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Design Guidelines for Supersonic Aircrafts in Civil Aviation

Tridib Banerjee Mr.
Nanyang Technological University, Singapore, official.tridib@gmail.com

Introduction

Designing an aircraft is an iterative process where multiple design constraints and market demands are to be satisfied simultaneously while hoping that eventually, all the parameters converge on a set of acceptable values. While there exists a multitude of established rules and regulations for designing a subsonic aircraft, it is difficult to even find empirical relations when it comes to supersonic large-scale aircrafts. The last successful supersonic flight in civil aviation was decades ago with technology and manufacturing methods have mostly become obsolete. This paper will set forth a comprehensive and a cohesive manuscript of guidelines for designing a supersonic civil aircraft in the modern era.

The paper will be split into two sections: pre-design guidelines and production guidelines. The first will address the less statistically dependent parameters like cost estimation, mission profile, environmental impact, and materials. The later will address the more computationally intensive parameters like weight estimation, wing design, tail design, fuselage design, and engine selection.

Pre-Design Guidelines

Mission Profile

The mission profile is a very vital aspect of aircraft design and it plays an even more critical role for a supersonic aircraft, even more so when it is made for civil aviation. Supersonic travel has always been restricted to harsher and more restrictive rules and regulations which specifically calls for a very well-thought-out mission profile that can take advantage of the aircraft's' supersonic speed while still obeying the regulatory authorities. The FFA (14 Code of Federal Regulations (CFR) 91.817 and Appendix B to Part 91) bans any civil aircraft from exceeding Mach 1 over land or close to the shores and while FAA is reconsidering the rules for supersonic aviation, it has made it clear that the speed restriction will still hold. The United Nations' Montreal-based International Civil Aviation Organization (ICAO) meanwhile is hoping to even tighten the rules and are backed by majority of Europe including, Germany, France, and Britain. It is needless to say, we must also consider a subsonic cruise even while developing a mission profile for a supersonic aircraft. Jimenez and Marvis (2005) proposed a mission profile for a modern SST based on a capacity of 175 passengers. The mission profile proposed in Figure 1 is a modified version of that profile. Jimenez and Mavris took into account the technological advancement year-after-year and optimized for highest supersonic cruise, thereby obtaining a Mach 0.95 subsonic cruise during the first lower altitude cruise-climb. While technology will definitely advance, it is still not prescriptible to cruise at the most critical part of the transonic regime where the drag divergence is maximum. Having established how critical subsonic cruise is, even for an SST, it is much more advisable to avoid cruising at highest drag divergence Mach.

This mission profile is also based on the viability of supersonic flight and is developed around the trans-Atlantic aviation routes where the flight can actually make a significant improvement to the existing transportation industry scenario. Although one must optimize the MTOW and GTOW based on their engine and payload capacity, certain target values for the initial stages of design. See Table 1.

Cost Estimation

Normally, researchers tend to route for parametric CERs models (Dean, 1995; Roy, 1999) for cost estimation as it is much more accurate compared to Analogous methods of estimations while being less time-consuming Detailed estimation methods. However, CERS don't perform well for an SST because there have been none under production for more than two decades. Therefore, the CERs that exist are either for subsonic aircrafts or are based on much more dated aircrafts which operated way below the desired economic efficiency. A much better approach would be that of Castagne et al. (2008). In their paper, they proposed that in case of targeting for the minimum weight, if the direct operating cost is used as the target optimization variable, much more accurate estimations can be made that will not be restricted by dated empirical relations. As we can see from Figure 2, when cost estimation is based on *Direct Operating Cost* minimization approach, we get a lower net cost estimation. This approach was also found to be much more accurate. Using a profile-3D render of a modern SST, we found results very similar to that obtained by Castagne et al. See Table 2.

As we can see, the DOC is much more accurate and even slightly more precise. The more important parameter is the cumulative error percentage which does not take into account the type of error (over-estimation or under-estimation) and gives a much better idea of error over multiple samples from the same data set. As can be seen, the weight-based estimation gives almost double the cumulative error (15.785%) compared to DOC based estimation (08.645%) which means it will produce poor results consistently. Readers can estimate their net cost per unit surface area of that particular material as,

$$cost = \mathrm{A_i} * \mathrm{C_i} * \mathrm{k_i}$$

where one can obtain the C_i from Figure 2 itself. One can get the k_i from industrial data sheets and can vary A_i accordingly to reach their desired design point performance.

Environmental Impact

Several factors come together to define the environmental impact of an aircraft and these are particularly important for high fuel consuming aircrafts like the supersonic ones. The chief ones are Sonic Booms, Engine Emissions, and High-Altitude Footprints.

Sonic booms.

Sonic booms are perhaps the most critical issue and their proper modeling is imperative to supersonic civil aviation. Sonic Booms can be significantly reduced with proper sizing of the aircraft as is eminent from Figure 3. This shows that smaller sized aircraft has significantly better sonic

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performance than that of their larger counterparts.

Near field.

The three critical parameters discussed here are the slenderness ratio, area partitioning, and altitude. As seen in NASA (2005) Technical Reports on sonic boom, the geometry of the aircraft strongly defines its shock imprint. Note that from here on, by area we shall mean equivalent area unless stated otherwise because, in near-field, the equivalent area is what matters. If we define the area distribution as

$$S_{eq} = 2\int_0^l kx^n dx$$

Where k is some scaling constant and x is an arbitrary axial length along the fuselage. We can see from Figure 4 that for a constant area and longitudinal nose length l (in this case it was taken as 70 ft), lower the value of n, slenderer the aircraft nose section will be. From Figure 5, we can see that these more slenderly designed aircraft nose sections produce a much more gradual shock front in the near field while recording a much lower 0.9 psf compared to that of $n = 2/3$ that reordered 1.2 psf peak pressure with a much steeper slope as well. These results were obtained for Mach 2.7 at 60,000 ft with a test aircraft weighing 460,000 lb and having a fuselage of 300ft long, which was divided into two sections: nose (70 ft), and body (230 ft). While it is advisable to have as aerodynamic a nose as possible, from Figure 6 where the nose area is represented as a fraction of total area which was kept constant, we can see that for a very much aerodynamic nose, say 9% the total area, it does not necessarily produce a better result in terms of sonic booms rather, in this case, where $n = 1/3$, it produced the worst result out of all the tested configurations. The final thing to consider would be the effect of overall slenderness ratio, that is the effect of the net area distribution. As demonstrated by Antonio Ferri (1970), the overall length indeed influences the sonic boom significantly and is a very critical parameter which must be sized to the best possible optimization. As we can see from Figure 7, increasing the overall length does affect the shock wave strength and its wave nature significantly but only up to a certain extent.

The test aircraft had a total gross weight of 460,000 lb. since the equivalent area is a function of the length, we can characterize the wave signature in terms of net weight loading as $\widetilde{w} = W/l^{n+1}$ where W i**s** in lb and l in ft. In this case, below $\widetilde{w} \approx 229$, increasing the length had no significant effect. The same trend was observed for gross weights of 340,000 lb and 230,000 lb. lastly for altitude, the fuselage equivalent area, in near-field is proportional to the altitude through dynamic pressure as

$$S_{eq} \propto \sqrt{M^2 - 1}\frac{W}{\rho v^2}$$

Now as altitude increases, the dynamic pressure decreases and thus, the required fuselage area for a particular slenderness ratio increases. As a result, higher the altitude, more the aircraft behaves like a blunt body in the near field. Therefore, it would be better not to go to an altitude beyond 60,000 ft. Thus, the target values for *Near-Field* sonic boom optimization can be summarized as tabulated below. The readers must note that these are only reference values and must always be further optimized to fit their respective design point better. See Table 3.

Far field.

When in far-field, the most critical criterion would be altitude. As experimentally verified by Harvey, Domenic, Vera, and David (1964), the peak overpressure in far-field could be approximated as

$$\Delta \mathrm{P_0} = \mathrm{K_r} \frac{\sqrt{\mathrm{P_a P_0}}}{\mathrm{H}^{3/4}} \left(\mathrm{M_\infty}^2 - 1\right)^{1/8} \mathrm{K_2} \left(\frac{\mathrm{d}}{\mathrm{l}}\right) (\mathrm{l})^{3/4}$$

Here all pressures are in *psf* and all length units are in *ft*. K_r usually varies from 1.8 to 2 depending upon the terrain and is usually taken as 1.8. K_2 is usually between 0.55 to 0.70, and a good guess would be 0.6. One can use this to size their aircraft and decide their cruise Mach and altitude accordingly. It is to be noted that unlike in near-field, for this case, the sonic boom decreases with altitude. Therefore, a comprise must be made to determine the design point altitude. Harvey et al also reported lateral shockwave recordings from 56 bomber aircrafts flying nearly at Mach 2 at an altitude of 61,000 to 66,000 ft as measured from the ground. These are plotted in Figure 8. As we can see, using curve fitting, an initial cubic relationship can be established for designers to gauge their aircraft's performance in low altitude cruise to better plot their intended flight path. The lateral peak overpressure can be defined as,

$$\Delta \mathrm{P_0} = \alpha \mathrm{R}^3 - \beta \mathrm{R}^2 - \gamma \mathrm{R} + \delta$$

$$\text{where}, \alpha = 1.725e^{-5}, \beta = 19.37e^{-5}, \gamma = 42.46e^{-3}, \delta = 1.331$$

Once the peak overpressure is known, one can use the loudness graph from Figure 9, as proposed by Kevin and Brenda (1991). It is to be noted that loudness changes with multiple parameters like whether the recording point is indoors or outdoors, the shape of the sonic boom, and presence of resonance/damping media like windows or panes or doors. The characteristics plotted in Figure 9 are for N-waves which is the most common sonic boom signature for almost all supersonic aircrafts and should suffice for most preliminary design purposes. The last critical factor for sizing the sonic boom's far-field signature is coefficient of lift. Even without optimization, just the absolute value of coefficient of lift can affect your sonic boom signature as reported by Xuan, Cheng, and Fang (2016) and shown in Figure 10. It is therefore advisable to design your aircraft with minimum co-efficient of lift, just enough to meet your economic and mission profile demands. Excess coefficient of lift is detrimental in terms of sonic boom signature. Initially, reports by NASA (Lindsay, & Domenic,1960) indicated that anything ΔP_0 below 1.51 *psf* was found to be tolerable however, the Congressional Research Service (Elias, Luther, & Morgan, 2018) has revised the tolerance level to below 1.0 *psf*. In either case, engineers designing a modern SST must target a sonic boom ΔP_0 of less than 0.5 *psf* to be market-friendly.

High altitude radiation.

High altitude radiation is also a really critical parameter that may restrict your cruise altitude and cruise time, thereby greatly altering your required cruise speed and your mission profile as a whole. The earth atmospheric shielding capabilities exponentially decreases with altitude. As compiled by Goldhagen (2000) and reported by O'Brien and Frieberg (1994) using a modified version of the LUIN-98F code (O'Brien, 1978), shown in Figure 11 are the detailed cosmic radiations as a function of altitude. According to the NRC dose standards 10 CFR part 20, it absolutely restricts crew (annual limit 5 rems) and especially pregnant woman (annual limit 0.5 rems) to pursue an occupation in aviation where the flight might fly above nearly 35,000 ft during the majority of the profile. The passenger limit, however, is the critical parameter. While NRC has a short-period hourly limit of 2 rems, the data of the effective dose rates shown in Figure 11 has an uncertainty of nearly 50 % as reported by Wilson (2000). While NASA LaRC is collecting new and much more precise data right now (Chee, Braby, & Conroy, 1998; Tume et al., 1996), a weighted estimate would restrict the altitude for passengers at nearly 55,000 ft. It is, however, important to stress that during the mission profile, for short periods, the flight might climb beyond this altitude, however, it is not advisable to have a steady-state cruise beyond 55,000 ft based on existing data.

Engine emissions.

Emission optimization is a design variable of real importance when it comes to supersonic flights, especially in the civil sector. In contrast to upper-troposphere, NO_x in middle-stratosphere, where supersonic jets fly, actually lends to catalyzed ozone destruction. As reported by Grooß, Brühl, and Peter, (1998) and Dameris (1998), a sample set of 500 supersonic flights flying at Mach 2.4 at altitudes of 60,000 ft to 70,000 ft with emission index of 15g NO_2 kg^{-1} can cause an annual global reduction of ozone anywhere between 1.5% to 0.5%, with lower polar stratosphere ozone decreasing by as much as 3%. While there are uncertainties, it is a very critical parameter and must be taken very seriously. Arthur A. Mirzoyan in 2010 did a very comprehensive optimization of emission based on multiple correlation models (CMs) (Schaefer, 2006; Schumann, 1997) and compared it with high-fidelity emission model (HF EM) (Lebedev, Secundov, Starik, Shepin, & Titova, 2005) to obtain the most accurate results and provided them for an optimized SSBJ (Supersonic Business Jet). The important results to extract from this research is the correlation between all the critical design variables and how to optimize for the least emission. Arthur reported that the best results were obtained when optimized for three simultaneous variables, engine emission, near-surface temperature, and engine noise. The values are tabulated in Table 4.

Here $\mathrm{D_{pv}}$ is defined as $\mathrm{D_{pv}} = \mathrm{EINO_{xcr}} \frac{\mathrm{W_{fcr}}}{\mathrm{v_\infty}}$. Therefore, when designing, we now know that sacrificing about 3% of range can lower our fuel consumption, and our emissions by nearly 29%. Even if not scaled linearly, this is enough to show the benefits of minor range scarification. This is why it is recommended to have an additional 3% tolerance in terms of range while designing.

Also, as found by the ESPR Project and reported by Sun and Smith (2017), an $EINO_{xcr}$ of 5 g/kg can be achieved by using a Lean-Premixed-Prevaporized combustor, artificial intelligence-driven NO_x feedback controls and Ceramics-Matrix-Composite (CMC) linear walls along with a net 6.1 dB of decrease in noise by integrating a mixed ejector and swept/leaned stator blades in the bypass duct. Furthermore, the ESPR project (Fujitsuna & Tsuji, 2004; Shinozaki, Natsumura, Kobayashi, Arai, & Nakajima, 2004; Takahashi, (2001) was also able to achieve a 29% reduction in CO_2 emissions as well. For this, (compared to 1990s technologies), they recommended bringing the working line closer to the surge line, incorporating Titanium-Matrix-Composites, (TMC), Ceramics-Matrix-Composites (CMC), and Titanium-Aluminide (TiAl) into the fan rotor, turbine shroud, turbine blade and the turbine shroud support for nearly 30% of weight reduction, lowering the tip clearance by using Thermal-Barrier-Coating (TBC) on the High-Performing-Turbine (HPT), and reducing the cooling airflow by 56% with the help of higher loaded turbine and currently available advance flow controls.

Materials

The last important pre-design parameter would be the materials. For supersonic transports, the chief properties of a material to consider would be strength to weight ratio, tensile strength and associated failure properties, fatigue strength and associated failure properties, low-speed impact strength, fracture toughness, notch sensitivity, machinability, formability, resistance to crack propagation, resistance to both stress creep and thermal creep, and also its debonding/delamination properties. One must also understand the main environmental conditions that the materials need to withstand like moisture, fluid, temperature and radiation exposure. Huda and Edi (2013) have comprehensively analyzed the modernization and advancement of material science since the Concorde. See Table 5.

However, the designer can choose different materials if one understands the criterion needed. Shown in Figure 6 are some of the main criterions for critical components.

In the initial stages, mainly for the fuselage and wings, the biggest weight contributors, one can also gauge the change in weight of the component due to change in material while owing to same structural loading as:

Tension: $\frac{W_a}{W_b} = \frac{\rho_a}{\rho_b}\frac{\sigma_{yb}}{\sigma_{ya}}$

Compression: $\frac{W_a}{W_b} = \frac{\rho_a}{\rho_b}\left(\frac{E_b}{E_a}\right)^{1/3}$

Bending $\frac{W_a}{W_b} = \frac{\rho_a}{\rho_b}\left(\frac{\sigma_{yb}}{\sigma_{ya}}\right)^{1/2}$

However, these are just for estimation in order to aid material selection and not for exact evaluation. That has to be done later on while considering all phenomenon like torsion, fracture and crack propagation, fatigue life, thermal stability, etc.

Production Guidelines

Weight Estimation

This is a really iterative and statistically sensitive process which can easily stray into non-optimal or non-realistic solutions if not dealt with caution. Initially theorized by Sobieszczanski-Sobieski and Haftka (1997), expanded through non-linear regression and neural networks by NASA (Patnaik, Coroneos, Guptill, Hopkins, and Haller, 2004), extended to supersonic aircrafts by Xue, Khawaja, and Moatamedi (2014) and statistically simplified into an 8×4 optimization problem by Joiner, Zahra, and Rehman (2018), we will expand upon the following two models:

Mach number.

a_1 $-(.00002914 * W_e)$
a_2 $-(.0000293 * MTOW)$
a_3 $+(.139 * l) + (.2392 * b) - (.2358 * h)$
a_4 $-(.0001942 * S)$
a_5 $+(.008393 * WL)$
a_6 $+(.2991 * LDR)$
a_7 $+(.000000773 * MTOW * h)$
a_8 $+(4.151 * (10^{-9} * MTOW * S))$
a_9 $-(.001991 * l * b)$

$$\mathrm{Mach} = -5.6143 + a_1 + a_2 + a_3 + a_4 + a_5 + a_6 + a_7 + a_8 + a_9$$

Range.

a_1 $-(.11526 * W_e)$
a_2 $+(.08326 * MTOW)$
a_3 $-(28.185 * l) - (227.287 * b) + (89.661 * h)$
a_4 $-(.64534 * S)$
a_5 $-(7.6520 * WL)$
a_6 $+(115.9715 * LDR)$
a_7 $-(.0000002481 * MTOW * W_e)$

$a_8 \quad +(.002516 * W_e * b)$
$a_9 \quad -(.0003904 * MTOW * b)$

$$\text{Range (nm)} = 8041.429 + a_1 + a_2 + a_3 + a_4 + a_5 + a_6 + a_7 + a_8 + a_9$$

Where all weights are in *lbs*, length in *ft*, area in *ft*^{2} and wing loading in *psf*. Joiner's model is restricted to a service ceiling of 50,000 ft and an L/D soft limit of 3.05. As we have established previously (in case of altitude) and will establish later on (in case of L/D), this is definitely not the case for modern supersonic transport aircraft. Therefore, we will modify and establish other empirical relations to suite our mission demands. If we look at the trends for supersonic transport aircrafts from the last half a century in Figure 13a-c, in general, the range per take-off weight as well as length per take-off weight, both increased while the take-off weight per max cruise Mach decreased. Therefore, once we have a desired Mach, wingspan, lift to drag ratio, wing area and length, we can use these empirical relations to estimate the initial guess values.

$$\Delta\frac{\text{MTOW}}{M_\infty} = -4.1\times10^3\Delta t,\ \Delta\frac{l}{\text{MTOW}} = +0.033\times10^{-3}\Delta t,\ \Delta\frac{\tilde{R}}{\text{MTOW}} = +0.033\times10^{-3}\Delta t.$$

Where all weights are in *lbs*, and lengths in *ft.* As we will see later on, the L/D ratio target value should be around 8. For length, if we look at the cabin volume against the number of passengers in Figure 14, we get

$$V_{cb} = -\,0.01998\,n_p^2 + 73.33\,n_p - 393.3$$

Chudoba, Coleman, Oza, and Czysz (2008) reported a very similar relationship as follows,

$$V_{cb} = -\,0.00507\,n_p^2 + 69.0\,n_p - 241.0$$

One can use this to estimate the cabin volume. For length, we must decide upon a slenderness ratio first. If we look at the slender-body wave drag equation

$$D_{wave} = \frac{2\rho_\infty v_\infty^2}{\pi}\int_0^l\int_0^l S''(x_1)\,S''(x_2)\,\ln|x_1 - x_2|dx_1dx_2$$

And solve for the minimum wave drag for a body with a maximum cross-sectional area, we get

$$C_{D_W} = \frac{9\pi^2}{8}\left(\frac{l}{d}\right)^{-2}$$

Therefore, as we can see, with an increase in slenderness ratio the wave drag decreases exponentially. In Figure 15, we can see the wave drag coefficient behavior against the slenderness ratio as well as the slenderness ratios of all the supersonic transport aircrafts that have gone or is very recently going into full-scale production. It is clear that with time, more and more designers started pushing for higher slender ratio. It is also eminent that the wave drag of an actual aircraft closely follows that of a Sear-Haack Body. A good target value would be ≥ 21. The diameter is not much of a sizable parameter since there's a limit to minimum aisle clearance as well as seat widths. It is understood that you would like to go more for a lined arrangement than a stacked one because of higher slenderness ratio. Practically, it should fall within the range of 2.3 (7.5) to 3.3(10.9) m (ft). One can define the cabin ratio as

$$V_{cb} = \frac{k\pi}{4}\frac{l^3}{s^2}$$

Where k is a geometric co-efficient varying between 1 to 0.92 depending upon the curvature of the fuselage. From here we can calculate the length of the aircraft. From there we can get our diameter, as long as it is within the previously established practical limit. The width and height of the cabin can be adjusted as long as it yields the diameter as its average value. For the height of the flight, we go back to Joiner's model. He reports an ideal height of 42.52 ft with the sensitivity as follows:

$\frac{\sigma}{2}$ **Shift**	**Max Speed (knots)**	**Max Mach**	**Service Ceiling (ft)**	**Range (nm)**
0.5 ft	2.67%	4.55%	61.18%	7.36%

For the wingspan and wingspan area is to be calculated later on under the wing design section, for now, an effective guess for cruise wing loading would be 56 to 76 psf depending on how slender the wings are. It is to be noted that we also want to minimize our wing area and therefore must opt for as high a wing-loading as structurally possible. Tabulated in Table 7 are some wing loading target values:

As for the wing area, it depends on the target L/D cruise value. Within the pragmatic limit, Figure 16 can be used for the initial estimation of the wing area. For, empty weight, it can be expressed as a fraction of MTOW. Under the mission profile section, a detailed breakdown of component target values is given. With tolerance, a good estimation would be

$$\text{MTOW} = 2.3 \text{ to } 2.4\ W_e$$

For Mach number, as Horinouchi (2005) reported, there are great material difficulties going beyond Mach 2.2. This can be seen in Figure 17. Horinouchi also reported that for trans-Atlantic flights, anything above Mach 1.6 is adequate. Recalling the sonic boom subsection, we know that the sonic boom peak overpressure strongly depends on the Mach number ($M_\infty{}^2$) as well as the altitude. However, the takeoff weight and also the fuel efficiency would have suffered. Henne (2005) reported that a reasonable value would be 1.8 to 1.9. All these values can be substituted in the Mach No. model to calculate the MTOW through an iterative process.

Wing Design

This is the single most critical parameter in the entire design process.

Practically it is possible to push a subsonic aircraft beyond Mach 1 but everything changes once we go beyond Mach 1.2. Starting with the airfoil selection, Creaven and Roy (2005) used computational fluid dynamics in near field, and computational acoustics in far-field, to simulate performances of 60 airfoils cruising at Mach 2.2 in an altitude of 60,000 ft with an ideal gas of molecular weight 28.966 kg/kmol and a specific heat capacity of 1006.43 J/kgK as fluid. As we can see from Figure 18, in the supersonic regime, diamond airfoils are clearly the more superior one, producing both – higher L/D as well as lower δP which means much higher efficiency as well as lower sonic booms. However, we must not forget the subsonic and transonic regimes.

Any supersonic transport aircraft in the modern world has to be capable of significantly more efficient cruise in all the three regimes compared to the aircrafts of the past and the diamond airfoil greatly suffers here. They have very poor subsonic lift and drag characteristics making them inefficient for takeoff, landing, climb and loiter. Their sharp edges are also very difficult to manufacture and at the same time it severely heats up, especially along the leading edge thereby making it extremely challenging to cruise for long time. In this paper we shall use the method proposed by Obispo (2009) and use a channeled airfoil based on NACA 66-206 series airfoil. The 6th series is chosen as this family of aerofoils was specifically designed for supersonic performance – desirable drag, critical Mach, and also maximum lift characteristics as reported by Abbot and Doenhoff (1959).

The 66-206 (UIUC Applied Aerodynamics Group, 2018) was specifically chosen for it particularly exhibits characteristics similar to that of a typical supercritical airfoil including a flat-topped portion to delay shocks from forming on the top surface as well as a cusped section on the bottom surface (near the trailing edge) to promote lift in the subsonic regime. As to why this was chosen over the NASA's supercritical airfoil, it is because, while both have the same drag characteristics, the supercritical airfoils are effectively thicker which markedly increases drag penalty. Also, while the drag rise occurs sooner for the 6th series (M 0.69 compared to M 0.8 for NASA's supercritical airfoil), the supercritical airfoil has a significantly worse negative pitch even at subsonic Mach due to rear loading, $C_m = -0.14$, compared to $= -0.09$ for the 6th series at M 0.72. This would lead to a need for larger control surfaces which is not desirable in a supersonic flight. The 6th series is also easier to manufacture which shall play a huge role in large-scale production. The earlier onset of drag divergence is also mitigated for the 66-206 as it is particularly designed to demonstrate a drag rise around M 0.81. The 66-206 also has a significantly lower overall drag coefficient. The pitching moment and drag characteristics shown in Figure 20 are based on experimental observations by Whitcomb in (1974), whereas the specific performance characteristics of NACA 66-206 shown in Figure 21 are based on the data reported by Van Dyke Milton (1952).

Another approach could be to use a supercritical airfoil-based wing with large delta extensions instead of traditional winglets as discussed by Gueraiche and Popov in 2018, or Herrmann, in 2008, however, in this paper, we shall implement a supersonic channel to further optimize its performance in supersonic regime.

Figure 22 is based on the results obtained by Obispo (2009). The key insight here is the breaking down of a prolonged bow shock at the stagnation zone into much smaller “bow-puffs” with significantly small and distributed stagnation points by the leading edge of the supersonic channel. This leads to an extremely significant reduction in induced pressure drag of the entire profile. On closer inspection, we find the wetted area to increase which should lead to higher skin friction drag which indeed occurs but the reduction in induced pressure drag more than mitigates the penalty due to friction drag. However, it was found that this configuration was detrimental in subsonic regime which is why the channel needs to be closed during the subsonic parts of the mission profile. It shall only be opened when the flight goes into supersonic cruise. In Figure 23 are shown the influence of the channel’s geometry on the lift and drag characteristics of the airfoil. As we can see, for an airfoil with a channel that has a sharp leading edge and height 12% of the airfoil thickness, we can get as high as 5.5 L/D during supersonic cruise. The difference is much more severe as the Mach number rises. Note that all these analyses are done at 35,000 ft. When at higher altitude, there should be a further increase in L/D value. Compared to the traditional ones, the channeled airfoil demonstrated 9% increase in L/D (considering finite wing effect) during supersonic cruise while reducing the drag by almost 20%.

Design point optimization.

For design point optimization and for calculation of aerodynamic performances, we must apply different approaches based on whether it is supersonic or subsonic regime. For further optimization, we divided the wing into three sections, an LREX (Leading Edge Extension Part) a conventional delta part and an Outboard part.

Subsonic regime.

First, each section was divided into two sub-sections: one that is generating both the potential and vortex lift and one that is generating only potential lift. For potential flow, slender wing theory is used. The wing is discretized into transverse sections and the flow around each section is modeled as a 2D flow with a flat plat suspended perpendicular to the free stream as shown in Figure 24 below. Using continuous vorticity distribution, it can be shown that the potential of the flow on either side of the plate is given by

$$\varphi(x, y, 0^{\pm}) = \pm v_{\infty} \sin \alpha \sqrt{\frac{b(x)^2}{2} - y^2}$$

The chordwise velocity component, as well as the total pressure and the difference in pressure between both sides of the wing, can then be defined as shown below. The net aerodynamic force is then given as ℵ and the drag and lift coefficients then consecutively become its components as shown below:

$$v_x(x, y, 0^{\pm}) = \frac{\partial \varphi(x, y, 0^{\pm})}{\partial x} = \pm v_\infty \sin\alpha \frac{b(x)}{2\sqrt{b(x)^2 - 4y^2}} \frac{\partial b}{\partial x}$$

$$p_t = p(x, y, 0^{\pm}) + \frac{1}{2}\rho\left(\overrightarrow{v_\infty} + \overrightarrow{v_x}(x, y, 0^{\pm}) + \overrightarrow{v_y}(x, y, 0^{\pm}) + \overrightarrow{v_z}(x, y, 0^{\pm})\right)^2$$

Assuming v_y, v_z to be small

$$p_t = p(x, y, 0^{\pm}) + \rho v_\infty v_x(x, y, 0^{\pm})$$

$$\Delta p = p(x, y, 0^-) - p(x, y, 0^+) = \rho v_\infty v_x(x, y, 0^+) - \rho v_\infty v_x(x, y, 0^-)$$

Substituting the expressing for $v(x, y, 0^{\pm})$

$$\Delta p = \rho {v_\infty}^2 \sin\alpha \frac{b(x)}{\sqrt{b(x)^2 - 4y^2}} \frac{\partial b}{\partial x}$$

$$\aleph = \int_0^c \int_{\frac{-b(x)}{2}}^{\frac{b(x)}{2}} \Delta p \, dy dx$$

In case of a triangular geometry $b(x) = 2x\tan\Lambda = \frac{xAR}{2}$ therefore,

$$c_{l_p} = 2\pi \tan\Lambda \sin\alpha\cos\alpha = {}^{\pi AR}/_{2} \sin\alpha\cos\alpha$$

$$c_{d_p} = 2\pi \tan\Lambda \sin\alpha^2 = {}^{\pi AR}/_{2} \sin\alpha^2$$

For vortex lift, we can use the Houghton and Carpenter correction where each section generates an additional aerodynamic force per unit length $d\aleph_v$ and the total aerodynamic force due to the vortex is its integral over the entire chord length as shown below. In this case, the value of C_{dp} is ≈ 1.95.

$$d\aleph_v = \frac{1}{2}\rho(v_\infty \sin\alpha)^2 \, b(x) C_{dp},$$

$$\int_0^c d\aleph_v = \frac{1}{2}\rho {v_\infty}^2 C_{dp} \sin\alpha^2 \, S$$

The lift and drag contribution by vortices will then be

$$c_{l_v} = C_{dp} \sin\alpha^2 \cos\alpha$$

$$c_{d_v} = C_{dp} \sin\alpha^3$$

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From here, we can formulate the following equations for our optimized wing as follows

$$A_{both} = \frac{1}{2}\frac{{C_r}^2(1-\lambda)^2}{\tan\ \Lambda}$$
$$A_{potential} = \frac{{C_r}^2\lambda(1-\lambda)}{\tan\ \Lambda}$$
$$b = \frac{C_r(1-\lambda)}{\tan\ \Lambda}$$
$$C_{l_{both}} = \left(\frac{\pi AR_{Vortex}}{4}\right)\mathrm{Sin}\ 2\alpha + \frac{1}{2}C_{Dp}\mathrm{Sin}\ 2\alpha\ \mathrm{Sin}\alpha$$
$$C_{l_{potential}} = C_{l_\alpha}(\alpha - \alpha_{L=0} - \alpha_i)$$
$$C_{d_{both}} = \left(\frac{\pi AR_{Vortex}}{2}\right)\mathrm{Sin}^2\ \alpha + C_{Dp}\mathrm{Sin}^3\ \alpha$$
$$C_{d_{potential}} = \frac{{C_l}^2}{k}\ \text{with polar break}$$
$$C_{l_{ref}} = \frac{A_{both}+A_{potential}}{S_{ref}}\left\{\frac{A_{both}}{A_{both}+A_{potential}}C_{l_{both}} + \frac{A_{potential}}{A_{both}+A_{potential}}C_{l_{potential}}\right\}$$
$$C_{d_{ref}} = \frac{A_{both}+A_{potential}}{S_{ref}}\left\{\frac{A_{both}}{A_{both}+A_{potential}}C_{d_{both}} + \frac{A_{potential}}{A_{both}+A_{potential}}C_{d_{potential}}\right\}$$
$$C_{l_{net}} = C_{l_{ref_{LRX}}} + C_{l_{ref_{delta}}} + C_{l_{ref_{outboard}}} + C_{l_{ref_{HT}}} + C_{l_{ref_{Flap}}}$$
$$C_{d_{net}} = C_{d_{ref_{LRX}}} + C_{d_{ref_{delta}}} + C_{d_{ref_{outboard}}} + C_{d_{0_{net}}}$$

Note that in all context, the α is different even for a single design point, and for each subsection. This is because of taper-effect. Due to having different taper, each section will experience a different local angle of attack.

Supersonic regime.

For this regime, we can use the supersonic shock theories for the slender body to obtain the necessary equations as

$$C_{l_{ref}} = \frac{A_{section}}{S_{ref}}\left(\frac{\pi}{2}AR_{section}\alpha\right)$$
$$C_{d_{ref}} = \frac{A_{section}}{S_{ref}}\left(\frac{\pi}{4}AR_{section}\alpha^2\right)$$
$$C_{l_{outboard}} = \frac{4\alpha}{\sqrt{M^2-1}}$$

$$C_{d_{outboard}} = \zeta_{SCC} \frac{4\alpha^2}{\sqrt{M^2 - 1}}$$
$$C_{l_{net}} = C_{l_{ref_{LRX}}} + C_{l_{ref_{delta}}} + C_{l_{ref_{outboard}}}$$
$$C_{d_{net}} = C_{d_{ref_{LRX}}} + C_{d_{ref_{delta}}} + C_{d_{ref_{outboard}}} + C_{d_{0_{net}}}$$

where $\zeta_{SCC} = 0.8$.

Specifications

Based on these criterions, after non-linear simultaneous iterative solving, the optimum values for such a wing configuration (Figure 25) achieved are tabulated in Table 8. This wing configuration was then solved for the previously proposed mission profile for optimal values of other critical design parameters to obtain performance values which are tabulated in Table 9.

Wing Loading

To test the results, we used the vortex-panel method at Mach 0.3 to simulate the wing-loading during the most weight-critical part of the mission profile, the take-off, at free stream α of 12° and the load distribution was in agreement. The lift and drag coefficients matched with great accuracy and the localized angle of attack due to the taper and twist varied as predicted, causing vortex lift to be the biggest lift contributor for this case, greatly benefitting from a high angle of attack as shown in Figure 26. From the local lift and drag distributions, it becomes more eminent how the outboard section contributes to significant lift generation while maintaining a very low drag penalty. If we plot the localized lift to drag ratio of the entire wing as a single collective unit, it becomes clear the purpose of the outboard part in the subsonic regime.

Thickness ratio optimization.

Based on your design point, your thickness optimization may vary. It is recommended and is a common practice to optimize the thickness ratio for the supersonic cruise. Note that here, for our specified wing geometry, it makes sense to only solve for the delta part when in supersonic regime. Based on the linearized perturbation velocity potential in steady supersonic flow and assuming we are concerned with the surface of the wing (z=0), at a very small angle of attack, we can determine the boundary conditions, using which, we can obtain a source distribution and ultimately a pressure distribution as shown below. It can then be solved for pressure distribution for both interior, and exterior to the Mach lines. The drag over the wing can then be defined and using superposition for pressure distribution previously obtained, we can then obtain a normalized drag equation for steady supersonic flow over the wing.

$$\varepsilon^2 \frac{\delta^2 \emptyset}{\delta x^2} - \frac{\delta^2 \emptyset}{\delta y^2} - \frac{\delta^2 \emptyset}{\delta z^2} = 0$$

$$\left(\frac{\partial\emptyset}{\partial z}\right)_{z=0} = w = \lambda_a v$$

$$\emptyset(x,y)_{z=0} = -\frac{1}{\pi}\iint_S \frac{w(\xi,\eta)d\xi d\eta}{\sqrt{(x-\xi)^2 - \varepsilon^2(y-\eta)^2}}$$

$$\frac{D}{q} = \int_S \frac{\Delta p}{q}\lambda_a dS$$

$$\frac{c_d\varepsilon}{\tau^2} = F_1(r,b) + \bar{m}F_2(r,b) + \bar{m}^2F_3(r,b)$$

Where, $F_1, F_2,$ and F_3 are determinable functions of (r,b). for variable thickness, assuming that this results in identical planform and span, the frontal area projection can then be solved to get the desired relation as follows.

$$A_f = 2\int_0^{c_r\epsilon_{LE}} t(y)dy = A_f = S\tau\left(1+\frac{2}{3}\right)\bar{m}$$

$$\tau' = \tau\left(1+\frac{2}{3}\bar{m}\right)$$

$$c_d' = c_d\left\{\frac{\left(1+\frac{2}{3}\bar{m}\right)^2 F_1}{F_1 + \bar{m}F_2 + \bar{m}^2F_3}\right\}$$

Therefore, for $\bar{m} = 0$, our supersonic drag c_d' is equal to the idealized drag. Thus, as we can see, it is best to keep the thickness constant throughout the wing, especially the delta part, and by extension, the LREX. This is discussed in great details by Henderson, Jr. (1952). The thickness of the outboard part can be varied according to design choice based on how you chose your design points and its lift-margins.

Tail Design

For supersonic aircrafts, tail design becomes more challenging as there is a serious demand for stability control but at the same time, the drag and weight penalty is also significantly higher than in subsonic regime and at the same time, the stability control power of the tail decreases with increase in speed and altitude.

Horizontal tail.

In subsonic regime, the total drag, the total lift and the induced drag can be defined as

$$D_H = \frac{1}{2}\rho_\infty {v_\infty}^2 S_H\eta_H\left(c_{d0_H} + \frac{1}{\pi AR_H\emptyset}c_{l\ H}^2\right)$$

$$c_{l_H} = \frac{1}{v_{e_H}\eta_H}\left(c_{l_w}\frac{x_{cg}-x_{ac}}{\bar{c}_w} + c_{m_w}\right)$$

$$D_{H_{ind}} \approx \frac{1}{2}\rho_\infty v_\infty{}^2 \frac{1}{\pi \emptyset \eta_H} D_{H_i}{}^2 \left(\frac{2}{\rho_\infty v_\infty{}^2 S}\left[x_{cg} - x_{ac}\right]\right)^2$$

and in supersonic regime, they can be defined as

$$D_H = \frac{1}{2}\rho_\infty v_\infty{}^2 S_H \eta_H \frac{4\alpha^2}{\sqrt{M^2 - 1}}$$
$$c_{l_H} = \frac{1}{v_{e_H}\eta_H}\frac{4\alpha_\infty}{\sqrt{M_\infty{}^2 - 1}}\left(\frac{x_{cg} - x_{ac}}{\bar{c}_w}\right)$$

Where η_H, v_{e_H}, and D_{H_i}, are defined as

$$\eta_H = \frac{v_H{}^2}{v_\infty{}^2}, v_{e_H} = \frac{l_H S_H}{S\bar{c}_w}, D_{H_i} = \frac{b_H MTOW}{AR_H v_{e_H}\bar{c}_w}$$

Cuerno-Rejado & Sanchez-Carmona (2006) published empennage correlations in which they listed empirical relations between horizontal tail parameters and fuselage, wing area, and free stream properties as tabulated in Table 10 and plotted in Figure 29.

These in conjunction with the previously defined relations can be used to decide the size of horizontal tail although one must consider the cons of both, poor deep-stall performance for T-tail and more wetted area for a conventional tail. Noting that a supersonic aircraft which generates most of its lift via the leading-edge vortices depends on high angles of attack, therefore the advisable choice would be to go with conventional tail when control is the deciding parameter. However, there are more critical parameters. Using the genetic algorithm as proposed by Rallabhandi & Mavris in 2008, we can see that in terms of sonic boom, in its own merit, the algorithm prefers T-tails the most. This can be reasoned as follows: The effective length of the aircraft is increased by T-tails and this could lead to the rearward shocks to coalescence thereby causing the boom to decrease. Again, that is not all. If we take range into account, we see that the canard easily outperforms the T-tail as shown in Figure 30. It was found that other than sonic boom performance, in terms of range, jet-take-off velocity, approach velocity, and static stability penalty, the canard was clearly superior.

It is therefore up to the engineer to decide which is the most important parameter for them. Unless sonic boom is absolutely uncompromisable, a canard is overall more advisable.

Vertical tail.

The main argument against vertical tails for supersonic aircraft is that its directional stability influence, that is their yaw control potential, decreases

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significantly with an increase in free stream incidence angle. This is evident in Figure 31. Colgren and Loschke (2002) discussed the criterion that limits the ability of an aircraft to function without its vertical tail. They enlist system components and their response time delay as tabulated in Table 11. This response delay adds up to nearly 130 milliseconds. This is a huge problem if the gust loading has a sharper gradient. This can be seen in Figure 32, where the system fails to respond to an amplitude-doubling β perturbations with a period of 200 milliseconds. Even for a perturbation with a period of 400 milliseconds, it couldn't prevent it from falling into dynamic instability. Only when the perturbation period was 800 milliseconds did the system respond in a controlled fashion to ensure both static and dynamic stability. This makes its obvious that if the designer chooses to eliminate the vertical tail, the limiting criterion would be a fast-enough control system.

Without the vertical tail, the shape of the fuselage and position of the wings becomes even more critical. As Brandon and Nguyen (1988) reported, an elliptical fuselage with higher curvature in the transverse plane, with high finesse ratio and nominal wing position shows the best overall stability performance. This is evident from Figures 33(a) and 33(b). Nicolosi, Della Vecchia, Ciliberti, and Cusati (2016) discussed in detail the isolated fuselage aerodynamics. For sizing, there are several design methods developed over the years, the most recent and robust one being VeDSC as demonstrated by Nicolosi, Ciliberti, Della Vecchia, Corcione, and Cusati (2017). For lift slope, it can be estimated as

$$\text{Subsonic } c_{l_{\beta_v}} = \frac{2\pi AR_v}{2+\sqrt{\frac{\varepsilon^2}{k^2}AR_v\left(1+\frac{\tan \Lambda_{v,c/2}{}^2}{\varepsilon^2}\right)+4}}$$

$$\text{Supersonic } c_{l_{\beta_v}} = \frac{4\beta}{\sqrt{M_\beta^2-1}}$$

$$\text{Where } \varepsilon = \sqrt{1 - M_\beta^2}$$

For the directional stability derivative, it is most accurately estimated by

$$c_{n_{\beta_v}} = K_{F_v}K_{W_v}K_{H_v}c_{l_{\beta_v}}\frac{l_v S_v}{bS}$$

Where K_{F_v} , K_{W_v} , and K_{H_v} are defined as $K_{F_v} = \frac{c_{n_{\beta_v}}(FV)}{c_{n_{\beta_v}}(V)}$, $K_{W_v} = \frac{c_{n_{\beta_v}}(WFV)}{c_{n_{\beta_v}}(FV)}$, and $K_{H_v} = \frac{c_{n_{\beta_v}}(WFVH)}{c_{n_{\beta_v}}(WFV)}$, (V) standing for vertical tail only configuration, (FV) standing for fuselage-vertical tail configuration, and so on. These correction factors can be estimated from their respective characteristic curves as shown in Figure 34(a-c), also, the discretization parameters Z_W, Z_h, r_f, b_{v1}, b_v, Z_{ftc}, and d_{fc} are elaborated in Figure 35.

Fuselage Design

Again, fuselage design is yet another critical parameter while designing a supersonic aircraft. However, unlike the wing design or tail design, this is a much more pervasive parameter which is why we have already discussed most of the parameters of fuselage design under other design processes like that of the wings, the tails, etc. The critical parameters left is the area distribution rule. Although many supersonic fighter jets do not obey the area rule, it is not because of its incredibility but rather the increase in engine thrust capabilities over the years which has allowed these aircrafts to just brute force their way through the sonic barrier. However, this remains a critical parameter while designing a supersonic transport aircraft where efficiency cannot be compromised. Area rule is the optimization of the increase in areal gradient in the path of perturbation propagation.

For a supersonic cruise, this line of perturbation propagation is the Mach line which is inclined to the axis of the aircraft with an angle $\mu = \sin^{-1}\left(\frac{1}{M}\right)$. This can be seen in Figure 36. A full surface integral along the x-axis in the interval $[0,2\pi]$ will then give us the total drag alone the entire Mach cone, and this oblique plane is where the gradient of area should be the least. Theoretically, this was first reported by Jones (1956), where he demonstrated how the higher-order harmonics of the area derivative Fourier expansion only contributes to drag and not physical dimensions and therefore must be minimized. Over the years, several methods have been developed for numerically solving the perturbation propagation like the adjoint method as discussed by Palacios, Alonso, Colonno, Hicken, and Lukaczyk (2012) in which we used to better match our area distribution of the equivalent body of revolution to that of a Sears-Haack body which is well established to have the lowest drag profile for a given geometric constraint as shown in Figure 37.

Engine Selection and Sizing

Foster, Saunders, Sanders, and Weir (2012) published the results from NASA's research on hypersonic propulsion. In their paper, they report the performance of different engine type at different Mach numbers as can be seen from Figure 38. As we can see, if we are to build a supersonic aircraft, for our given mission profile, the best choice would be TBCC (turbine-based combined cycle) turbojets. In terms of the bypass ratio, it greatly depends on the size and passenger capacity, and also the range. In case of a long-range supersonic transport aircraft, the better choice would be a turbojet engine with high bypass ratio. Brear, Kerrebrock, and Epstein (2006) completed a very comprehensive review of power-plant design constraints for supersonic aircrafts using GASTURB. They assume a fuel of heating value 43 MJ/kg and mixing efficiency of 50% (although this is not a significant parameter) and at cruise

Mach of 2. Shown in Figure 39 are his obtained results from optimization. As it is eminent, higher bypass eventually leads to lower TSFC which is critical for an SST aircraft. Also, since with decrease in cruise Mach, the TSFC also decreases, therefore any design point optimized value for cruise Mach 2 should provide an adequate margin for a mission profile with cruise Mach 1.9, and thus should be a good initial value. Figure 40 can be used for sizing. This shows the current TFSC FOM (Figure of merit) versus the best case possible. The effect of the TFSC on mass of fuel and maximum range possible can be estimated from Figure 41.

Conclusion

In this paper, we find out the best mission profile for a modern SST in civil aviation. We find out the advisable weight fraction of every component, their cost estimation, and their sensitivity to change. We discuss how to calculate and mitigate the environmental impacts, the sonic booms, both near and far-field, the high-altitude radiations, and the engine emissions. We discuss the materialistic properties needed for each part and the preferred material for different Mach values. We also define how to preliminarily size and scale materialistic properties. We define empirical relations for weight estimation. We do a detailed design and analysis of a hybrid supersonic wing applicable to all three regimes. We discuss airfoil modification. We discuss the implementation of a supersonic channel. we also discuss the effects of variable wing thickness ratio and the slenderness ratio. We discuss in detail the tail and fuselage design as well. We also discuss the sonic boom performance for different tail configurations. We discuss the stability as well and finally the engine selection, sizing and its effect on normalized mass of fuel over a given range.

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