Fundamental Design Optimization of an Innovative Supersonic Transport Configuration and Its Design Knowledge Extraction

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The generation of shock waves is inevitable during supersonic cruising, which results in the generation of wave drag as well as sonic boom on the ground. Some innovative concepts, such as the supersonic biplane concept and supersonic twin-body fuselage concept, have been proposed recently to reduce the supersonic wave drag dramatically. In this study, these two concepts are adopted, and then the aerodynamic and sonic boom performance of innovative supersonic transport (SST) wing-body configurations are discussed using numerical approaches. This study is performed to obtain design knowledge for the innovative SST using an optimization method. In this research, the number of design variables is limited to only three in order to obtain fundamental design knowledge of the innovative SST configuration. The three design variables are utilized to deform the wing section shape. The wing section shape of a Busemann-type-biplane/twin-body model is optimized under the conditions of a design Mach number of 1.7 and angle of attack of 2 degrees. The optimized results show the tradeoff relationship between lift-drag ratio and maximum overpressure of sonic boom distribution on the ground. To obtain detailed knowledge of the design space, analysis of variance and visualizations of response surfaces of objective functions are also performed.

Key Words: SST, Sonic Boom, Multi-Disciplinary Optimization, Biplane Wing, Twin-Body Fuselage

1. Introduction

The Concorde supersonic transport had an advantage in its cruising speed, which overwhelmed other conventional transports. However, the Concorde ended operations, being decommissioned in 2003, mainly due to the problems of economic efficiency and environmental burden. Recently, there are many attempts to realize a next-generation supersonic transport. One of them is to reduce strong shock waves by interfering with them. The reduction of shock waves will contribute to reducing wave drag and sonic boom, which would be beneficial for a low-boom/low-drag SST. In this research, a supersonic biplane concept and a supersonic twin-body concept are adopted to reduce supersonic wave drag.

1.1. Supersonic biplane concept

In this concept, the strength of the wave drag is successfully reduced by interfering with the shock waves between the biplanes.1)–4) According to Kusunose et al.,1) the wave drag at zero lift of the biplane airfoil is reduced by nearly 90% compared to an equal volume diamond-wedge airfoil in two-dimensional inviscid simulations. By simply splitting the diamond airfoil into two elements and positioning them oppositely, the Busemann biplane airfoil is composed. At lifted conditions, the above-mentioned concept can be expanded. This expanded concept is called Licher’s supersonic biplane concept. The Licher biplane,5) whose thickness-chord ratio of the lower element is bigger than the ratio of the upper element, provides better performance than the Busemann biplane at lifted conditions.

1.2. Supersonic twin-body concept

A twin-body fuselage concept6) has also been proposed by the present authors to reduce the wave drag due to the fuselage volume of an aircraft. The same kind of twin-fuselage concept has been investigated previously by Wood et al.,7) in which fundamental wind tunnel experiments were performed at a Mach number of 2.7. According to Jones8) and Kroo,9) the supersonic wave drag due to the volume of a fuselage body is expressed as follows:

\[ D_{\text{wave}} = qk \frac{V^2}{l^4} \]  \hspace{1cm} (1)

where \( q \) and \( k \) are respectively the dynamic pressure and a constant. The wave drag due to the volume is proportional to the square of its volume \( V \), while it is inversely proportional to the fourth power of the length of the aircraft \( l \). (Note: This equation does not consider interference drag between bodies.) Since the wave drag of the airplane’s fuselage is proportional to the square of its volume when the body length is fixed, if we split a large single-body fuselage into two individual small bodies, the wave drag of each split body will be reduced to quarter of that of the original. The total drag of the twin-body configuration, thus, becomes one-half of the original single-body fuselage under the constant-volume and constant-length conditions. According to Yamazaki and Kusunose,6) over 20% total drag reduction was achieved in an optimized twin-body fuselage when compared to the Sears-Haack (S.H.) single-body fuselage under the constraint of fixed fuselage volume. This has been achieved by reducing the interference drag between the bodies. The S.H. body.
1.3. Objectives of this research

The fusion of the two advanced concepts yields an innovative SST configuration, which is a biplane wing/twin-body fuselage configuration. It has been proven that the innovative wing-body SST configuration is aerodynamically effective for reducing wave drag of a large-sized, twin-body (about 400 passengers) fuselage. However, the L/D of the 400-passenger model is less than 5, while that of the Concorde is known to be as about 7–8 (at a freestream Mach number of 2.0). One of the major reasons is its larger fuselage volume than that of the Concorde. In this study, therefore, the biplane wing/twin-body fuselage configuration (Fig. 1) is designed with its half-sized body, and its aerodynamic/sonic boom performance is investigated. Currently, we are considering that the total weight of the SST (including its engines and tail wings) will be 350–400 tons, its lift coefficient 0.15–0.20 and its center-of-gravity 0–25% mean aerodynamic chord.

We conducted biplane wing section shape optimization to obtain design knowledge for the innovative SST configuration. The shape optimization is performed using three geometric design variables. Practical optimization results are obtained in this relatively simple optimization problem. In addition, the three-dimensional design variables space is visualized to obtain detailed design knowledge.

2. Computational Methodologies

In this section, the computational methodologies utilized are concisely introduced.

2.1. CFD approach

Three-dimensional supersonic inviscid flows are analyzed using an unstructured mesh CFD solver of TAS (Tohoku University Aerodynamic Simulation)-code. Compressible Euler equations are solved using a finite-volume cell-vertex scheme. The numerical flux normal to the control volume boundary is computed using the approximate Riemann solver of Harten-Lax-van Leer-Einfeld-Wada (HLLEW). The second-order spatial accuracy is achieved using the Unstructured MUSCL (U-MUSCL) approach with Venkatakrishnan’s limiter. The LU-SGS implicit method for unstructured meshes is used for time integration. Three-dimensional unstructured meshes are generated using the TAS-mesh package, which includes surface mesh generation applying an advancing front approach and tetrahedral volume mesh generation using the Delaunay approach. The high accuracy of this unstructured mesh CFD approach has already been confirmed in other literature.

2.2. Sonic boom analysis

The sonic booms on the ground are predicted using the nonlinear acoustic propagation solver of Xnoise, which was developed by the Japan Aerospace Exploration Agency (JAXA). An augmented Burgers equation is numerically solved using an operator splitting method that takes into account the effects of nonlinearity, geometrical spreading, the inhomogeneity of the atmosphere, thermo-viscous attenuation and molecular vibration relaxation. In this approach, initial (input) pressure distributions are extracted from CFD solutions on the lower side of the SST configurations (Fig. 2(a)). Then the propagation of the pressure distribution (Fig. 2(b)) to the ground (Fig. 2(c)) is solved using the augmented Burgers equation. We investigated the influence of the extracted position of the initial pressure distribution, which indicated that the extraction at two fuselage lengths below was sufficient for accurate sonic boom evaluations in this study.

2.3. Skin friction drag estimation

The skin friction drags of various SST configurations are estimated by introducing simple algebraic skin friction models. Assuming that the boundary layer along the body is fully turbulent, the skin friction drag coefficient can be estimated as:

\[ C_{Df} = C_f \frac{S_{wet}}{S_{ref}} \]

where \( C_f \) is the averaged turbulent skin friction coefficient on the wetted area of the body, and \( S_{wet} \) and \( S_{ref} \) are respectively the wetted area of the body and reference area. The skin friction coefficient for turbulent boundary layer conditions can be calculated using the Prandtl-Schlichting flat-plate skin friction formula:

\[ C_f = 0.455 \left( \frac{1}{\log_{10} Re} \right)^{0.58} (1 + 0.144M_{\infty}^2)^{0.65} \]

where \( Re \) and \( M_{\infty} \) are respectively the Reynolds number and freestream Mach number. The Reynolds number is given applying the cruise conditions of the Concorde (total length of

![Fig. 1. Design specifications of twin-body/biplane wing supersonic transport model.](image)

![Fig. 2. Sonic boom analysis.](image)
62 m). Since the speed of sound \((a_\infty)\) and kinematic viscosity \((\nu_\infty)\) at the altitude of 18,000 m are respectively 295.069 m/s and \(1.1686 \times 10^{-4} \text{ m}^2/\text{s}\) according to Handbook of Aerospace Engineering, \(^{27}\) the Reynolds number for the fuselage body is calculated as:

\[
Re = \frac{M_\infty a_\infty}{\nu_\infty} \approx 266 \times 10^6 \quad (4)
\]

The Reynolds number for the main wing is calculated in the same manner with its mean chord length (i.e., \(Re\) of 32.5 \(\times\) \(10^6\), mean chord length of 0.122 m). The skin friction drag coefficients of the fuselage and wing are separately estimated with the corresponding Reynolds numbers using Eqs. (2) and (3). In Kusunose et al., \(^{1,2}\) and Hu, \(^{28}\) predicted friction drags based on the algebraic skin friction models are compared with those based on viscous CFD computations. It has been concluded that the simple algebraic skin friction models are reasonably accurate for the prediction of friction drag in supersonic flows.

3. Validation Study

The validity of the present computational approaches is concisely discussed in this section. The sonic booms generated from the N-wave model (NWM) and low-boom model (LBM) at \(M_\infty\) of 1.58–1.59 are discussed. These are test case conditions for the D-SEND#1 project of JAXA. \(^{29,30}\) In its experiments, the sonic boom distributions were measured at about 3,500 m away from the models. In this validation study, the CFD analyses are performed at the same free-stream Mach number to extract the initial pressure distributions as shown in Fig. 3(a). Then the sonic boom propagations are solved to compare with the experimental sonic boom distributions, whose results are shown in Fig. 3(b). The fluctuations of the experimental data in the range of 0.03 < time < 0.05 s are considered to be the effect of atmospheric turbulences. Qualitative agreement with the experimental data can be confirmed for both NWM and LBM cases, which indicates the validity of the present CFD/sonic boom analysis approaches. The validity of the present CFD analysis method in supersonic flow was also confirmed by Umeda et al. \(^{22}\) through comparing the experimental data of NEXST-1.

4. Basic Shape Definition

In this section, the basic shape of the twin-body/biplane wing configuration is defined.

4.1. Definition of wing shape

The section airfoil thickness ratios of the Busemann biplane are set to 5\% of the chord length for both the upper and lower wings. For the Licher biplane, \(^{5}\) the sum of the thickness ratios of the upper and lower elements is set to 10\% of the chord length, so the sectional airfoil areas of the two biplane models are the same. The thickness ratio of the lower element is about 43.9\% larger than the upper element for the Licher biplane model. Unswept tapered biplane wing configurations are designed for a three-dimensional biplane wing in this research. The aspect and taper ratios are respectively set to approximately 7 and 0.25. The chord length of the main wing at the root section is approximately \(l/6\). Since the effectiveness of the wingtip plate in the Busemann biplane wing was demonstrated by Odaka and Kusunose, \(^{31}\) a vertical wingtip plate is arranged between the wings to increase the two-dimensionality of the flow around the biplane wing (Fig. 4). The inner side of the wingtip plate is a flat-plate shape and its outer side has a thickness distribution based on the S.H. body radius distribution (maximum thickness ratio of 3.36\%). For appropriate shock interactions, the vertical distance between the wings is shortened at the outer wing by adding a dihedral angle to the lower wing. The non-dimensional section airfoil shape is the same at all semi-span positions.

4.2. Definition of body shape

The cross-sectional area distribution of the twin-body configuration is set to that of a S.H. body, which is well-known as the supersonic single-body configuration with the lowest wave drag for specified volume and length. The volume of the twin-body is set to half of the previous 400-passenger model. \(^{11}\) The body length is set to six times that of the wing root chord length. The cross-sectional shapes of the twin bodies are deformed from circular (conventional S.H. body) to elliptical to reduce wing-body parasite drag (Fig. 5(a)). The twin-body/biplane wing configurations defined from Fig. 4 and Fig. 5(a) are discussed as Fig. 5(b). In Fig. 6, the pressure contours around the original Busemann bi-
plane/twin-body configuration are shown under the conditions of \(M_1\) of 1.7 and angle of attack of 2 degrees. Successful shock interactions between the wings can be observed. The total number of mesh points of this innovative SST configuration is approximately 2.6 million, which has been determined to achieve qualitatively accurate simulations of shock wave interactions.

### 5. Wing Section Shape Optimization

In this section, the wing section shape of the biplane wing/twin-body configuration is optimized to obtain design knowledge for the innovative SST configuration using an optimization method of the Kriging response surface model.

#### 5.1. Computational conditions

The aerodynamic performance is discussed at the design freestream Mach number of 1.7 and angle of attack of 2 degrees using inviscid CFD computations. The skin friction drag is estimated utilizing standard algebraic (turbulent) skin friction models based on the wetted area of an SST configuration. In the sonic boom propagation analyses, standard atmosphere temperature/humidity profiles are utilized. The fuselage length and the cruise altitude are respectively set to 62 m and 18,000 m, that are given from the flight conditions of the Concorde.

#### 5.2. Optimization method

A surrogate model-based global design optimization method that makes use of a Kriging response surface model is utilized. An ordinary Kriging surrogate model is used to construct the surrogate models of aerodynamic functions in the design variable space. Firstly, initial sample points are generated in the design variable space using a Latin hypercube sampling (LHS) method, and then these are evaluated using CFD computations. Using the information of the initial sample points, initial surrogate models are constructed. The search of a promising location in the design variable space is executed applying a real-coded multi-objective genetic algorithm to the surrogate models. The promising locations are explored using the criteria of expected improvement (EI). The EI function expresses a potential for improvement in design variable space considering both estimated function value as well as uncertainty of the surrogate model. The CFD computations are executed for the explored promising locations where EI is maximal, and then new surrogate models are created by adding its information. Using the iterative process described above, the accuracies of the surrogate models are efficiently increased around the promising locations in the design variable space.

#### 5.3. Design variables

The initial geometry is the Busemann biplane/twin-body configuration. The number of design variables is three in total, and these are utilized to modify the \(x\) and \(z\) coordinates of the mid-chord apex of the upper and lower wings, as shown in Fig. 7 and Table 1. The design variables can be varied in relatively wider ranges than general optimization problems to extract detailed aerodynamic design knowledge in this design optimization. The \(z\) coordinate of the mid-chord apex of the lower wing moves in the same distance/direction as that of the upper wing. The deformed sectional shape of the biplane is employed at all span positions. This definition enables a constant wing volume during the optimization process.

#### 5.4. Treatment of wing-body junction region

In this study, a wing section shape optimization is performed. The body near the wing is deformed using an unstructured dynamic mesh method based on the spring analogy system. It is difficult to directly deform grid points on a three-dimensional surface. Thus, the three-dimensional surface of the fuselage is mapped to a two-dimensional parameter space \(s-t\) (Fig. 8(a)). Then, the unstructured dynamic mesh method is applied in the two-dimensional space. Finally, the deformed two-dimensional grid is remapped to the three-dimensional space (Fig. 8(b)). Using this method, the wing-body junction can be deformed while preserving the outer shape of the original fuselage. Sufficient accuracy of the present mesh deformation method was confirmed by Ban and Yamazaki.
5.5. Objective functions

The maximization of $L/D$ and the minimization of maximum overpressure ($P_{\text{max}}$), which is the peak pressure value of propagated pressure distribution on ground, are the objective functions of this problem. The former corresponds to aerodynamic performance, while the latter corresponds to sonic-boom performance. The Pareto optimal solutions are searched by solving the multi-objective optimization problem.

5.6. Results and discussion

The number of initial sample points is set to 25, in which 24 points are generated using the LHS method. An additional single point corresponds to Busemann’s biplane. This number of initial sample points is considered to be sufficient for the present three design variables space. Next, 39 additional sample points are iteratively evaluated to observe detailed tendencies of non-dominated solutions. This number of additional sample points is considered to be sufficient to make accurate surrogate models in the three-dimensional space. Figure 9 shows the performance of the solutions obtained in the multi-objective low-boom/low-drag shape optimization. The tradeoff relationship between $L/D$ and $P_{\text{max}}$ can be observed. For the purpose of comparison, Licher biplane performance was also included in Fig. 9. It is interesting that the performances of the Busemann/Licher models are comparable to the Pareto optimal solutions obtained. Three optimal designs are selected for comparison and named as Compromise, Max$_{L/D}$ and Min$_{P_{\text{max}}}$. “Compromise” is a compromised solution selected from the Pareto designs. Max$_{L/D}$ has the highest value of $L/D$, while Min$_{P_{\text{max}}}$ has the lowest value of $P_{\text{max}}$. The aerodynamic/sonic boom performances of these designs at $M_\infty = 1.7$ and angle of attack of 2 degrees are summarized in Table 2.

![Wing shape modification.](image1)

![Grid deformation of wing-body junction region.](image2)

![Wing shape modification.](image3)

![Grid deformation of wing-body junction region.](image4)

### Table 1. Design variables.

<table>
<thead>
<tr>
<th>Design variable</th>
<th>Design range</th>
</tr>
</thead>
<tbody>
<tr>
<td>dv1</td>
<td>Upper wing midchord apex coordinate $X$ $\pm 0.15c$</td>
</tr>
<tr>
<td>dv2</td>
<td>Upper and lower wing midchord apex coordinate $Z$ $\pm 0.025c$</td>
</tr>
<tr>
<td>dv3</td>
<td>Lower wing midchord apex coordinate $X$ $\pm 0.15c$</td>
</tr>
</tbody>
</table>

![Figure 9. Pareto optimal solutions obtained.](image5)

### Table 2. Aerodynamic/sonic boom performance of representative SST configurations at $M_\infty = 1.7$ and angle of attack of 2 degrees.

<table>
<thead>
<tr>
<th>$C_L$</th>
<th>$C_{DP}$</th>
<th>$C_{DF}$</th>
<th>$L/D$</th>
<th>$P_{\text{max}}$ [Pa]</th>
<th>$P_{\text{max}} - P_{\text{min}}$ [Pa]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Busemann</td>
<td>0.1202</td>
<td>0.01197</td>
<td>0.01200</td>
<td>5.02</td>
<td>30.5</td>
</tr>
<tr>
<td>Licher</td>
<td>0.1514</td>
<td>0.01272</td>
<td>0.01200</td>
<td>6.12</td>
<td>34.4</td>
</tr>
<tr>
<td>Compromise</td>
<td>0.1379</td>
<td>0.01221</td>
<td>0.01200</td>
<td>5.69</td>
<td>32.4</td>
</tr>
<tr>
<td>Max$_{L/D}$</td>
<td>0.2115</td>
<td>0.01872</td>
<td>0.01201</td>
<td>6.89</td>
<td>42.4</td>
</tr>
<tr>
<td>Min$<em>{P</em>{\text{max}}}$</td>
<td>0.0650</td>
<td>0.01478</td>
<td>0.01200</td>
<td>2.43</td>
<td>28.1</td>
</tr>
</tbody>
</table>

Figures 10, 11 and 12 are respectively the non-dimensional section airfoil shapes of the representative designs, $C_D$ distributions of the designs in the symmetrical plane, and the pressure visualizations of the designs in the symmetrical plane. In Fig. 10, the $x$ coordinates of the mid-chord apex of the upper and lower wings of the representative designs are moved slightly in the inflow direction. Since the gap between the two elements of the biplane is fixed, the movement leads to a reduction in $C_{DP}$ due to better shock wave interference. It can be observed that the Max$_{L/D}$ has a thicker lower wing than the upper wing in Fig. 10. The thicker lower wing leads to a strong shock wave at the front/upper-side of the lower wing. Then, the strong shock wave leads to an increase in $C_L$ at the rear/lower-side of the upper wing. In Fig. 11, higher pressure is confirmed in the Max$_{L/D}$ model at the rear/
lower-side of the upper wing ($0.5 < x/c < 0.8$), which also propagates toward the ground, increasing $P_{\text{max}}$. Thus, $L/D$ is significantly affected by the thickness difference between the upper and lower wings. The Max$_L/D$ and Compromise models have larger $C_{\text{DP}}$ than the Busemann model. The reason is that they have thicker lower wings and that leads to increasing both drag and lift. Since the higher pressure at the rear/lower-side of the upper wing leads to increasing $P_{\text{max}}$, the Max$_L/D$ and Compromise models have larger $L/D$ and $P_{\text{max}}$ than the Busemann model. In Fig. 10, it can be observed that Min$_{P_{\text{max}}}$ has a thicker upper wing and the x coordinate of the mid-chord apex of the lower wing is moved in the inflow direction more than that of the upper wing. For the former, the thicker upper wing, in other words, thinner lower wing, leads to lower pressure at the front/upper-side of the lower wing. As a result, the shock wave reflected at the rear/lower-side of the upper wing and propagated to ground is reduced. For the latter, two reasons can be considered. Firstly, the thicker upper wing generates strong shock waves at the front/lower-side of the upper wing, which leads to a larger shock-wave angle. To achieve better interference of the shock waves, the x coordinate of the mid-chord apex is moved in the inflow direction. Secondly, it can be observed that Min$_{P_{\text{max}}}$ has larger expansion for the lower wing at $0.35 < x/c < 0.4$ (Fig. 11). The expansion wave is impacted at the rear/lower-side of the upper wing and propagates to the ground. Then, it is confirmed that Min$_{P_{\text{max}}}$ has the weakest compression wave propagating toward the ground (although losing the model’s lift significantly). As a result, Min$_{P_{\text{max}}}$ has lower pressure at $0.73 < x/l < 0.83$, which can be seen in Fig. 13. In Fig. 14, the pressure waveforms at two body lengths below the representative designs are compared. We can see the largest peak at the middle of the distribution for the
Max.L/D model ($x \geq 35$). This is due to the effect of larger pressure variation propagating in the lower direction, which can be seen in Fig. 12(b). In Fig. 15, the pressure waveforms at the ground are compared, indicating a reduction in $P_{\text{max}}$ in the Min.Pmax configuration.

### 5.7. Visualizations of design variables space

An analysis of variance (ANOVA)\(^3\) is performed to analyze the significances of design variables, as shown in Fig. 16. This can be performed on the response surface models that were constructed in the optimization process. The areas of dv1, dv2 and dv3 correspond to the effects of the single design variables. Other areas correspond to the interactions of multiple design variables. It is confirmed that the $L/D$ is sensitive to the $z$ coordinate of the mid-chord apex (dv2), but is almost insensitive to the $x$ coordinates of the mid-chord apex of the upper wing (dv1) and of the lower wing (dv3). In other words, the $L/D$ deeply depends on the thickness difference between the upper and lower wings. It is confirmed that $P_{\text{max}}$ is sensitive to the $x$ coordinate of the mid-chord apex of the upper wing (dv1) and the $z$ coordinate of the mid-chord apex of the upper wing (dv2), but is almost insensitive to the $x$ coordinate of the mid-chord apex of the lower wing (dv3). It is also confirmed that $L/D$ and $P_{\text{max}}$ are significantly affected by the thickness difference between the upper and lower wings. Since the rear/lower-side of the upper wing is important for sonic boom performance, dv1 is also important for $P_{\text{max}}$.

Since we used only three design variables in this optimization, the distributions of objective functions can be visualized in the three-dimensional design variables space. The response surfaces of $L/D$ and $P_{\text{max}}$ are visualized in Fig. 17. The differences between the left- and right-hand-side figures are only the viewing locations, for better understanding of the three-dimensional space. The $x$, $y$ and $z$ coordinates are normalized by the ranges of the design variables. The smaller-size spheres indicate the positions of the Pareto optimal solutions. The medium-size spheres indicate the positions of the Busemann/Licher models (almost at the center of the space). The larger-size spheres indicate the positions of the three representative designs. Roughly speaking, the Pareto optimal solutions with larger $L/D$ stand along dv2,
6. Concluding Remarks

In this research, twin-body fuselage/biplane wing 200-passenger SST models have been discussed. Aerodynamic performance was evaluated using inviscid CFD computations. The skin friction drag was also considered utilizing standard algebraic (turbulent) skin friction models based on the wetted areas of SST configurations. Furthermore, the sonic boom performance of the innovative SST configurations was discussed. Low-boom/low-drag design optimization for the wing section shape of a Busemann biplane wing/twin-body configuration was carried out under the conditions of $M_{\infty}$ of 1.7 and angle of attack of 2 degrees using an optimization method of the Krigeing response surface model. Busemann/Licher biplanes were also investigated for comparison purposes. By the low-boom/low-drag design optimization, the tradeoff relationship between $L/D$ and $P_{\text{max}}$ could be observed. It was confirmed that the performances of Busemann/Licher models are comparable to those of the Pareto optimal solutions obtained. The $\epsilon$ coordinates of the mid-chord apex of the wings have a significant effect on the tradeoff relationship between $L/D$ and $P_{\text{max}}$. Simply saying, the model which has a thicker lower wing leads to larger $L/D$, while the model which has a thicker upper wing leads to smaller $P_{\text{max}}$. As the optimization result, the optimized SST configuration “Max $L/D$” achieved a 37.3% increase in $L/D$ and “Min $P_{\text{max}}$” achieved a 7.9% reduction in maximum overpressure compared to the Busemann biplane wing/twin-body configuration with the same wing volume. Furthermore, significances of design variables were analyzed using ANOVA, and the visualization of response surfaces of objective functions was performed to extract some important engineering design knowledge. It was clarified in these analyses that the difference in thickness between the upper and lower wings strongly affects aerodynamic and sonic boom performance. The major achievement of this paper was to extract important design knowledge for innovative SST configurations in a fundamental three design variable space, which can be utilized in future, in more detailed design optimizations.

Our final target performance is 8.0 in $L/D$ and 24 Pa in $P_{\text{max}}$ with the present fuselage volume. These target performance values are specified to satisfy twice the passenger capacity of the Concorde, to have better aerodynamic performance than the Concorde, and to have a $P_{\text{max}}$ one-quarter that of the Concorde. It is said that the Concorde’s performance was 7 in $L/D$ and 2 psf (96 Pa) in $P_{\text{max}}$. The optimized model has not yet achieved the target performance (i.e., the Compromise model has 5.69 of $L/D$ and 32.4 Pa of $P_{\text{max}}$). Although the present optimization achieved a large improvement in $L/D$, the three design variables were not yet sufficient to perform detailed shape optimization. Since we have obtained fundamental design knowledge of innovative SST configurations in this study, an investigation of the appropriate definition for the design variables will be performed. Then, the target performances can be realized by reducing body-body interference drag and wing-body interference drag of innovative SST configurations. In addition, optimizations under constant $C_L$ and with better sonic boom metrics, such as A-weighted sound exposure level, will be performed for more realistic aerodynamic/sonic boom evaluations in our future works. Furthermore, we will also discuss the structural design, flight stability, off-design performance and so on to demonstrate inclusive availability.

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References


30) Nak, Y.: Sonic Boom Data from D-SEND#1, JAXA-RM-11-010E, 2012.


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