		6.92 ft	7.833 ft.
Height:	28 ft		18' 6"
Empty Weight:	173,500 lb (78,700 kg)	253,600 lb	59,000 lbs.
Maximum Weight:	418500 lb	537,109 lb	170,000 lbs.
Fuselage Diameter:	8 ft 7 in	8.92 ft	5.33 ft.
Maximum Speed:	Mach 2.04	Mach 3	Mach 3.3+
Cruising Speed:	Mach 2.04	Mach 3	Mach 3.2
Range:	3,900 nm (4,500 mi, 7,250 km)	5,700 nm)	3,200 nm

Table.2 Take off speed data of several aircraft [31]

Aircraft	Takeoff Weight	Takeoff Speed
Boeing 737	100,000 lb 45,360 kg	150 mph 250 km/h 130 kts
Boeing 757	240,000 lb 108,860 kg	160 mph 260 km/h 140 kts
Airbus A320	155,000 lb 70,305 kg	170 mph 275 km/h 150 kts
Airbus A340	571,000 lb 259,000 kg	180 mph 290 km/h 155 kts
Boeing 747	800,000 lb 362,870 kg	180 mph 290 km/h 155 kts
Concorde	400,000 lb 181,435 kg	225 mph 360 km/h 195 kts

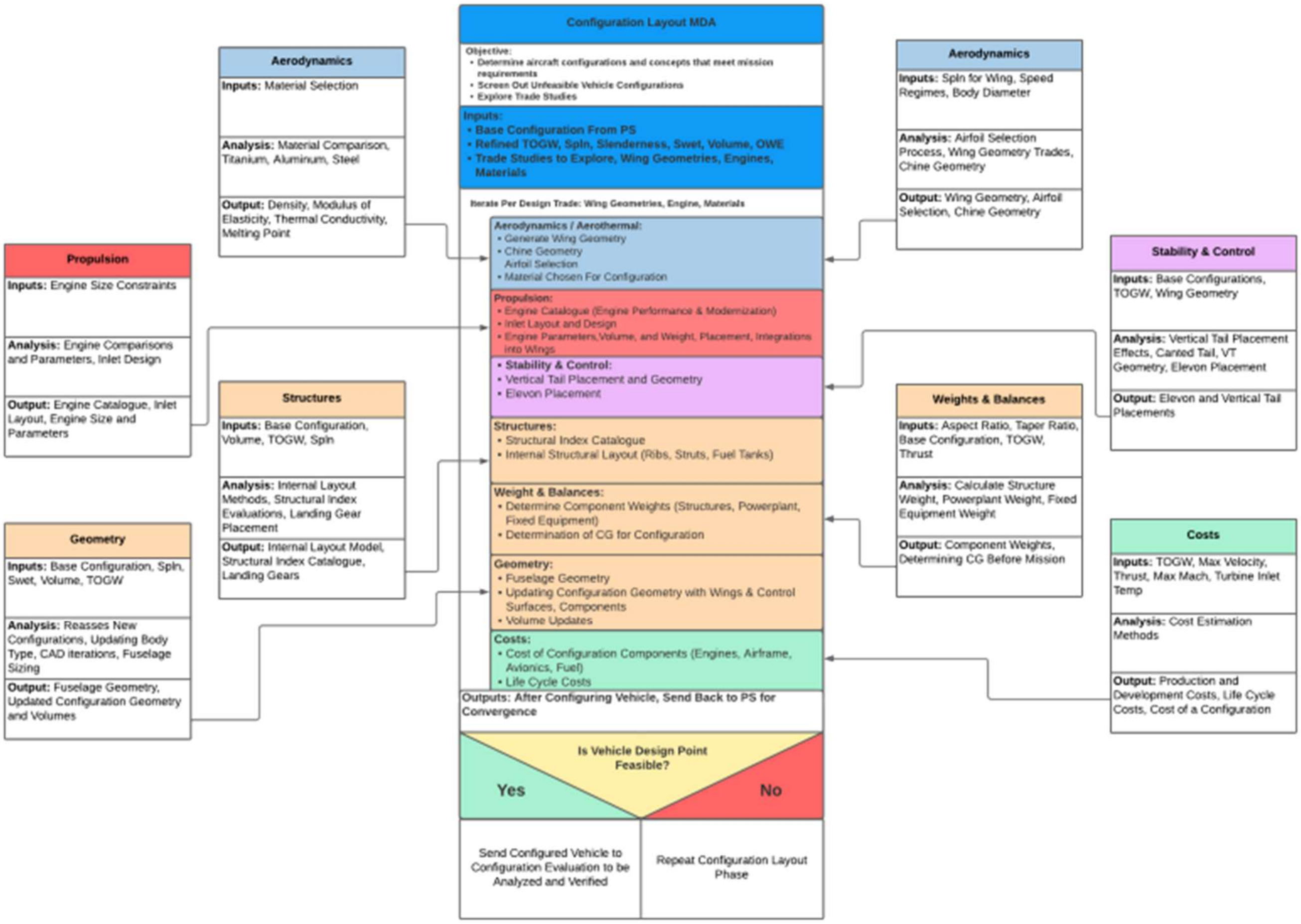


Fig. 11 Multi- Disciplinary Synthesis's MDA

III. Performance

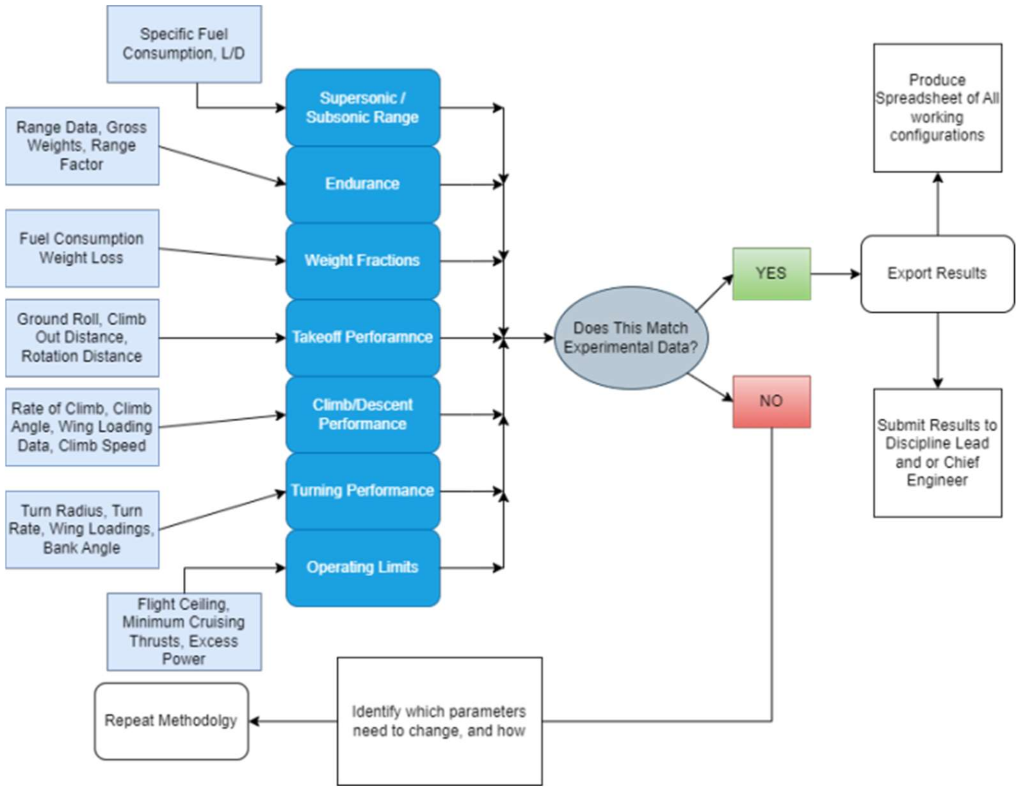


Fig. 12 Wes's Draft Interdisciplinary Analysis (IDA) for Performance

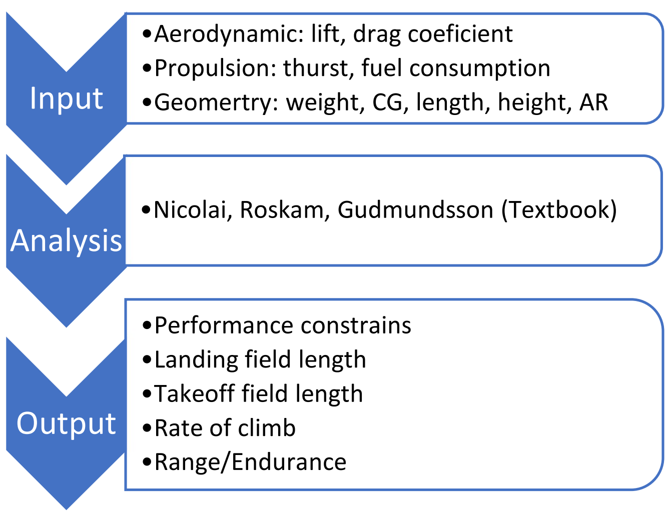


Fig. 13 Performance I-A-O

1. Parametric Sizing

All the performance constraints are in the function of thrust loading and wing loading.

The takeoff wing loading (W/S) $_{TO}$ is a very essential parameter that drives the aircraft performance. A trade study must be made to get the desired aircraft design, high value of the takeoff wing loading will give a better cruise efficiency, low value of the takeoff wing loading will give a better maneuverability.

Estimated takeoff gross weight (TOGW):

$TOGW = W_{pay} + W_{fuel} + OEW = 3000 + 80{,}000 + 57000 = 140000$ lb

$$\left(\frac{W}{S}\right)_{TO} = \frac{140000}{1800} = 77.77\ lb/ft^2$$

Performance is one of the most important factors that drive the sizing of the aircraft. Based on the general methodology of constraint analysis which requires performance characteristics of interests. An aircraft design must have a T/W that satisfies mission take off ground run distance, desired rate of climb, maximum angle of climb, level constant turn velocity, climbing constant velocity, cruise speed, service ceiling, and cruise L/D.

L/D is essential in range and loiter equations. L/D is highly dependent on the configuration. In subsonic, L/D is mainly dependent on wingspan and wetted area. In supersonic, the L/D is mostly affected by the drag. The L/D of the SR-71 is estimated by:

$$\frac{L}{D_{\max}} = K_{\mathrm{LD}}\sqrt{A_{\mathrm{wetted}}} = K_{\mathrm{LD}}\sqrt{\frac{A}{(S_{\mathrm{wet}}/S_{\mathrm{ref}})}}$$

where

$K_{\mathrm{LD}} = 15.5$ for civil jets
14 for military jets
11 for retractable prop aircraft
9 for nonretractable prop aircraft
13 for high-aspect-ratio aircraft
15 for sailplanes

[25]

1.1 Initial Performance Constraint Sizing (Roskam's Method)

Mission required altitude: 85000 ft, initial size of SR-71 starts with assumptions of based the XB-70 and Concorde specifications:

$C_{Dmin} = 0.016$,

Rate of climb = 119.5 ft/s

Stall speed = $\frac{V_{TO}}{1.2} \approx 208.33\ \text{mph} = 305.55\ \text{ft/s}$

Roskam's book, wetted area and C_{Do} can be estimated based on the takeoff weight:

$$\log_{10} S_{wet} = c + d\log_{10}TOGW$$
$$S_{wet} = 5729.46\ ft^2$$

Using Nicolai's method for estimating C_f in range from 0.004 to 0.006. Assume 0.005 for this estimation.

Table 3. Roskam's table a & b variable for C_{Do} estimation.

Equivalent Skin Friction Coefficient, c_f	a	b
0.0090	-2.0458	1.0000
0.0080	-2.0969	1.0000
0.0070	-2.1549	1.0000
0.0060	-2.2218	1.0000
0.0050	-2.3010	1.0000
0.0040	-2.3979	1.0000
0.0030	-2.5229	1.0000
0.0020	-2.6990	1.0000

$$\log f = a + b\log S_{wet}$$
$$f = 10^{-2.301+1*\log_{10}(5729.46)} = 28.6492$$

$$C_{Do} = f/S_{pln} = \frac{28.6492}{1800} = .015911$$

Where c = -1.1868 and d = 0.9609 are obtained from **Table. 3**

Table. 4 Regression Line Coefficients for Take-off Weight Versus Wetted Area [27]

Airplane Type	c	d
1. Homebuilts	1.2362	0.4319
2. Single Engine Propeller Driven	1.0892	0.5147
3. Twin Engine Propeller Driven	0. 8635	0.5632
4. Agricultural	1.0447	0.5326
5. Business Jets	0.2263	0.6977
6. Regional Turboprops	-0.0 866	0. 8099
7. Transport Jets	0.0199	0.7531
8. Military Trainers*	0.8565	0.5423
9. Fighters*	-0.1289	0.7506
10. Mil. Patrol, Bomb and Transport	0.1628	0.7316
11. Flying Boats, Amph. and Float	0.6295	0.6708
12. Supersonic Cruise Airplanes	-1.1868	0.9609

Table. 5 C_{Lmax} Range. [27]

Airplane Type	$C_{L_{max}}$	$C_{L_{max_{TO}}}$	$C_{L_{max_L}}$
1. Homebuilts	1.2 - 1.8	1.2 - 1.8	1.2 - 2.0*
2. Single Engine Propeller Driven	1.3 - 1.9	1.3 - 1.9	1.6 - 2.3
3. Twin Engine Propeller Driven	1.2 - 1.8	1.4 - 2.0	1.6 - 2.5
4. Agricultural	1.3 - 1.9	1.3 - 1.9	1.3 - 1.9
5. Business Jets	1.4 - 1.8	1.6 - 2.2	1.6 - 2.6
6. Regional TBP	1.5 - 1.9	1.7 - 2.1	1.9 - 3.3
7. Transport Jets	1.2 - 1.8	1.6 - 2.2	1.8 - 2.8
8. Military Trainers	1.2 - 1.8	1.4 - 2.0	1.6 - 2.2
9. Fighters	1.2 - 1.8	1.4 - 2.0	1.6 - 2.6
10. Mil. Patrol, Bomb and Transports	1.2 - 1.8	1.6 - 2.2	1.8 - 3.0
11. Flying Boats, Amphibious and Float Airplanes	1.2 - 1.8	1.6 - 2.2	1.8 - 3.4
12. Supersonic Cruise Airplanes	1.2 - 1.8	1.6 - 2.0	1.8 - 2.2

<u>Stall Constraint</u>

From Table. 4 CL_{Max} for supersonic cruise SR-71 can be assumed = 2.0 for initial sizing.

$$V_{stall} = \sqrt{\frac{W}{S}\frac{2}{\rho C_{Lmax}}}$$

$$\frac{W}{S} = \frac{V_{stall}^2}{2}\rho C_{Lmax}$$

Landing Distance Constraint

Edwards's fielding = 15000ft, so $S_{LFL\ max} = 15000ft, C_{LmaxL} = 2$

$$S_{LFL} = \left[.01583 + 1.556\tau\sqrt{\frac{\rho C_{Lmax}}{\frac{W}{S}}} + \frac{1.21}{g\left[\frac{.605}{C_{Lmax}}(C_{D\ LDG} - \mu C_{L\ LDG}) + \mu - \frac{T_{gr}}{W}\right]}\right] * \frac{W/S}{\rho C_{Lmax}} \qquad [23]$$

Where $C_{D\ LDG} = .016$(assumed), $C_{L\ LDG} = 2$(assumed) lift and drag coefficient during ground roll.
$\mu(= 0.3)$ is ground friction coefficient during ground braking
$\frac{Tgr}{W}$ is thrust loading during ground roll, where T is idle or reverse thrust (assumed $= 0$)

Take-off Field Length Constraint

From Table. 4, assumed $CL_{TO} = 1.8$, $CL_{Max} = 1.5$, $C_{D\ TO} = 0.016$, $\mu = 0.04$

$$\frac{T}{W} = \frac{1.21}{g\rho C_{Lmax}S_G}\left(\frac{W}{S}\right) + \frac{.605}{C_{Lmax}}(C_{D\ TO} - \mu C_{L\ TO})\mu$$

[23]

Climb Gradient Constraint

Assumed $V_{\infty\ climb}$ = 1,850 knots = 3122.45ft/s (Mach 3), $\Lambda_{LE} = 60$, $AR = 1.9$
Lift-induced drag constant k = $\frac{1}{\pi ARe}$ =.211904274
Span efficiency e $= 4.61(1 - 0.045AR^{0.68})\cos(\Lambda_{LE})^{0.15} - 3.1 = 0.7906443368$

$$\frac{T}{W} = \frac{Vv}{V_\infty} + \frac{\frac{1}{2}\rho {V_\infty}^2}{W/S}C_{Dmin} + k\left(\frac{W}{S}\right)$$

[23]

Cruise Constraint

Mission requirements Mach 3 at altitude 85000 ft. $V_{\infty\ cruise}$ = Mach 3 = 3226.667 ft/s

$$\frac{T}{W} = qC_{Dmin}\left(\frac{1}{W/S}\right) + k\left(\frac{1}{q}\right)\left(\frac{W}{S}\right)$$

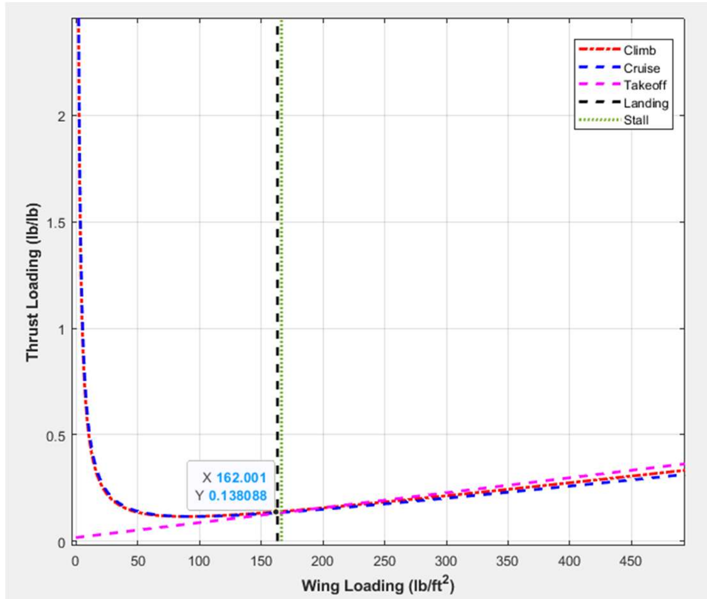


Fig. 14 Initial sizing solution space.

Constraint plots are computed based on mission requirements and assumptions based on the Concorde and XB-70 coefficients. From the plot, the optimum design point can be obtained at T/W = 0.138 and W/S = 162.001 (lb/ft^2).

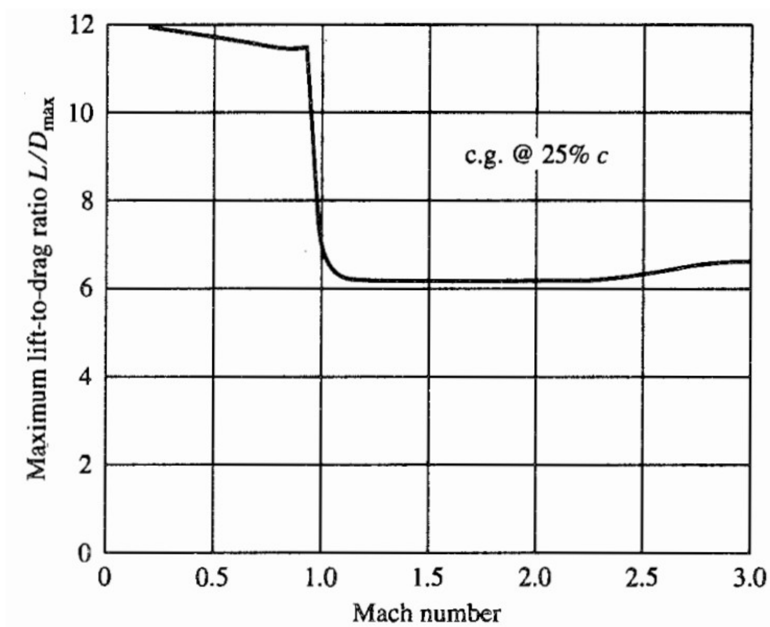


Fig. 15 L/D as a function of free-stream Mach number for the Blackbird SR-71 (Anderson's Aircraft Performance & Design)

$$\left(\frac{T}{W}\right)_{cruise} = \frac{1}{L/D} \quad [23]$$

From the optimum point the lift drag ratio at cruise (Mach 3) is estimated around 7.2417 which is very close to the SR-71 data. With estimated takeoff cross weight (142000lb) and W/S (=162.001 lb/ft^2), the initial minimum wing area is estimated: 876.5 ft^2.

Table. 7 Roskam's suggested fuel fraction.

Mission Phase No. (See Fig.2.1) Airplane Type:	Engine Start, Warm-up	Taxi	Take-off	Climb	Descent	Landing Taxi, Shutdown
	1	2	3	4	7	8
1. Homebuilt	0.998	0.998	0.998	0.995	0.995	0.995
2. Single Engine	0.995	0.997	0.998	0.992	0.993	0.993
3. Twin Engine	0.992	0.996	0.996	0.990	0.992	0.992
4. Agricultural	0.996	0.995	0.996	0.998	0.999	0.998
5. Business Jets	0.990	0.995	0.995	0.980	0.990	0.992
6. Regional TBP's	0.990	0.995	0.995	0.985	0.985	0.995
7. Transport Jets	0.990	0.990	0.995	0.980	0.990	0.992
8. Military Trainers	0.990	0.990	0.990	0.980	0.990	0.995
9. Fighters	0.990	0.990	0.990	0.96-0.90	0.990	0.995
10. Mil.Patrol, Bomb, Transport	0.990	0.990	0.995	0.980	0.990	0.992
11. Flying Boats, Amphibious, Float Airplanes	0.992	0.990	0.996	0.985	0.990	0.990
12. Supersonic Cruise	0.990	0.995	0.995	0.92-0.87	0.985	0.992

With previous estimated optimum L/D and estimated empty weight (59000lb) fuel fraction:

$$phrase\ 1(Start):\ \frac{W_1}{W_{TO}} = .990 \qquad phrase\ 5(cruise):\ \frac{W_5}{W_4} = 0.754$$

$$phrase\ 2(Taxi):\ \frac{W_2}{W_1} = .995 \qquad phrase\ 6(Loiter):\ \frac{W_6}{W_5} = 1$$

$$phrase\ 3(Takeoff):\ \frac{W_3}{W_2} = .995 \qquad phrase\ 7(Descent):\ \frac{W_7}{W_6} = .985$$

$$phrase\ 4(Climb):\ \frac{W_4}{W_3} = .895 \qquad phrase\ 8(Landing:\ \frac{W_8}{W_7} = .992$$

No loiter required in the mission, so loiter weight fraction is assumed 1. Phase 1-4 and 7,8 weight fractions are based on the Roskam's suggested table, and phrase 5 is calculated based on Roskam's method by Wes.

$$W_{final} = \frac{W_1}{W_{TO}}\frac{W_2}{W_1}\frac{W_3}{W_2}\frac{W_4}{W_3}\frac{W_5}{W_4}\frac{W_6}{W_5}\frac{W_7}{W_6}\frac{W_8}{W_7} * TOGW = 91763.927\ lb$$

So, the total fuel weight estimate = 54250,237 lb.

2. Configuration Layout

With the initial sizing from the Mach cone, now aircraft wing is sized based on the stall speed. The maximum lift coefficient is picked from range 1.7 to 2.4 and the wing area from 1000 to 2000 square feet.

$$V_{stall} = \sqrt{\frac{W}{S}\frac{2}{\rho C_{L_{max}}}}$$

Table 6. Stall speed at takeoff estimation based on wing area and CL_{max}

CL max | Wing area[ft^2]	1400	1500	1600	1700	1800	1900	2000	2100	2200
0.8	324.784901	313.772	303.808456	294.737	286.433	278.794	271.735	265.186	259.089
0.9	306.210141	295.8271	286.433359	277.881	270.052	262.849	256.194	250.02	244.271
1	290.496447	280.6462	271.734544	263.621	256.194	249.361	243.047	237.189	231.736
1.1	276.977494	267.5857	259.088722	251.353	244.271	237.756	231.736	226.151	220.952
1.2	265.185761	256.1938	248.058566	240.652	233.872	227.634	221.87	216.523	211.545
1.3	254.782238	246.143	238.326961	231.211	224.697	218.704	213.166	208.029	203.246
1.4	245.514308	237.1894	229.657606	222.801	216.523	210.748	205.412	200.462	195.853
1.5	237.189356	229.1467	221.870326	215.246	209.181	203.602	198.447	193.664	189.212
1.6	229.657606	221.8703	214.82502	208.411	202.539	197.137	192.145	187.515	183.203
1.7	222.800604	215.2458	208.410882	202.188	196.492	191.251	186.408	181.916	177.733
1.8	216.523267	209.1813	202.538971	196.492	190.956	185.863	181.156	176.791	172.726
1.9	210.748273	203.6022	197.136958	191.251	185.863	180.905	176.325	172.075	168.119
2	205.412008	198.4469	192.145339	186.408	181.156	176.325	171.86	167.718	163.862
2.1	200.461593	193.6643	187.51465	181.916	176.791	172.075	167.718	163.676	159.913
2.2	195.852665	189.2116	183.203392	177.733	172.726	168.119	163.862	159.913	156.236
2.3	191.547685	185.0526	179.176452	173.827	168.929	164.424	160.26	156.398	152.802

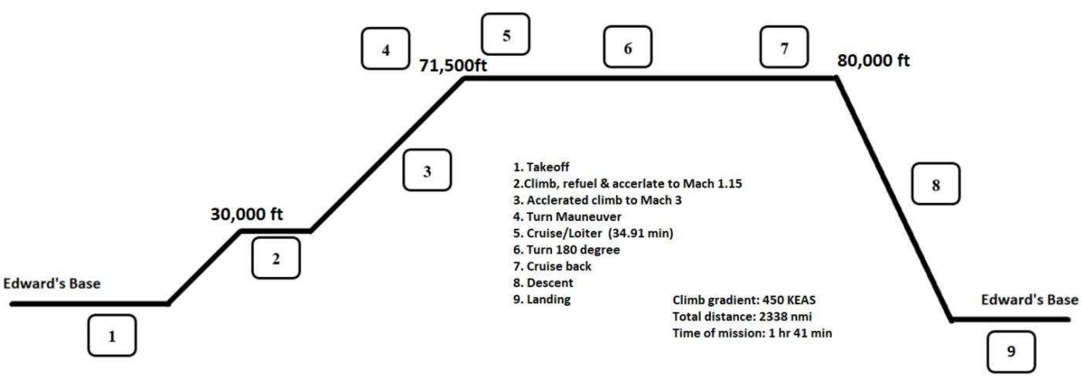


Fig. 16 Performance Map (made with Wesly)

3. Configure Evaluation

From aerodynamic the C_{Lmax} is approximate 1.4, Wing area = 1800 ft^2

3.1 Takeoff speed:

$$V_{TO} = 1.2\,V_{Stall} = 1.2 * 216.5 = 259.8\,ft/s$$

3.2 Takeoff distance

Ground run

Assume takeoff thrust = 32,500 lbf, $take\ off\ drag = 6029.8\ lbf\ based\ on\ estimaited\ aerodynamic\ C_D$

$$S_G = \frac{V_{TO}^2}{2a} = \frac{259.8^2}{\frac{32.2}{140000}[65000 - 6029.8 - .04(140000 - 516840)]} = 3963.34\,ft$$

Rotation distance

$$S_{ROT} = V_{TO} = 269.8\,ft$$

Transition distance

$$S_{TR} = Rsin\theta_{climb} = .2156V_s^2 * \sin(3 \to 20^o) = 10105.56 * \sin(3 \to 20^o) = 528.88ft \to 3456.33ft$$

Total field length with no obstacle assumption:

$$S_{TOFL} = S_G + S_{ROT} + S_{TR} = 7689.14\,ft$$

3.4 Landing speed

Approach speed: $V_{Flare} = 1.3\,V_{Stall} = 281.45\,ft/s$

Touch down speed: $V_{TD} = 1.1\,V_{Stall} = 238.15\,ft/s$

3.5 Landing distance

Approach distance, assume obstacle height = 50 ft, with landing weight = 91763.927 lb, $\theta_{app} = 3^o$, drag = 7097lbf

$$S_A = \frac{h_{obst} - h_F}{\tan\theta_{app}} = \frac{50 - .1512(V_{app}^2)(1 - \cos\theta_{app})}{D/W} = 434.26\,ft$$

Flare distance

$$S_F = Rsin(\theta_{app}) = .1512V_{Stall}^2 \sin(\theta_{app}) = 370.9\,ft$$

Free roll distance

$$S_{FR} = V_{TD} = 238.15\,ft$$

Braking distance

$$S_{BR} = \frac{V_{TD}^2 W}{2g[T - D_{land} - \mu(W - L)]_{0.7V_{BR}}} = 2265.137\,ft$$

Total field length for landing:

$$S_{LFL} = S_A + S_F + S_{FR} + S_{BR} = 3308.44\,ft$$

The total landing and takeoff field length of the configuration is not over limit of the Edward field length, which is 14,000ft.

3.6 Rate of Climb

$$ROC = \frac{(T - D)V_\infty}{W}$$

Table 8. Subsonic rate of climb

Alt	ρ	CDo	V(ktas)	V(ft/s)	CL	CD	TR	PR(ft.lb/s)	HP_R	HP_A	R/C(ft/s)	ROC(ft/min)
10000	0.001756	0.01591	100	300	0.984237838	0.234927306	33416.53973	10024961.92	18227.20349	24006.73745	22.70531201	1362.318721
θ=	0.931244		150	400	0.553633784	0.085208444	21547.06336	8618825.344	15670.59153	32008.98327	64.18653897	3851.192338
δ=	0.687693		200	500	0.354325622	0.044294643	17501.55681	8750778.405	15910.50619	40011.22909	94.68141139	5680.884683
T_alt	44012.35		250	600	0.246059459	0.029598582	16840.65077	10104390.46	18371.61902	48013.47491	116.4501481	6987.008887
b	57		300	700	0.180778378	0.023298755	18043.22944	12630260.61	22964.1102	56015.72073	129.8456128	7790.736767
AR	1.805		350	800	0.138408446	0.020241153	20473.90511	16379124.09	29780.22562	64017.96655	134.5054108	8070.324648
S	1800		400	900	0.10935976	0.018613917	23829.13455	21446221.1	38993.12927	72020.21236	129.749255	7784.955301
W	140000		450	1000	0.088581405	0.01768404	27949.04431	27949044.31	50816.44421	80022.45818	114.7379121	6884.274723
e	0.78		500	1100	0.073207773	0.017121693	32742.93102	36017224.13	65485.86205	88024.704	88.54545053	5312.727032
T_A	64000		550	1200	0.061514865	0.016765536	38156.22605	45787471.26	83249.94775	96026.94982	50.19536527	3011.721916

Table 9. Supersonic rate of climb

Alt	ρ	CDo	V(ft/s)	CL	CD	TR	PR(ft.lb/s)	HP_R	HP_A	R/C(ft/s)	ROC(ft/min)
40000	5.85E-04	0.135	1115.17	8.06E-01	0.714345896	468033.1736	521936759	948975.9255	24017.6082	-3633.764818	-218025.8891
θ=	0.751865		1227.92	8.46E-01	0.501320269	398233.4536	488996989.2	889085.435	26445.81483	-3388.941365	-203336.4819
δ=	0.185085		1339.54	7.66E-01	0.40517079	383034.3534	513091396.4	932893.448	28849.97982	-3551.599339	-213095.9604
T_alt	11845.44		1451.17	7.42E-01	0.347828082	385912.1499	560025192.6	1018227.623	31254.1448	-3877.395807	-232643.7484
b	57		1562.8	7.17E-01	0.308514853	396980.5728	620401800.5	1128003.274	33658.30979	-4299.212358	-257952.7415
AR	1.805		1674.43	6.93E-01	0.279397352	412707.0704	691049135.7	1256452.974	36062.47477	-4794.391247	-287663.4748
S	1800		1786.06	6.69E-01	0.256706478	431433.5354	770565645	1401028.445	38466.63976	-5352.921379	-321175.2828
W	140000		1897.69	0.644906448	0.238370541	452259.3247	858246836.2	1560448.793	40870.80474	-5969.770669	-358186.2401
e	0.78		2009.32	6.21E-01	0.223141925	474638.86	953699505	1733999.1	43274.96973	-6642.130512	-398527.8307
T_A	64000		2120.94	5.97E-01	0.210217381	498210.56	1056677085	1921231.063	45679.13471	-7368.239719	-442094.3831
			2232.57	5.97E-01	0.210217381	552033.8614	1232456142	2240829.349	48083.2997	-8614.359479	-516861.5688
			2232.57	5.62E-01	0.19874073	521896.012	1165171179	2118493.054	48083.2997	-8133.752604	-488025.1563

For the rate of climb for supersonic phrase shows the negative value because there was no after burner thrust assumption.

3.7 Range and endurance

Range and endurance for cruise phrase can be expressed:

$$R = 2\sqrt{\frac{2}{\rho_\infty S}}\frac{C_L^{\frac{1}{2}}}{C_D}\left(W_o^{0.5} - W_1^{0.5}\right)$$

$$E = \frac{1}{c_t}\frac{C_L}{C_D}\ln\left(\frac{W_o}{W_1}\right)$$

The determination of thrust specific fuel consumption and CD for the entire airplane is currently underway. Upon completion of these assessments, we will conduct an evaluation of the aircraft's range and endurance.

IV. Aerodynamic

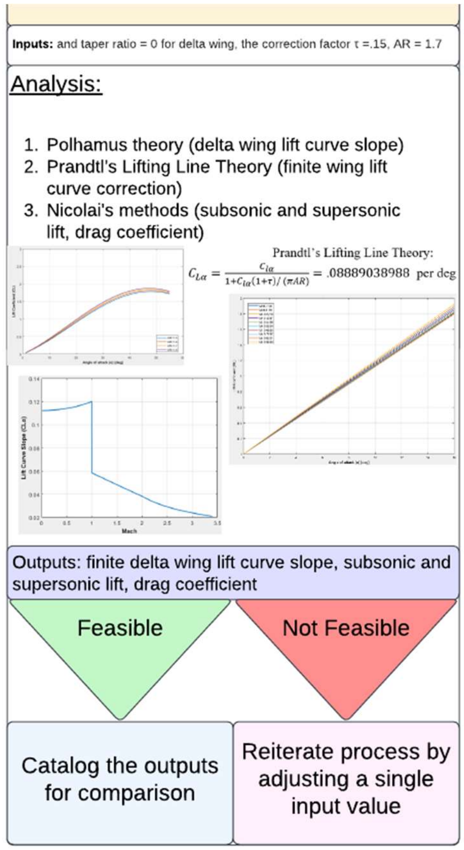


Fig 17. IDA Aerodynamic

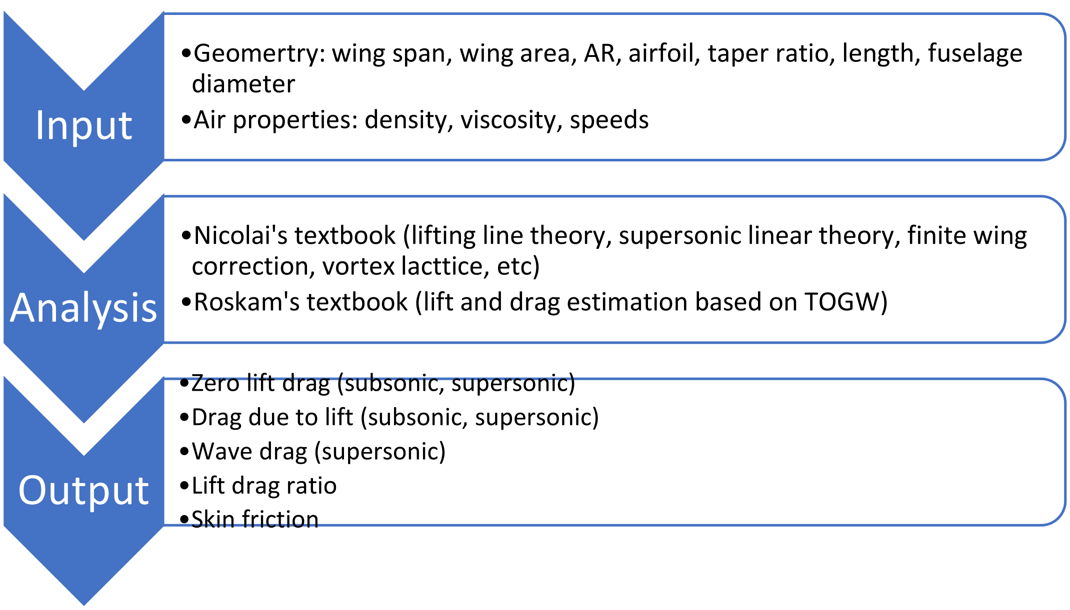


Fig 16. I-A-O Aerodynamic

1. Parametric Sizing

1.1 Wing sizing based on C_{Lmax}

Using Roskam's C_{Lmax} estimation, C_{Lmax} is in range 1.2 to 1.8 for supersonic cruise vehicle. From the historical data is usually in range of 200ft/s to 400ft/s (Concord =305ft/s, XB-70 =283ft/s). Supersonic aircraft with delta wing usually has higher take off speed. With the constant estimated weight, stall speed and C_{Lmax} are vary, the wing area can be estimated by:

$$S_{pln} = \frac{2W}{V_{stall}^2 \rho C_{Lmax}}$$

Table 10. Wing area based on C_{Lmax}

Vstall [ft/s]	Clmax	1.2	1.3	1.4	1.5	1.6	1.7	1.8
200	Wing area (ft^2)	2453.04	2264.35	2102.61	1962.43	1839.78	1731.56	1635.36
220		2027.31	1871.36	1737.69	1621.85	1520.48	1431.04	1351.54
240		1703.5	1572.46	1460.14	1362.8	1277.63	1202.47	1135.67
260		1451.5	1339.85	1244.15	1161.2	1088.63	1024.59	967.669
280		1251.55	1155.28	1072.76	1001.24	938.664	883.448	834.368
300		1090.24	1006.38	934.492	872.193	817.681	769.582	726.827
320		958.219	884.51	821.331	766.576	718.665	676.39	638.813
340		848.803	783.511	727.546	679.043	636.603	599.155	565.869
360		757.112	698.872	648.953	605.689	567.834	534.432	504.741
380		679.513	627.243	582.44	543.61	509.635	479.656	453.009
400		613.26	566.087	525.652	490.608	459.945	432.89	408.84

1.2 Subsonic Lift

Picking airfoil
NACA 64-006 is picked as initial input for C_L estimation, assume standard roughness $C_{l\alpha} = 0.090625$, and taper ratio = 0 for delta wing correction factor $\tau = .15$ [13], AR = 1.7.

Prandtl's Lifting Line Theory:

$$C_{L\alpha} = \frac{C_{l\alpha}}{1 + C_{l\alpha}(1+\tau)/(\pi AR)} = .08889038988 \text{ per deg [23 (9-55)]}$$

With same method, apply for $C_{l\alpha}$ with 60^o flap deflect at location 20% chord.

$$C_{L\alpha\, max} = \frac{C_{l\alpha}}{1 + C_{l\alpha}(1+\tau)/(\pi AR)} = .18047 \text{ per deg [23]}$$

Lift curve slope based on Nicolai method:
Assume lift of the maximum thickness line $\Delta = 60^o$
Lift curve slope for subsonic regime:

$$C_{L\alpha} = \left(\frac{2\pi AR}{2 + \sqrt{4 + AR^2\beta^2(1 + [(\tan^2\Delta)/\beta^2])}} \right) / \alpha_{stall}$$

$$\beta = \sqrt{1 - M_\infty^2}$$

Table 11. CLα (per deg) as function of Mach and Aspect Ratio

Mach | AR	1.2	1.3	1.4	1.5	1.6	1.7	1.8
0	0.09197	0.09668	0.10104	0.10508	0.10883	0.1123	0.11553
0.111	0.09202	0.09674	0.10111	0.10516	0.10891	0.11239	0.11562
0.222	0.09217	0.09692	0.10131	0.10537	0.10914	0.11264	0.1159
0.333	0.09243	0.09721	0.10164	0.10574	0.10954	0.11308	0.11637
0.444	0.09279	0.09763	0.10211	0.10626	0.11011	0.1137	0.11703
0.555	0.09327	0.09818	0.10272	0.10694	0.11086	0.11451	0.1179
0.666	0.09387	0.09886	0.10349	0.1078	0.1118	0.11553	0.119
0.777	0.09459	0.09969	0.10443	0.10884	0.11295	0.11677	0.12035
0.888	0.09546	0.10068	0.10555	0.11009	0.11432	0.11828	0.12197
0.999	0.09647	0.10185	0.10688	0.11157	0.11596	0.12006	0.1239

Lift coefficient:

$$C_L = C_{L\alpha}\alpha + C_1\alpha^2 \quad [11]$$

Where C_1 = 4.3 for delta wing with aspect ratio =1.7.

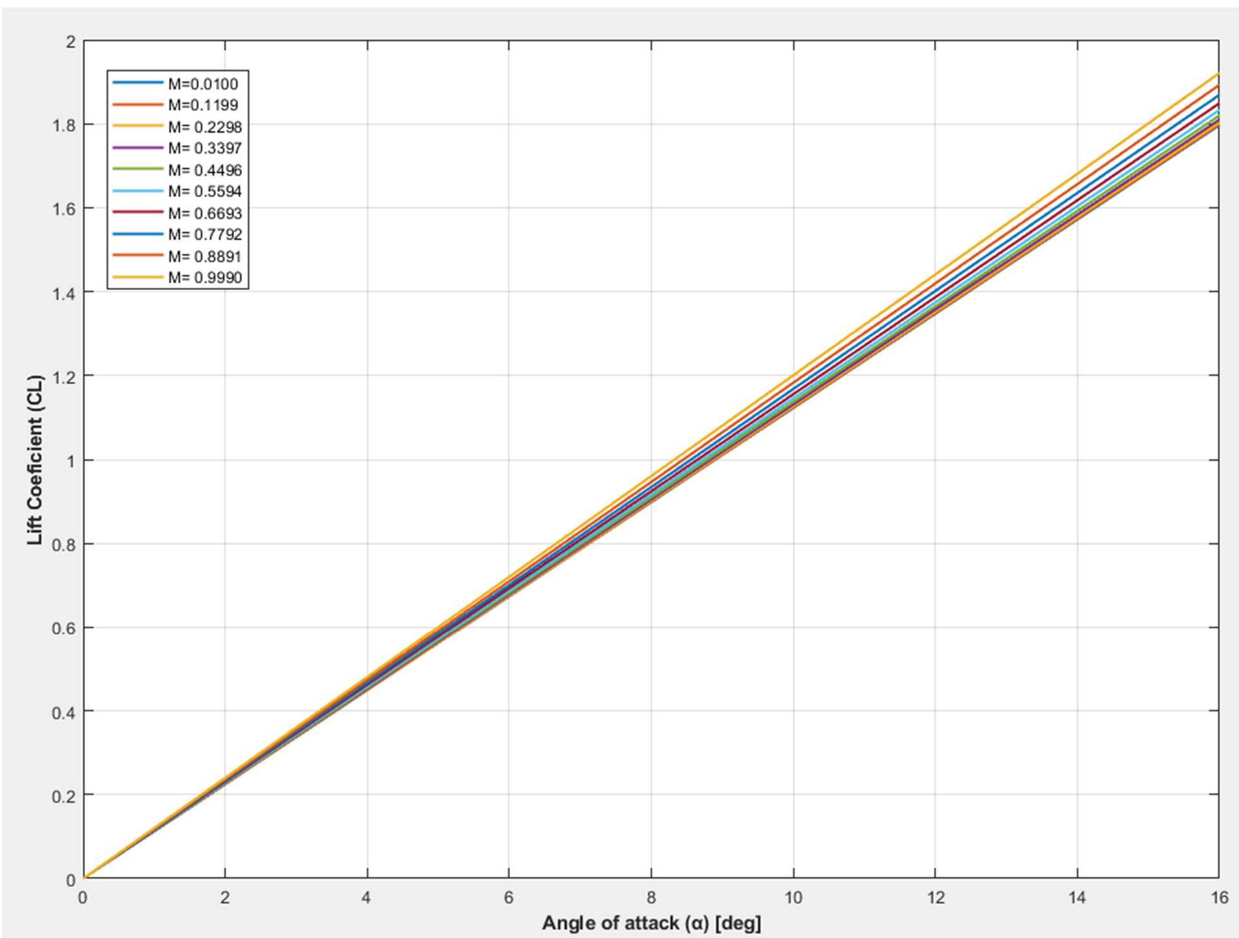


Fig 18. Subsonic linear lift coefficient for AR=1.7 as function of Mach number and angle of attack.

Based on Polhamus theory, lift coefficient of the delta wing can be estimated by

$$C_L = K_p \sin\alpha \cos^2\alpha + K_v \cos\alpha \sin^2\alpha \quad [20]$$

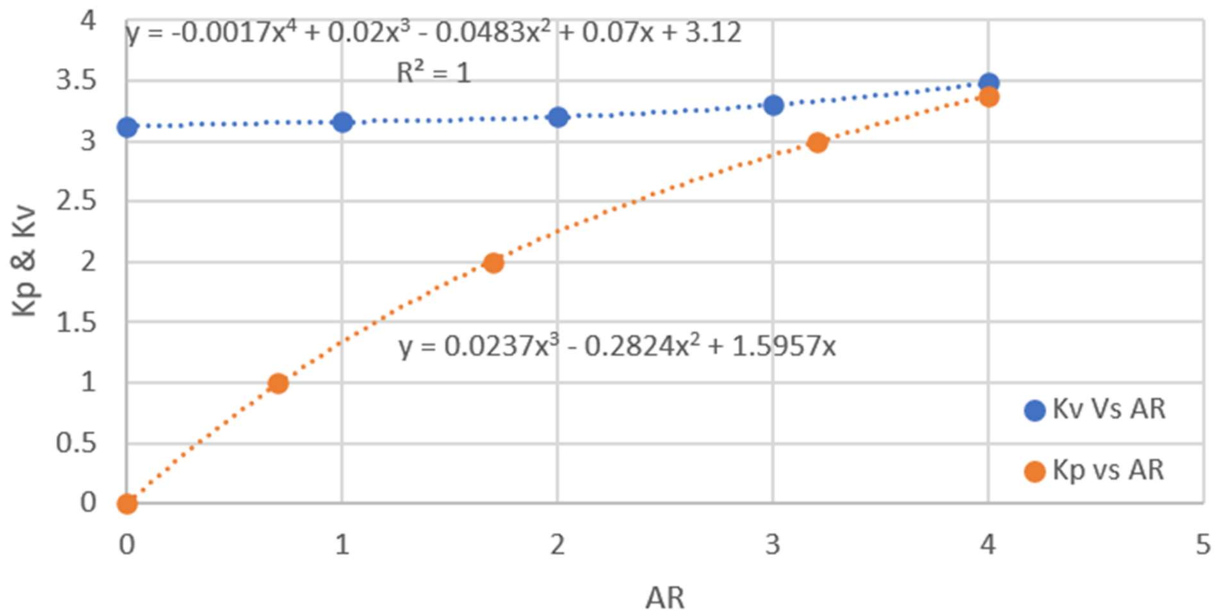


Fig. 19 Modified K_v and K_p curves based on Polhamus's paper.

Based on literature research, and previous estimations aspect ratio of supersonic aircrafts fall into range of 1.5-2.

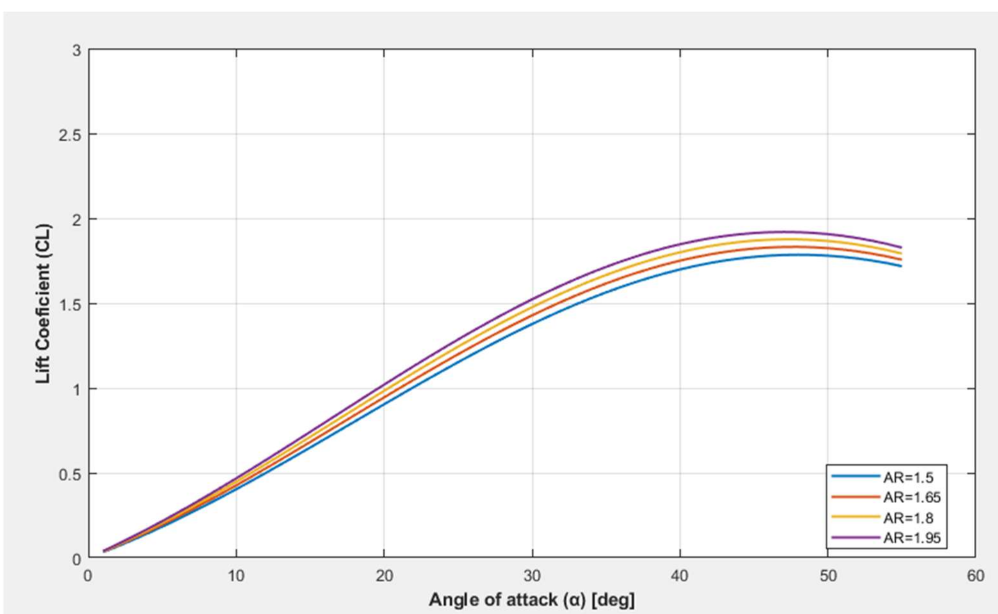


Fig. 20 Lift Curve Slopes for Delta Wing with Varying Aspect Ratio (AR)

1.3 Supersonic Lift

Supersonic lift curve slope is estimated using supersonic linear theory:

Assume wing swept 60^o, AR =1.7, taper ratio =0

AR $tan\wedge_{LE} = 2.9448$

$$\frac{tan\wedge_{LE}}{\sqrt{M^2-1}}$$

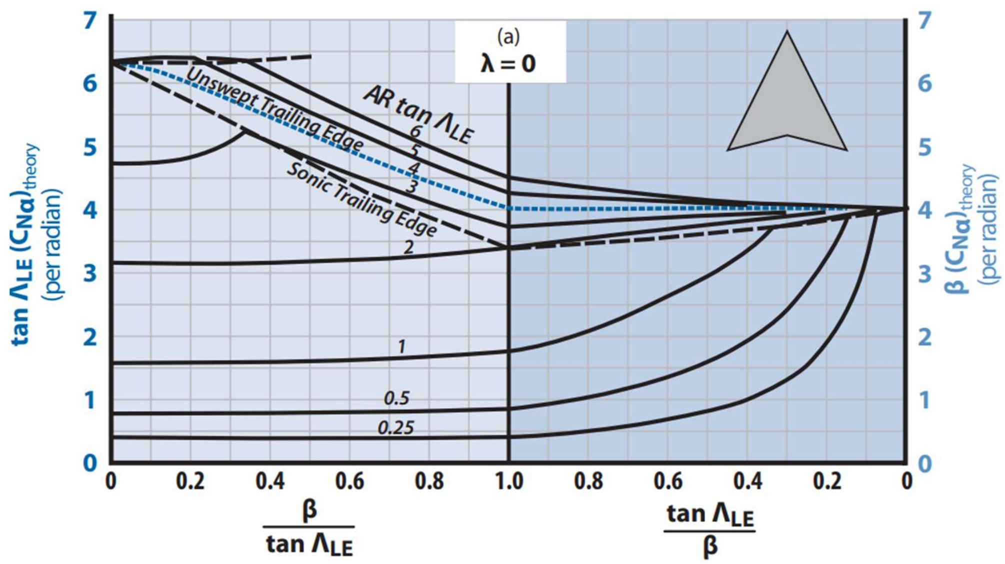


Fig 21. Nicolai's Theoretical wing lift curve slope [11]

The slope of the normal force coefficient (CN) is equivalent to the lift coefficient (CL) at small to moderate angles of attack. Lift curve slope as function of Mach number is developed based supersonic linear theory corrected for three-dimensional flow.

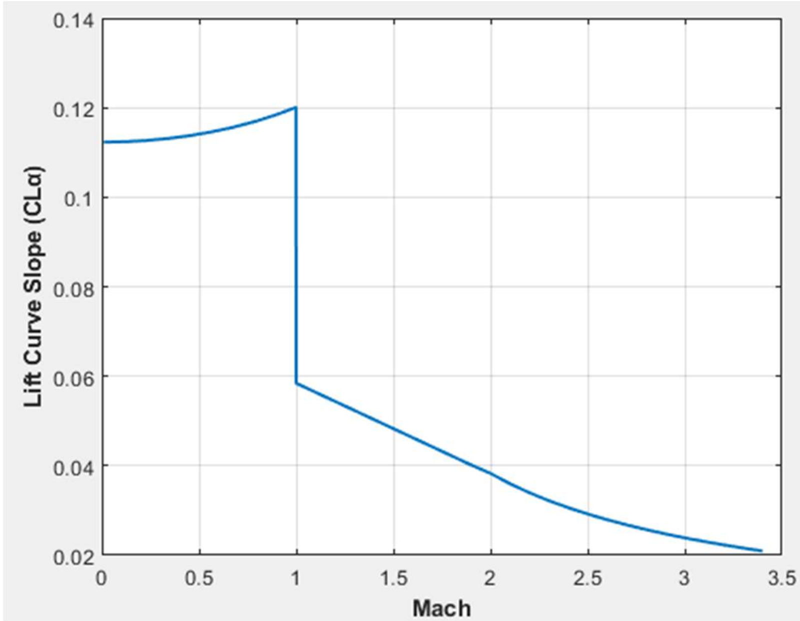


Fig 22. Lift curve slope for AR=1.7 as function of Mach number.

1.4 Subsonic Drag

The total drag coefficient for a wing in subsonic flow is expressed as a sum of terms, including the drag-due-to-lift. The zero-lift drag coefficient (C_{D_0}) is approximately minimum drag coefficient (C_{Dmin})

$$C_D = C_{D_0} + KC_L^2 \quad [11]$$

$$where\ K = \frac{1}{\pi ARe} \quad [11]$$

$$C_{Do} \approx 1.2C_{DF} \quad [11]$$

$$C_{DF} = C_F \frac{S_{wet}}{S_{ref}} \quad [11]$$

$$C_F = \frac{1.328}{\sqrt{Re_l}} \quad (laminar\ Re < 5 * 10^5) \qquad [11]$$

$$C_F = \frac{0.455}{[\log_{10} Re_l]\text{^}2.58} \quad (turbulent\ Re > 5 * 10^5) \quad [11]$$

$$e = 4.61(1 - 0.045AR^{0.68})\left(cos(\Lambda_{LE})\right)^{0.15} - 3.1 \ = .7865 \quad [25]$$

Pick planform area = 1850 ft^2, mean geometric chord of the delta wing is approximately $C_{MGC} = \frac{2}{3}c_r\frac{1+\lambda+\lambda^2}{1+\lambda} \approx 33.3\ ft$ with assumption $\lambda = 0$ and $c_r = 50ft$.

$Air\ viscosity\ is\ computed\ by\ Sutherland's\ formula$ $\mu = 1.79 * 10^{-5} * \left(\frac{T}{T_0}\right)^{1.5}\left(\frac{T_0+198.72}{T+198.72}\right)$

for Reynold number calculation. From that the C_F of the wing can be estimated (MATLAB Code Appendix):

Table 12. Subsonic drag estimation as function of Mach number

$Mach$	0.01	0.119888889	0.229777778	0.339666667	0.449555556	0.55944	0.66933	0.77922	0.88911	0.999
C_F	0.030449402	0.008794057	0.006352209	0.005224591	0.004541372	0.00407	0.00372	0.00345	0.00323	0.00305
C_{DF}	0.094301962	0.027235243	0.019672824	0.016180587	0.014064654	0.01261	0.01153	0.01068	0.01	0.00943
C_{Do}	0.113162354	0.032682292	0.023607389	0.019416704	0.016877584	0.01513	0.01383	0.01282	0.012	0.01132
C_D	0.116164818	0.035690048	0.026629497	0.022462518	0.019956958	0.01825	0.01701	0.01607	0.01533	0.01475

1.5 Supersonic Drag

Supersonic drag is estimated using supersonic thin airfoil theory, with assumption biconvex and base drag =0 and the thickess = 6% for the NACA 64-006 airfoil:

$$C_d = C_{DF} + \frac{4}{\sqrt{M_\infty^2 - 1}}\left[\frac{4}{3}\frac{t}{c}\right] + C_{dB}\left(\frac{S_B}{S_{ref}}\right)$$

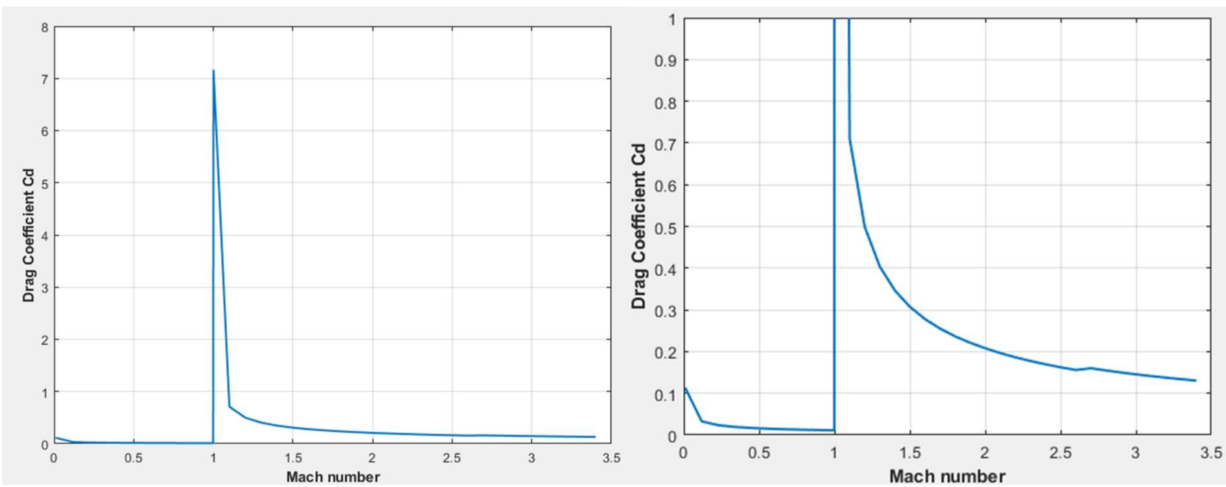


Fig 23. Initial drag estimation as function of Mach number

1.6 Lift drag ratio

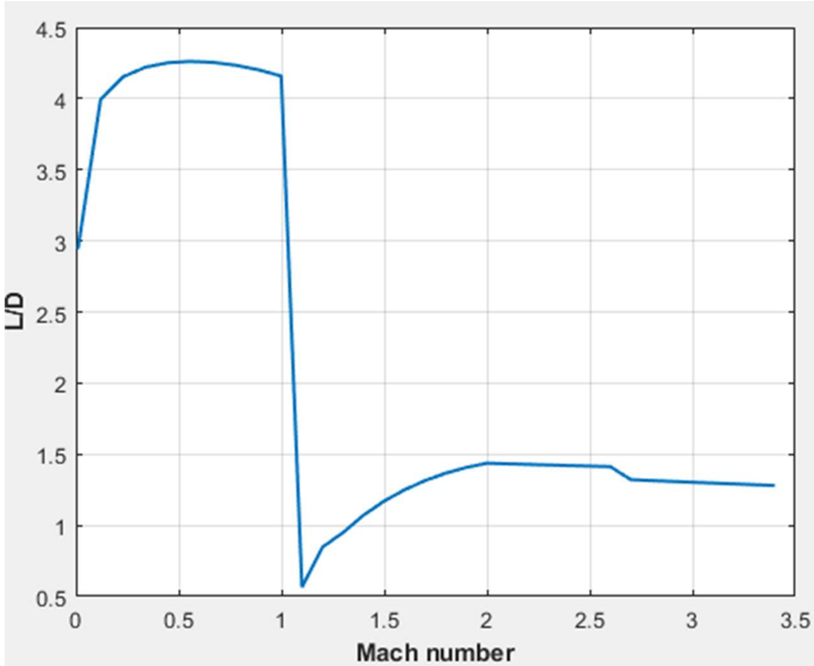


Fig 24. Lift drag ratio for delta wing 60 degree swept, 6 degree AOA & aspect ratio = 1.7

2. Configuration Layout

Based on the mission requirements and other supersonic configurations, to design a supersonic aircraft, the design's configuration must have delta wing, slender fuselage, ramjet propulsion, chine, or canard for stability and control. All the components are designed to minimize drag, when reaching the supersonic, the drag increase exponentially.

Wing configuration: Airfoil must be thin to reduce form drag in supersonic the thinner the structure, the higher the drag reduction. Nevertheless, there is a tradeoff between structural strength and drag reduction. Highly swept wing for wave drag reduction in supersonic speed. Even so, the sweeping wing has a few disadvantages, including high induce drag and a shallow lift curve slope. Delta wing was picked for many supersonic designs because swept wing may not generate enough lift for the aircraft, delta wing with bigger surface area which produces more lift. The delta wing helps to reduce twist and bending moment because pressure force is distributed along the swept edge not concentrated to the leading edge like the traditional subsonic straight wing. Additional, delta wing generate vortex lift along the longitudinal of the wing which help during low speed take off or landing.

Fuselage configuration: The thin fuselage has a high fineness ratio to reduce supersonic wave drag. The design should have a consistent cross-sectional area along its length, and the fuselage should be smaller at the connected wing region to keep the plane's total cross-sectional area as uniform as feasible. During the supersonic flight, the aerodynamic center of the aircraft shifts backward from .25 up to 0.5 of mean geometry wing chord. The aircraft tends to pitch down due to the longer moment arm of center of pressure force when get into the supersonic regime. To maintain the level flight, control surface must be deployed which creates more drag due to trim. So, another front surface area such as canard or chine need to be added to minimize the drag. The structure generates more lift at the front to counter the pitch down moment. According to published literature, the SR-71 Blackbird aircraft chose to adopt a chine rather than a canard, as seen in its brother, the XB-70, for a variety of reasons. Unlike the canard, the chine has the benefit of maintaining a smoother and more uniform cross-sectional area of the aircraft. A chine also has the advantage of being a fixed structure, which removes the requirement to install extra control surface systems, resulting in weight optimization.

Tail or stabilizer configuration: supersonic aircrafts have the horizontal stabilizer integrated with the delta wing. Vertical stabilizers are added on the back region of the wing.

Propulsion: to achieve supersonic flight, powerful and efficient engines that can perform well at high speeds and altitudes. Ramjet is a good choice for this purpose. The main benefit of using a ramjet engine is that it does not require a compressor or turbine which generates more drag in supersonic regime. The ramjet engine utilizes the high speed of the incoming air to compress the fuel and create propulsion.

Since the chine also contributes lift the aircraft, but chine cannot be modeled right now, so I choose the double delta wing configuration to simulate the lift and drag for the entire aircraft. However, we must consider that the center of pressure of the second delta wing is more aft than the chine. Since the chine or second delta wing of the model act as destabilizer which is bigger static margin for delta wing. So, the double delta wing model is more stable than the chine model.

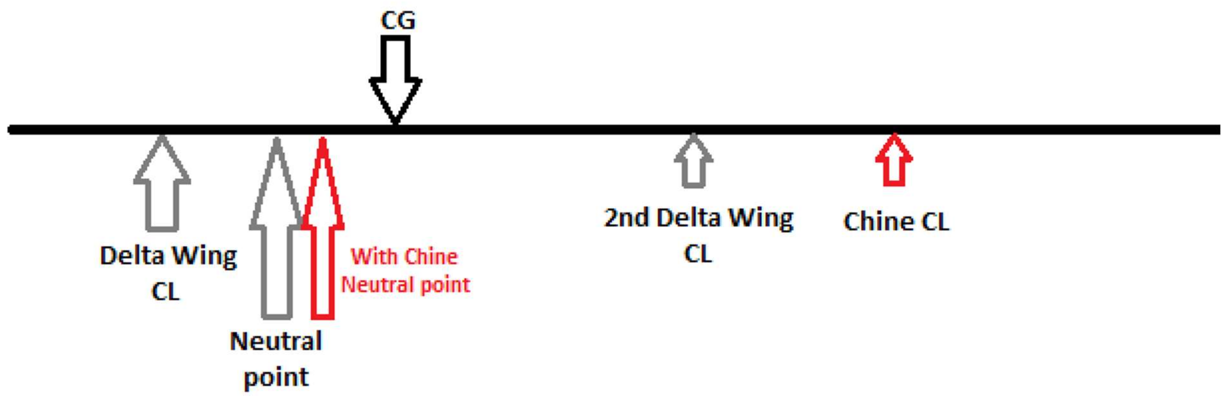


Fig 26. Neutral point visualization

3. Configuration Evaluation.

Aerodynamic properties drag and lift are also critical in aircraft design. To determine the drag of an airplane, divide it into different components (such as the wing, body, and tail) and calculate the drag of each component. Interference effects between components must also be considered. In Anderson's Aerodynamic textbook present multiple ways for making these calculations, which must be performed at various flight conditions (ground roll, landing, takeoff, rate of climb, etc.). The equivalent flat plate area approach, which effectively dimensionless the drag of each component in terms of the area of a flat plate with an equivalent drag, can be used to make the drag computations. This method provides a more visually appealing representation of the components. Lift is typically estimated in terms of lift coefficients, which are based on the manipulation of lift coefficients. It is required to estimate lift coefficients under varied operating situations (such as varying flap deflection angles). Existing aircraft design data can be utilized to define initial flap deflection angles and wing planform geometry values. However, to pick precise flap angles that fulfill the aircraft's operational needs, more comprehensive aerodynamic analysis of the wing and other aircraft components will be required later.

The SR-71 is a supersonic aircraft. Significant contributions caused by wave drag must be accounted. Also, the estimation of zero-lift (C_{Do}) and induced drag (C_{Di}) are necessary. Zero-lift (C_{Do}) is computed for each of the aircraft's major component elements and then added to obtain the 'whole aircraft' drag coefficient ($C_{Do} = C_f$ F Q[S_{wet}/S_{ref}]). Also, some amount of drag must be accounted for the C_{Do} estimation due to the some other components of aircraft (wheel, landing gear, antenna, etc). Induced drag-drag due to lift can be estimated as a function of $C_{L.}$ Then lastly, the estimation of wave drags which is the function of Mach number.

Based on literature research for supersonic aircraft. Because for thermal problem, aluminum alloys limit the speed below Mach 3. Titanium was the solution for the aerothermal problem. Supersonic aircrafts must have swept wing to reduce wave drag in supersonic flight. Also, the airfoil of the supersonic wing must be as thin as possible, To minimize the wave, drag at supersonic speed, aircraft design must follow couple rules such as thin airfoil, area rule, narrow fuselage, wingspan inside Mach cone. The SR-71 did follow all the rules. Navier Stoke's theorem which is based on viscous flow and computer flow simulations can be helpful after CAD model is developed

When flight over Mach 1, a Mach cone form, if the wing stays in the Mach cone, the flow over the wing is mostly subsonic which is lower drag. Mach cone is a function of Mach number. However, The delta wing configuration is seen in most of supersonic over Mach 2 speed such as Concord, and XB 70. For aircraft with delta wing, tail is usually combined with the wing, elevator and aileron are combined which is called elevon. With the speed requirement of Mach 3, our Aero team deicide to verify which swept angle and wingspan is suitable for the SR-71 design to limit down the wing configuration design.

Further process of evaluating the aerodynamic configuration of an aircraft or vehicle design aims to analyze its aerodynamic properties to determine its flight performance. This evaluation usually involves simulations using computational fluid dynamics (CFD) such as Ansys Fluent after the is completed conceptual CAD.

NACA 64-006 is picked for evaluation:

For takeoff (phrase 1), the speed is approximately 206 ft/s (Mach 0.183). The altitude is 0 ft, the density at sea level is 0.002378 slug/ft^3. So, the Re is 5.242414035780330E+07.

For first climb and refuel phrase (phrase 2), the speed is Mach, the altitude is 25000 ft, the density is 0.001065 slug/ft^3. So, the Re is 329441319.188381.

For second climb segment (phrase 3), the speed is Mach 2.075 The mean altitude for second climb segment is 50750 ft, the density at sea level is 3.6200E-04 slug/ft^3. So, the Re is 859412137.013169

For cruise back and forth (phrase 4, 5, 6 ,7), the speed is Mach 3, The mean altitude is 75000 ft, the density at sea level is 1.0776E-04 slug/ft^3. So, the Re is 859412137.013169.

For descent (phrase 8), the speed is Mach 1.5. The mean altitude is 40000 ft, the density at sea level is 5.8539E-04 slug/ft^3. So, the Re is 429706068.506584.

For landing (phrase 9), the speed is approximately 238 ft/s (Mach 0.211). The altitude is 0 ft, the density at sea level is 0.002378 slug/ft^3. So, the Re is 60445320.3032595.

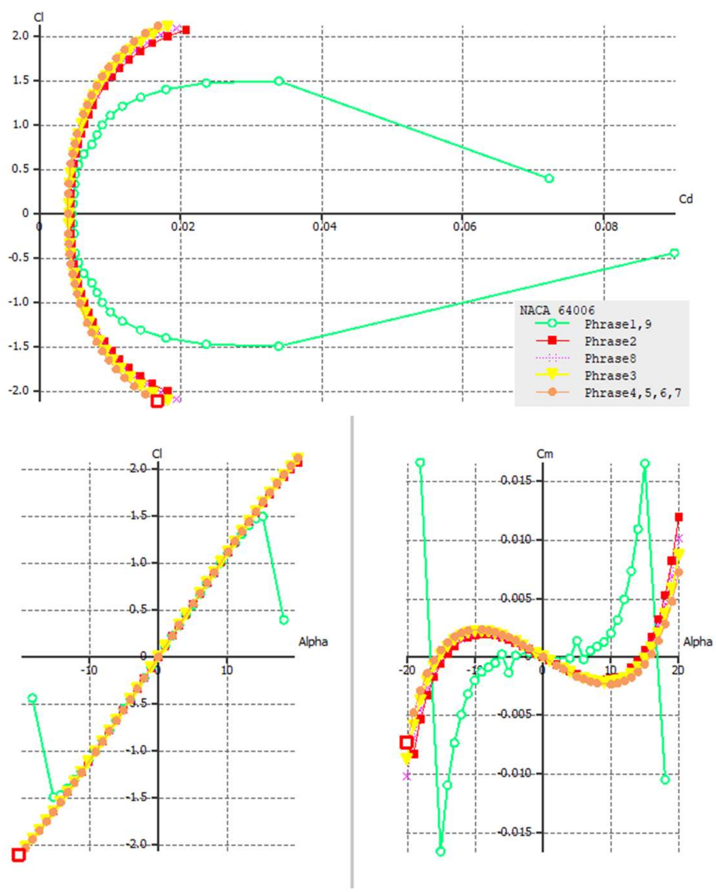


Fig 27. NACA 64-006 parameters for difference flight phrase from -20 deg to 20 deg

With the airfoil data, finite wing lift and induce drag coefficient can be estimated for every phrase by several method such as lifting line theory, vortex lattice, horseshoe vortex, and 3D panels.

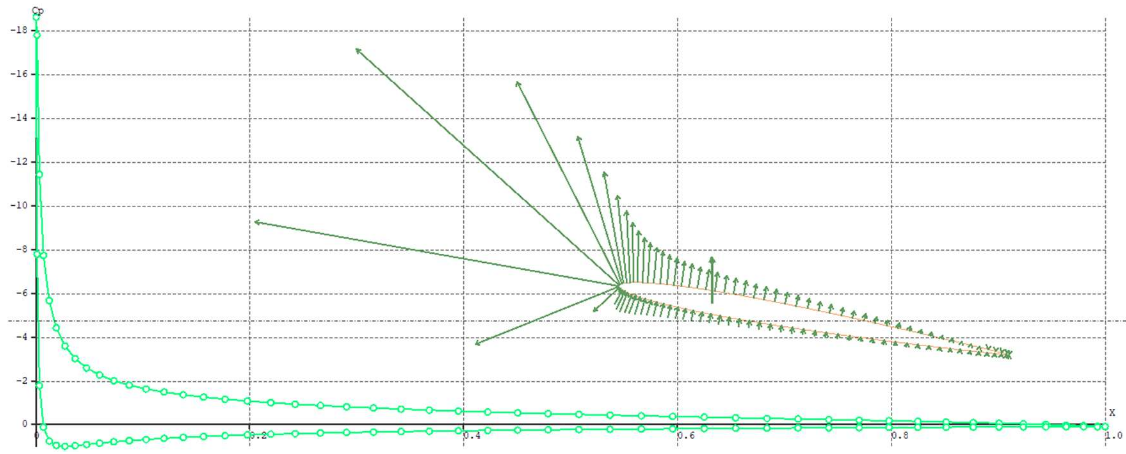


Fig 28. Pressure distribution on NACA 64-006, AOA = 10 deg

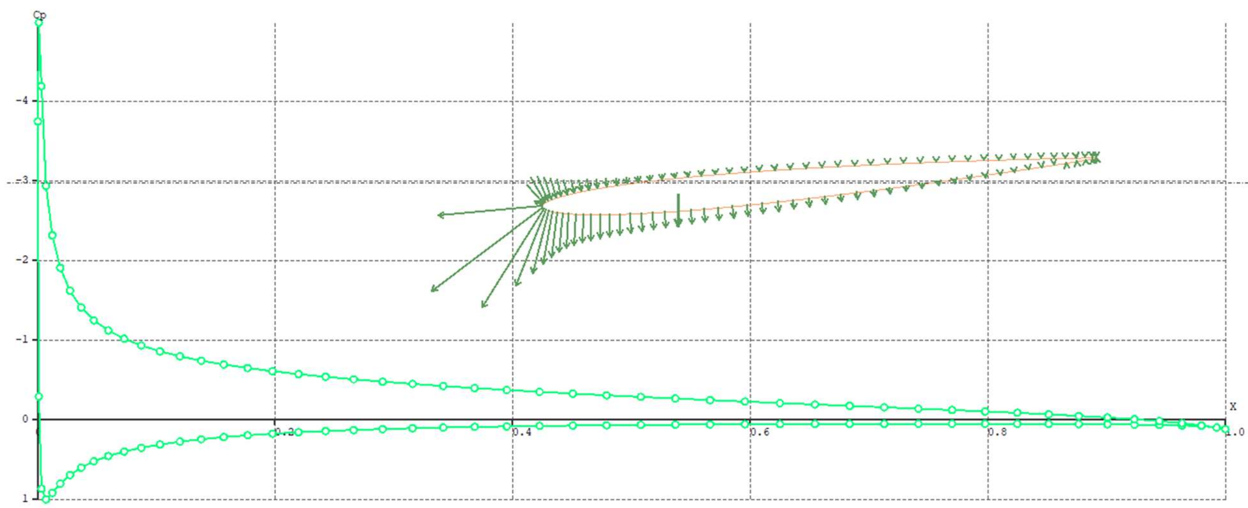


Fig 29. Pressure distribution on NACA 64-006, AOA = -5 deg

From the airfoil analysis, the center of pressure location is also the function of angle of attack. Center of pressure moves from approximately .25 to .21 as increasing or decreasing angle of attack where .25 is at ∓ 1 deg α. For zero angle of attack, the center of pressure is moving aft as the speed increasing. Zero AOA, center of pressure is obtained for every flight phrase:

Phase 1: $\bar{X}_{CP} = 0.12567$
Phase 2: $\bar{X}_{CP} = 0.40373$
Phase 3: $\bar{X}_{CP} = 0.43582$
Phase 4,5,6,7: $\bar{X}_{CP} = 0.49736$
Phase 8: $\bar{X}_{CP} = 0.4163$
Phase 9: $\bar{X}_{CP} = 0.134$

From synthesis, our team found the convergence point for 140592 lbs and given tau 0.05 at 2871 ft^2 planform area by hypersonic convergence method. From the defined planform area, I can process to aerodynamic evaluation.

Wing and chine of the SR-71 are simulated as double delta wing aircraft:

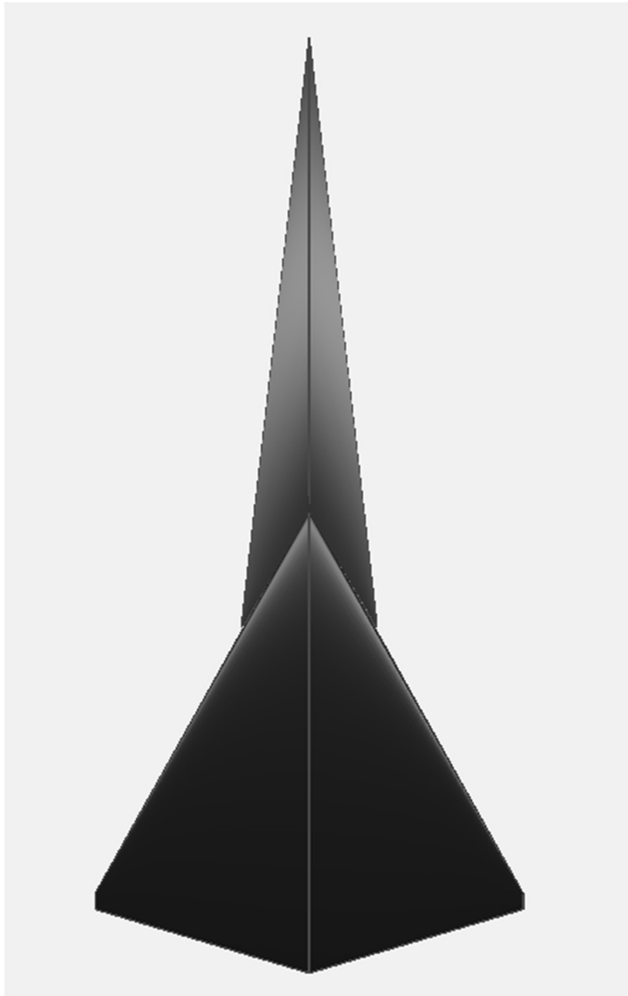


Fig 30. Double delta wing.

The given wing planform 2871 ft^2 yield 2 delta wings.
Front delta wing specifications:
Planform wing area = 693 ft^2
Wingspan = 19.8 ft
Root chord = 69.892 ft
Mean geometry chord= 46.595 ft
Aspect ratio = .566
Taper ratio = 0
Sweep angle = 73.691 deg

Main delta wing specifications:
Planform wing area = 2178 ft^2
Wingspan = 62.459 ft
Root chord = 67.551 ft
Mean geometry chord= 45.055 ft
Aspect ratio = 1.791
Taper ratio = 0
Sweep angle = 51.796 deg

The front and main delta wing are divided into 20x20 panel to utilize the vortex lattice method which allow to estimate the lift and induced drag for every flight phrase. Friction and wave drag can then be added to the total drag build up. Friction and wave drag are estimated based on Nicolai's methods.

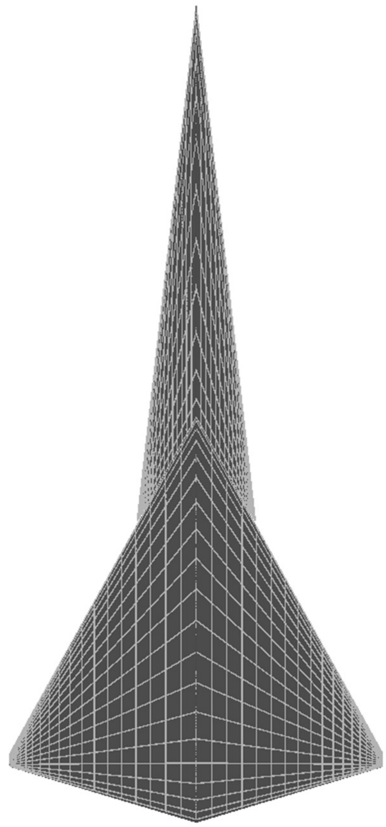


Fig 31. 20x20 panels double delta wing

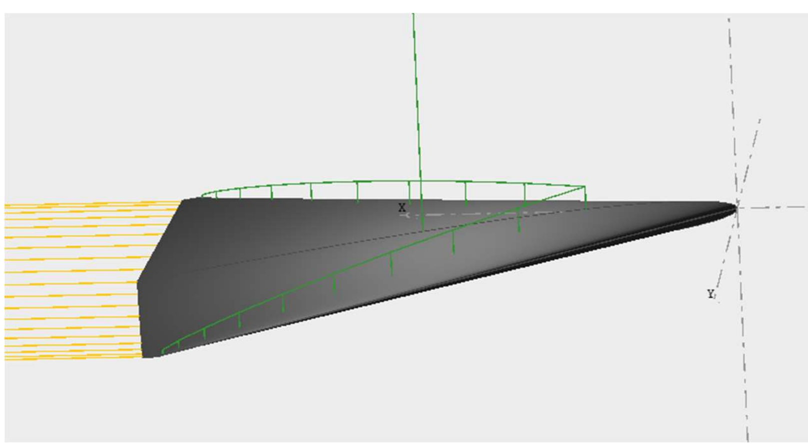


Fig 32. Lift and induce drag coefficient simulation main wing.

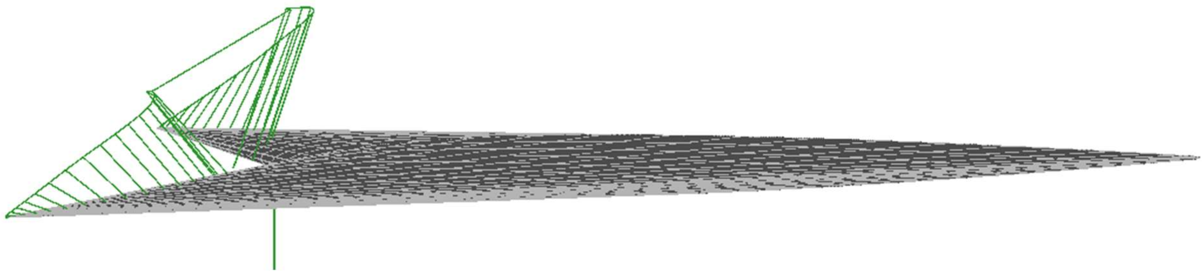


Fig 33. Lift and induce drag coefficient simulation front delta wing.

To initiate the takeoff process (phrase 1), the aircraft reaches a speed of around 206 feet per second, which is equivalent to Mach 0.183. The altitude at this point is zero feet, and the density of the air at sea level is 0.002378 slug per cubic foot. As a result, the Reynolds number (Re) for this stage is 5.242414035780330E+07

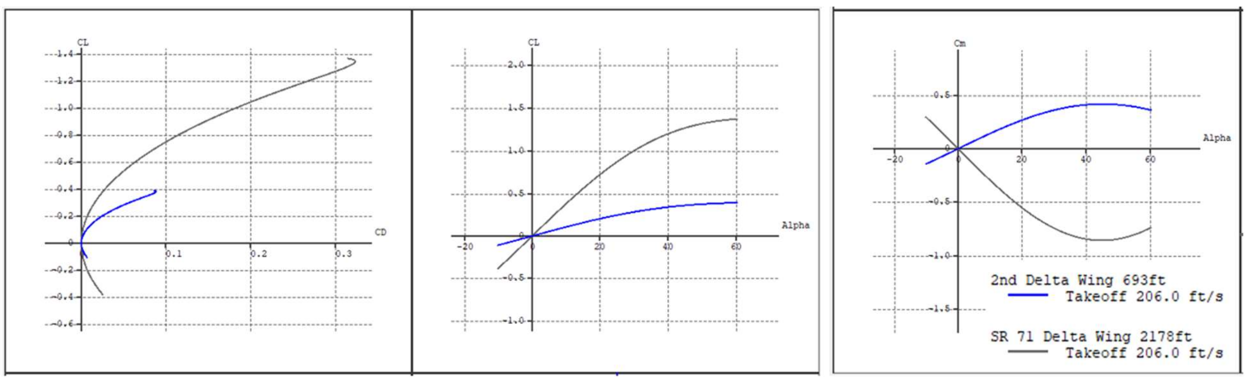


Fig 34. Drag polar, lift curve and moment curve for takeoff.

During the first climb and refueling phase (phrase 2), the aircraft travels at Mach speed 1.15, with an altitude of 25,000 feet and air density of 0.001065 slug per cubic foot. The Reynolds number (Re) for this segment is 329441319.188381.

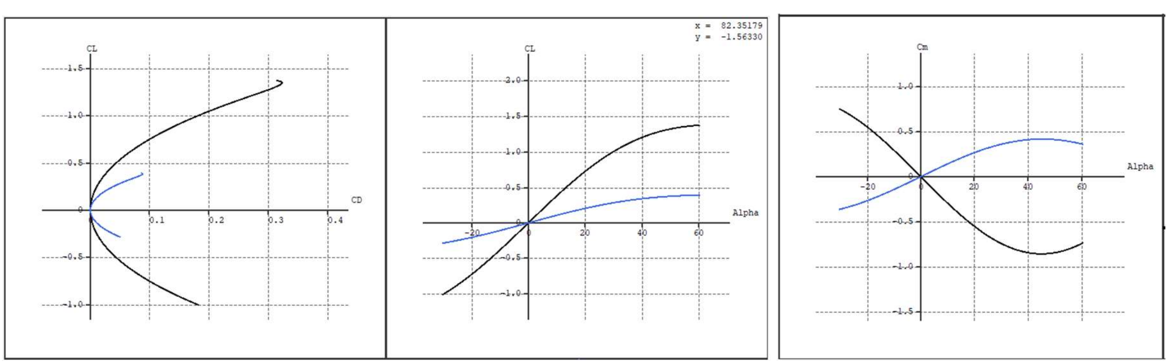


Fig 35. Drag polar, lift curve and moment curve for climb.

In the second climb segment (phrase 3), the aircraft attains a speed of Mach 2.075. The average altitude for this segment is 50,750 feet, and the air density at sea level is 3.6200E-04 slug per cubic foot. The Reynolds number (Re) for this phase is 859412137.013169.

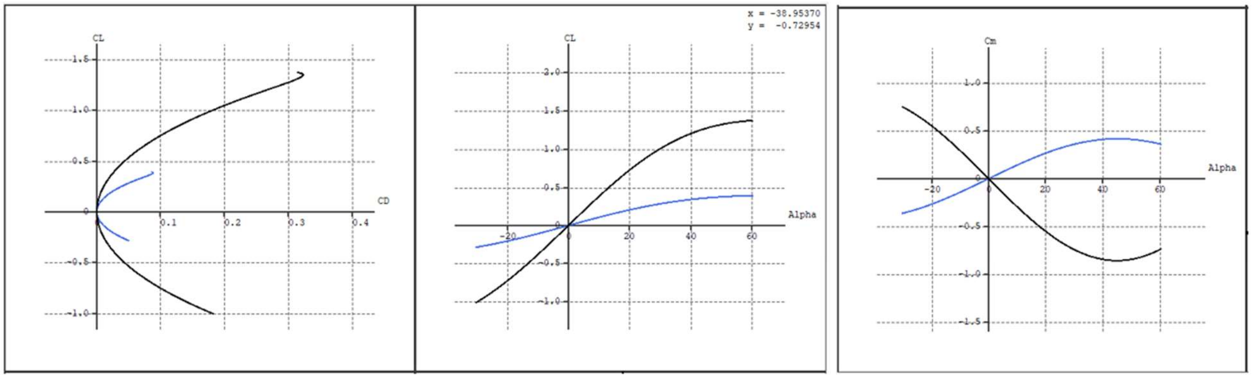


Fig 36. Drag polar, lift curve and moment curve for 2nd segment climb.

For the back-and-forth cruise (phrases 4, 5, 6, 7), the aircraft travels at Mach 3 with an average altitude of 75,000 feet, and the air density at sea level is 1.0776E-04 slug per cubic foot. The Reynolds number (Re) for this phase is 859412137.013169.

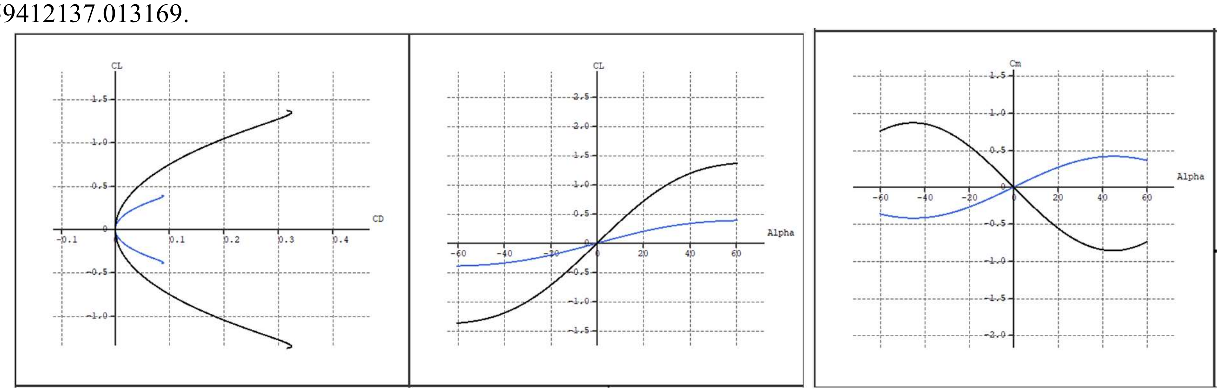


Fig 37. Drag polar, lift curve and moment curve for cruise.

While descending (phrase 8), the aircraft maintains a Mach speed of 1.5 with a mean altitude of 40,000 feet, and the air density at sea level is 5.8539E-04 slug per cubic foot. The Reynolds number (Re) for this phase is 429706068.506584.

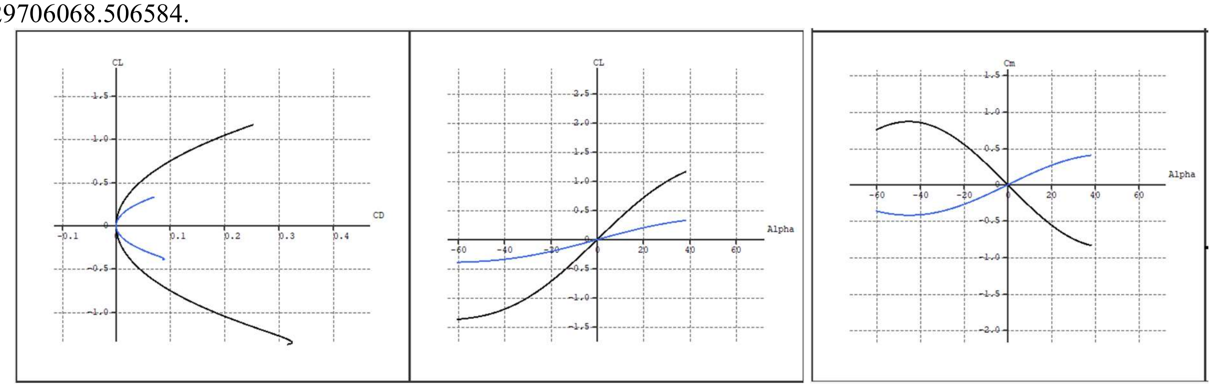


Fig 38. Drag polar, lift curve and moment curve for descent.

Lastly, during landing (phrase 9), the aircraft travels at a speed of approximately 238 feet per second, which is equivalent to Mach 0.211. The altitude at this point is zero feet, and the density of the air at sea level is 0.002378 slug per cubic foot. As a result, the Reynolds number (Re) for this stage is 60445320.3032595.

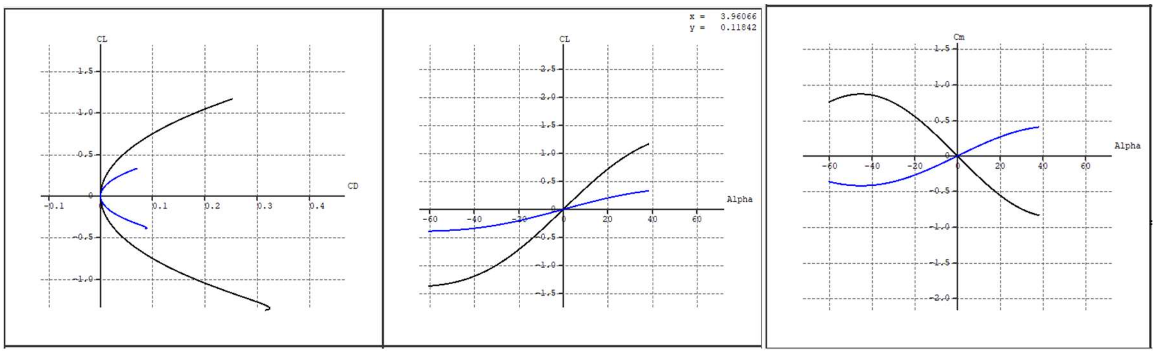


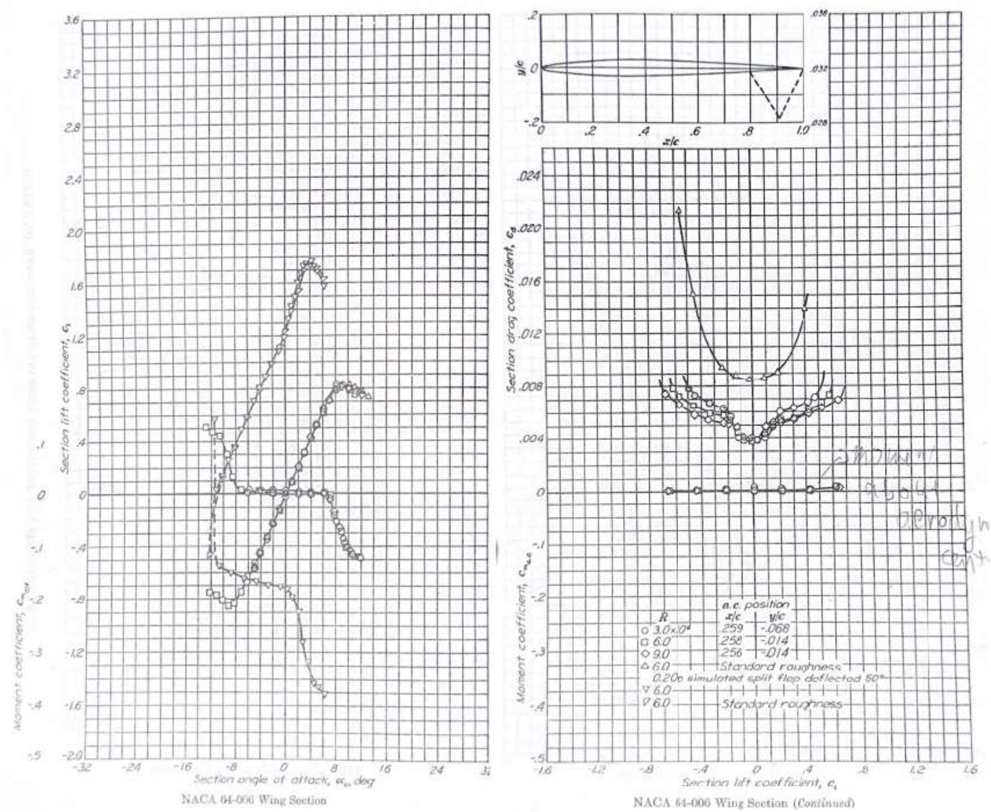
Fig 39. Drag polar, lift curve and moment curve for landing.

With induce drag estimations for every phrase, I can process to total drag build up estimation which consists of skin friction drag, induced drag, wave drag. Propulsion drag, fuselage drag, rudder drag are defined by propulsion team and geometry team, then added to the total drag build up. The out put then send to stability & control team and performance for further evaluation.

V. Conclusion

The process of conceptual design is useful for gaining insights into the characteristics of the SR-71. There are several methods to approach and converge the vehicle in the conceptual design stage such as Loftin, Roskam, AVD, Hypersonic convergence. Differences between the predicted values and the actual values can help identify trends in the development of modern aircraft.

Appendix



NACA 64-006 airfoil

Aerodynamic MATLAB:

```
clc
close all
clear all

%% Nicolai subsonic lift curve slope
AR =1.2:.1:1.8;
M_sub=linspace(0.01,.999,10);
beta = sqrt(1-M_sub.^2);
delta= 60;
for i = 1:length(beta)
    for j = 1:length(AR)
CL_alpha_sub(i,j) = ((2*pi*AR(j)) ./ (2 + sqrt(4 + AR(j)^2.*beta(i).^2.*(1 +
(((tand(delta))^2))./(beta(i).^2)))))/16;
    end
end
alpha = linspace(0,16,10);
CL = CL_alpha_sub(:,6)*alpha+4.3*(alpha/(180/pi)^2);
figure (1)
plot (alpha,transpose(CL),'LineWidth', 1.5)
xlabel('Angle of attack (α) [deg]', 'fontWeight', 'bold');
ylabel('Lift Coeficient (CL)', 'fontWeight', 'bold');
```

```
legend('M=0.0100', 'M=0.1199','M= 0.2298','M= 0.3397','M= 0.4496','M= 0.5594','M=
0.6693','M= 0.7792', 'M= 0.8891','M= 0.9990', 'Location', 'best')
grid on

%% Supersonic lift
%input data from excel
CL_alpha_sup = [0.050383316, 0.052902481, 0.04786415, 0.04635265, 0.044841151,
0.043329651, 0.041818152, 0.040306653, 0.038795153, 0.037283654, 0.037283654,
0.035096525, 0.033191915, 0.031515219, 0.030025554, 0.028691609, 0.027488936,
0.026398123, 0.025403536, 0.024492416, 0.023654238, 0.022880228, 0.022163007,
0.021496319, 0.020874818];
CL_alpha_subAR17=transpose(CL_alpha_sub(:,6));
CL_alpha_mach_AR17 = [CL_alpha_subAR17,CL_alpha_sup];
CL_alpha_mach_AR17 =CL_alpha_mach_AR17*14;
M_sup=[1.1 1.2 1.3 1.4 1.5 1.6 1.7 1.8 1.9 2 2 2.1 2.2 2.3 2.4 2.5 2.6 2.7 2.8 2.9 3
3.1 3.2 3.3 3.4];
M=[M_sub,M_sup];
figure(2)
plot (M,CL_alpha_mach_AR17,'LineWidth', 1.5)
xlabel('Mach number', 'fontWeight', 'bold');
ylabel('Lift Curve Slope (CLα)', 'fontWeight', 'bold');
grid on

%% Subsonic drag
C = 33.3; %mean geomertric chord
Swet=5729.46;%ft
Sref=1850;%ft
e = .7865;
[T,P,rho,c] = FeetStandardAtmosphere(0); %temperature, pressure, density, speed of
sound.
Viscosity =1.79E-5*(T/518.7)^1.5*((518.7+198.72)/(T+198.72));
Vinf = M*c;
Re= rho*Vinf*C/Viscosity ;

CF = zeros(size(Re)); %allocate CF
for i = 1:length(Re)
    if Re(i) < 5E5
        CF(i) = 1.328/(Re(i)^(1/2));
    else
        CF(i) = 0.455/(log10(Re(i))^2.58);
    end
end
CDF = CF*Swet/Sref;
CDo_sub = 1.2*CDF;
K=1/(pi*1.7*e);
CD_sub = CDo_sub+K*CL_alpha_mach_AR17.^2;
Table = [ M(1:10);CF(1:10);CDF(1:10); CDo_sub(1:10); CD_sub(1:10)];
%% Supersonic drag
Cdw = 4./ sqrt(M_sup.^2-1)*(4/3*.06); %wave drag co
Cdo_sup=CDF(11:35)+Cdw;
K_sup = sqrt(M_sup.^2-1)/4;
Cd_sup = Cdo_sup + K_sup.*CL_alpha_sup;
```

```
CD_mach=[ CD_sub(1:10) , Cd_sup];
figure(3)
plot (M,CD_mach,'LineWidth', 1.5)
xlabel('Mach number', 'fontWeight', 'bold');
ylabel('Drag Coefficient Cd', 'fontWeight', 'bold');
grid on
ylim([0 1])

%% Drag polar
LD = (CL_alpha_mach_AR17./CD_mach);
figure(4)
plot (M,LD,'LineWidth', 1.5)
xlabel('Mach number', 'fontWeight', 'bold');
ylabel('L/D', 'fontWeight', 'bold');
grid on

%% lift force
L = 1.8 * rho*238.15^2*1800
D =  .21 * rho*238.15^2*1800
```

Performance constraint MATLAB:

```
clc
clear all
close all
%% MAE 4350 Constrain plot based Gudmundsson book, 3/5/2023,
%
g = 32.2;%ft.s^2
%Mission requirement
S_LFL = 15000; %ft landing field length

%ASSUMPTION:
CDmin=0.016;
CLmaxL=2;
CLmaxTO = 1.8;
CLmax = 1.5;
CLmax0 = CLmaxL;CD_LDG = CDmin;CL_LDG = CLmaxL; %assume

u = 0.04; % rolling friction
Vinf_climb  = 3122.45; % V infinity climb ft/s
Vinf_cruise = 3226.667; % V infinity cruise ft/s
Vv = 119.5; %ft/s
Vstall = 305.55; %ft/s
TOGW = 142000; %lb
c = -1.1868; d = .9609; %Regression Line Coefficients
Swet = 10^(c + (d*log10(TOGW))); % Wetted area
h_ground=0; %altitude ft
h_cruise = 85000; % altitude cruise ft
[T_ground,P_ground,D_ground,c_ground] = FeetStandardAtmosphere(h_ground);% [Rankie], 
[ft*lbf/slug*R] , [slug/ft^3] , [ft/s]
[T_cruise,P_cruise,D_cruise,c_cruise] = FeetStandardAtmosphere (h_cruise);
k=.211904274; %Lift-induced drag constant
tau = 1; % sec Time for free roll before braking begins
```

```
A = D_ground*CLmax;
%% WING LOADING (WL) & THURST LOADING (TL)
n = 600;
TW =0.001:1:n;
WL_stall = Vstall^2/2*D_ground*CLmax; %Stall Constraint
WL_stall=linspace(WL_stall,WL_stall,n);
%WL_land = ((S_LFL/0.3)^(1/2)*D_ground*CLmaxL)/2.4; %Landing constranint
syms WL_land
eq = S_LFL == (0.01583 + 1.556*tau*(A/WL_land)^(1/2) + 1.21/(g*(0.605/CLmax0*(CD_LDG-
0.3*CL_LDG)+0.3)))*WL_land/A;
WL_land = vpasolve(eq, WL_land);
WL_land =linspace(WL_land,WL_land,n);
WL=0.001:1:n;
TW_TO = 1.21/(g*D_ground*CLmax*S_LFL)*(WL)+.605/CLmax*(CDmin-u*CLmaxTO)+u; %Take-off
Field Length Constraint
TW_Climb = Vv/Vinf_climb^2 + (1/2 *D_cruise *Vinf_climb^2)./(WL)*CDmin+k/(1/2
*D_cruise *Vinf_climb^2)*WL;
TW_Cruise = (1/2 *D_cruise *Vinf_cruise^2)*CDmin*(1./WL) + (k/(1/2 *D_cruise
*Vinf_cruise^2)).*WL;
plot(WL, TW_Climb, '-.r', WL, TW_Cruise, '--b', WL, TW_TO, '--m', WL_land, TW, '--k',
WL_stall, TW, ':', 'LineWidth', 2)
legend('Climb', 'Cruise', 'Takeoff', 'Landing', 'Stall', 'Location', 'best')
grid on
xlim([0,n+1]);
ylim([0,3]);
xlabel('Wing Loading (lb/ft^2)', 'fontWeight', 'bold');
ylabel('Thrust Loading (lb/lb)', 'fontWeight', 'bold');
```

Acknowledgments

The preferred spelling of the word “acknowledgment” in American English is without the “e” after the “g.” Avoid expressions such as “One of us (S.B.A.) would like to thank…” Instead, write “F. A. Author thanks…” *Sponsor and financial support acknowledgments are also to be listed in the “acknowledgments” section.*

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